Spacecraft design and dimensioning data

Chapter 4

4.1. Introduction

The design and dimensioning data that shall be taken into account by any User intending to launch a spacecraft compatible with the Ariane 4 launch vehicles are detailed in this chapter.

Figure 4.1.a shows the launch vehicle coordinate systems.

4.2. Safety requirements

The User is required to design the spacecraft in conformity with the CSG Safety Regulations.

R, L, T = Launch vehicle axes
X_f, Y_f, Z_f = Fairing axes (X_f, Y_f = Fairing vertical separation plane)
X, Y, Z = SPELDA, VEB axes

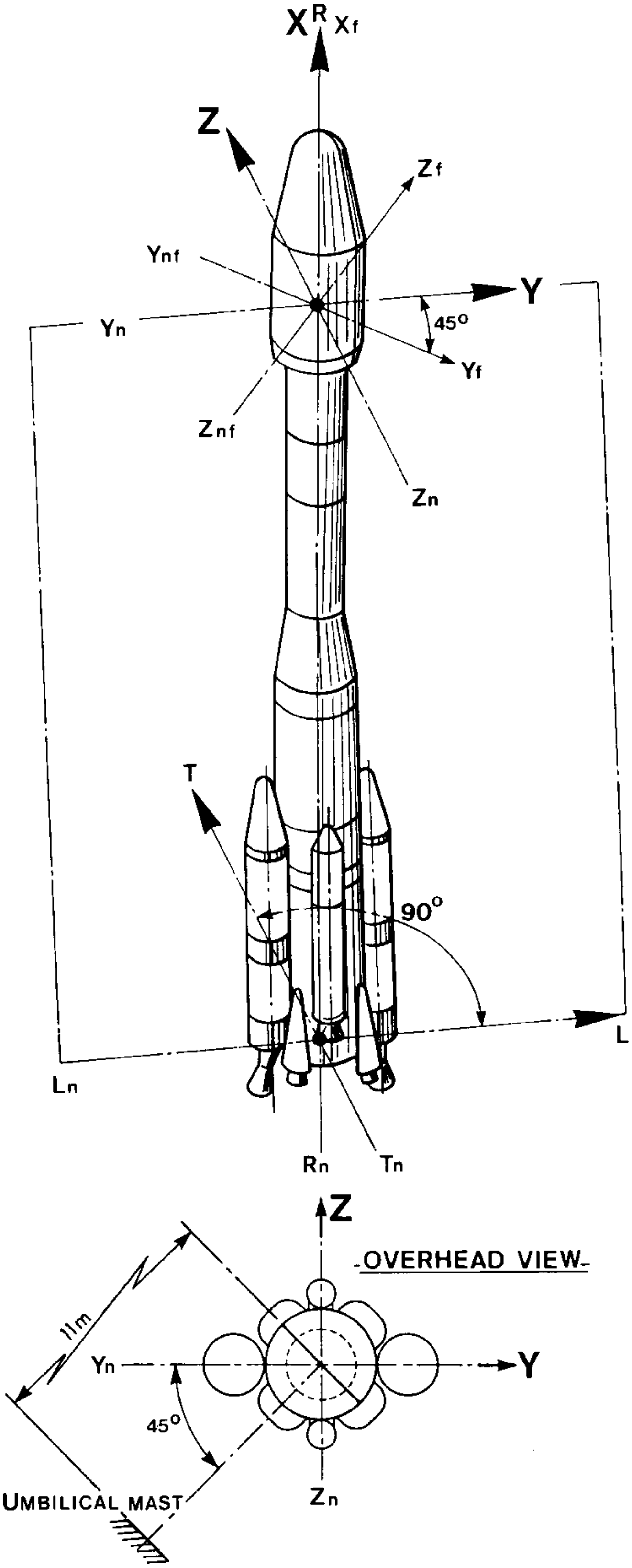


Fig. 4.1.a – ARIANE 4 Launch Vehicle Axes

4.3. Mechanical interfaces

4.3.1. Payload compartment configurations

For a single spacecraft launch, one of the available fairing designs protects the spacecraft mounted on top of an adaptor which can be a standard Ariane or User's design.

For dual launch two configurations are available:

- SPELDA configuration:
  - for the upper spacecraft position, one of the two compatible fairing designs (type 01 and 02) protects the spacecraft mounted on top of an adaptor (standard Ariane or User's design) fixed on to the SPELDA upper interface flange,
  - for the inner position, the SPELDA protects the spacecraft mounted on top of an adaptor (standard Ariane or User's design) fixed on the VEB interface frange.

- SYLDA configuration:
  - one of the two available fairing designs protects the dual launch system SYLDA.

Nose fairing, SPELDA and SYLDA are described below.

- Nose fairing description:
  The Ariane 4 nose fairing consists of a two half-shell carbon fibre structure with a longitudinal Ariane type separation system. This nose fairing has an external diameter of 4 m.

  Separation of the nose fairing is obtained by means of a pyrotechnic cord, located close to the plane joining the two half-shells.

  This cord shears the rivets connecting the two parts, and imparts a lateral impulse to the half-fairings, driving them apart by a piston effect. The gases generated by the system are retained permanently inside an envelope, thus avoiding any contamination of the payload by the separation system.

- SPELDA supporting structure description:
  The SPELDA structure consists of a carbon fibre cylindrical shell supporting the fairing and enclosing the inner spacecraft, and an upper troncated conical shell supporting the upper spacecraft.

  Separation of the SPELDA structure is achieved by means of a Linear Charge Cord Device (LCCD) which cuts the SPELDA structure along a horizontal plane, and springs impart a vertical impulse to jettison the SPELDA-top.

- SYLDA description:
  The SYLDA 4400 consists of a load-bearing carbon structure, comprising a conical adaptor carrying the inner spacecraft, and a shell, enclosing the inner spacecraft and carrying the upper spacecraft. The shell comprises two separable parts, held together in flight by a clampband.

  The attachment frames and separation systems for the upper and inner spacecraft are identical, and the interfaces at these points are the 937 ones (shoe angle 20°). The two spacecrafts have no mechanical or electrical interfaces with each other. External or internal spring configurations may be selected by the User for spacecraft separation.

Upper SYLDA

Lower SYLDA

Short SPELDA Mini SPELDA Stretched mini SPELDA

4.3.2. Spacecraft accessibility

Spacecraft accessibility is provided through the fairing and carrying structures (SPELDA and SYLDA). For details, refer to Annex 5.

4.3.3. Launch vehicle/spacecraft adaptors

4.3.3.1. Introduction

The spacecraft is mounted on the top of the launch vehicle via an adaptor.

The standard interface plane between the launch vehicle and the spacecraft is the bolted interface plane with a diameter of 1920 mm. For mounting a spacecraft on this 1920 interface, the User can either utilize one of the standard Ariane adaptors in which case the spacecraft separation system is provided with the Launch Vehicle, or utilize any other type of adaptor compatible with the 1920 or 2624 interface, in which case the separation system has to be provided by the User.

Another possibility is to utilize the VEB lower frame (2624 mm diameter interface), for which standard Ariane adaptors are also proposed.

In all cases the mounting plane for the spacecraft is orthogonal to the Ariane longitudinal axis within ± 2 mrd (milliradian).

Standard Ariane adaptors (including SYLDA) form part of the launch vehicle. Other adaptors may be designed and manufactured by the User to mate with the bolted interface planes (1920 or 2624).

4.3.3.2. Standard Ariane adaptors (see annexes)

- Adaptors 1194A, 1194B, 1194V.
- Adaptors 937, 397C, 937V4 (shoe angle 20°).
- Adaptors 937B, 937D, 937VB4, 937VD (shoe angle 15°).
- SYLDA dual launch adaptor.
- Adaptor 1663SP.
- Adaptor 1666A.

4.3.3.3. Other adaptors mounted on the 1920 diameter interface plane (see annex 6).

Other adaptors than those described in the User's Manual can be used on Ariane. In such a case, the specification requirements are to be taken out the following applicable document :

- A4-ST-0-P-19-01 "Design specifications for spacecraft adaptor using ∅ 1920 mm interface plane".

This document has to be requested by the user from ARIANESPACE.

Figures 4.3.3.3.a and 4.3.3.3.b show the volumes allowed to the spacecraft or adaptor on the vicinity of the 1920 mm bolted interface.

4.3.3.4. Adaptors mounted on the 2624 diameter interface plane (see annex 6)

Other adaptors than those described in the User's Manual can be used on Ariane. In such a case, the specification requirements are to tbe taken out the following applicable document :

- A4-ST-0-P-19-02 "Design specifications for spacecraft adaptor using ∅ 2624 mm interface plane".

This document has to be requested by the user from ARIANESPACE.
Figures 4.3.3.4.a and 4.3.3.4.b show the volumes allowed to the spacecraft or adaptor on the vicinity of the 2624 mm bolted interface.

4.3.4. Usable volume

Refer to annex 6.

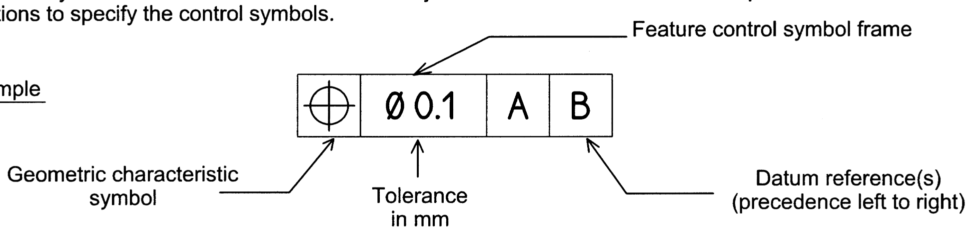
GEOMETRIC CHARACTERISTIC SYMBOLS

Symbols for shape and tolerances extracted from the R.E. AERO 772 75 specification approved by BNAe

CHARACTERISTIC		SYMBOL
Form tolerances	- Straightness	⏤
	- Flatness	⏥
	- Roundness	○
	- Cylindricity	⌭
	- Profile of a line	⌒
	- Profile of a surface	⌓
Location tolerances	- Angularity	∠
	- Perpendicularity	⊥
	- Parallelism	//
	- Postion	⌖
	- Concentricity	◎
	- Symmetry	⌯
	- Run out	↗

A control symbol label is attached to the item subjected to the tolerance. This label provides for the necessary positions to specify the control symbols.

Example



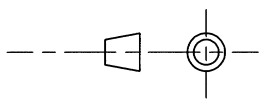
Basic dimension

12	Theoretically exact dimension without tolerance

Surface roughness symbol

1.6 maximum value (when not otherwise specified) expressed in micrometers. (1 µm= 0.001mm)

Projection



Inertias

Relatives to axes parallels to the launch vehicle passing through the center of gravity of the considered section.

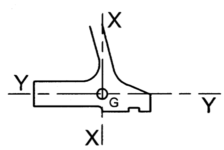
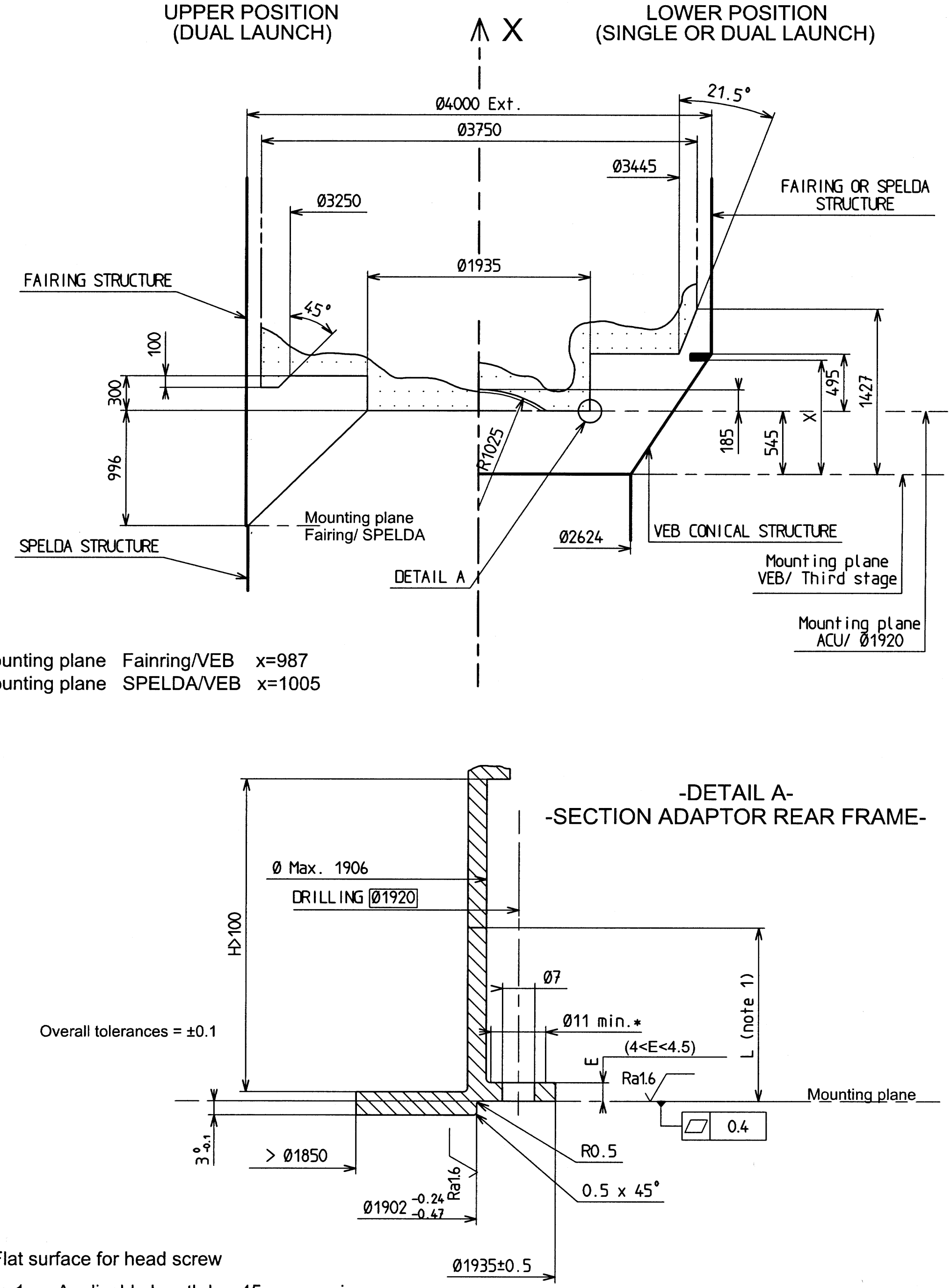


Fig. 4.3.3.a – Explanation of Symbols and Topics



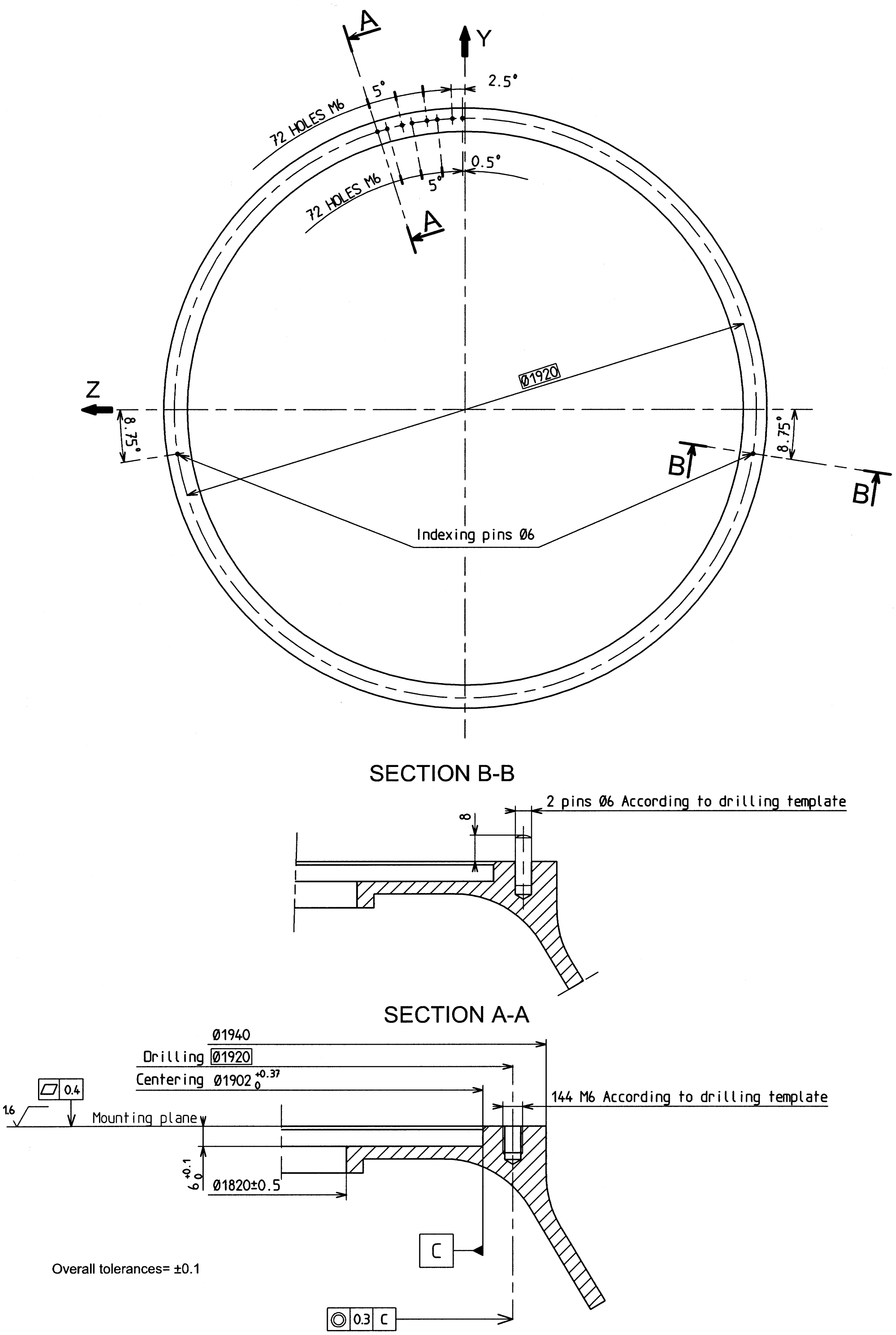
* Flat surface for head screw

Note 1 : Applicable length L = 45 mm maximum
Area : 330 mm² minimum
Inertia x : 31200 mm² minimum
Inertia y : 58000 mm² minimum

Note 2 : A drilling template provided by ARIANESPACE has to be used

Note 3 : Centering Ø 1902 is optional

Fig. 4.3.3.3.a - ∅ 1920 Reference Frame – Spacecraft Side



Overall tolerances= ±0.1

Fig. 4.3.3.3.b - Ø 1920 Reference Frame Launch Vehicle Side

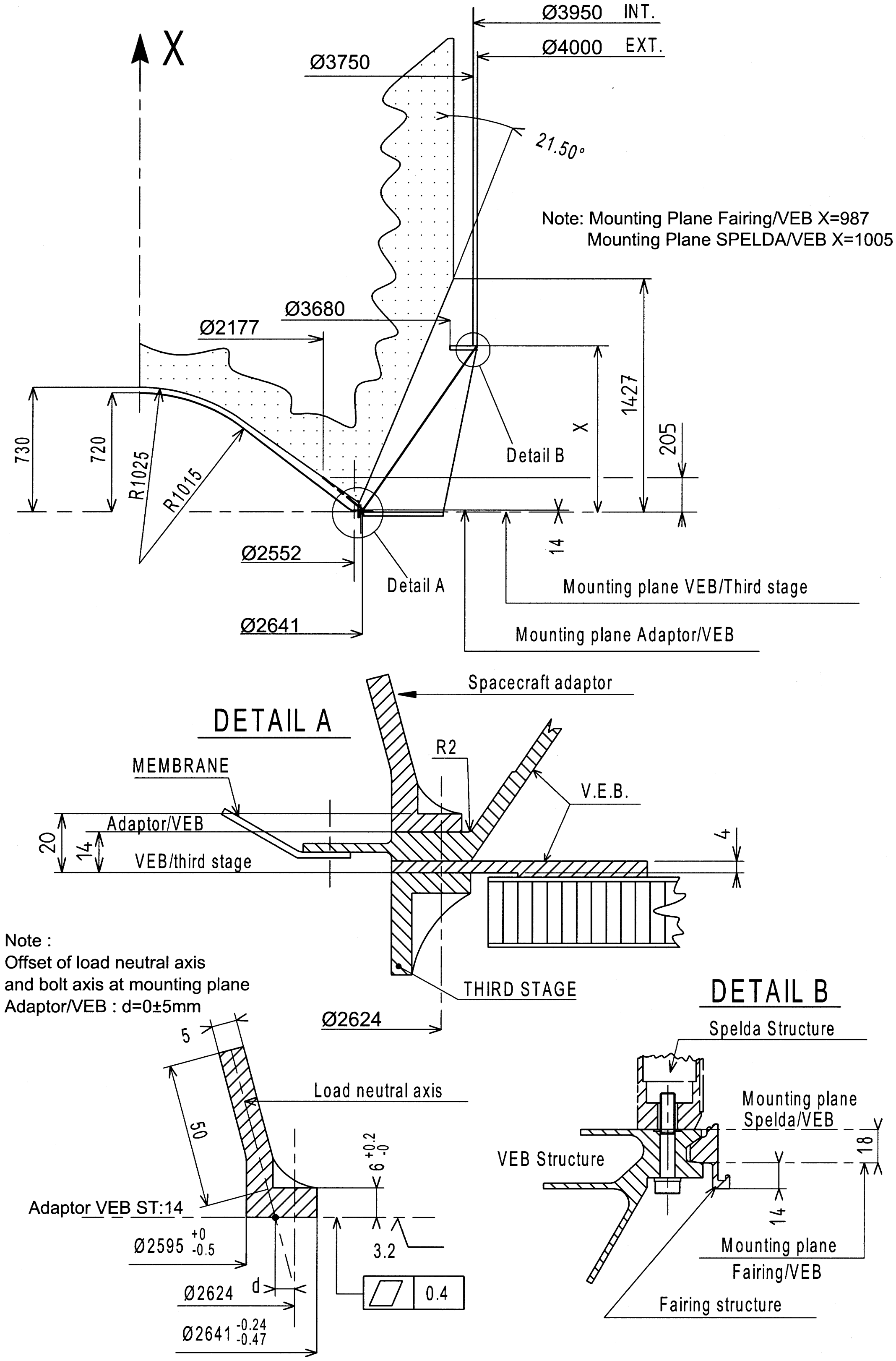


Fig. 4.3.3.4.a - ∅ 2624 Interface Frame

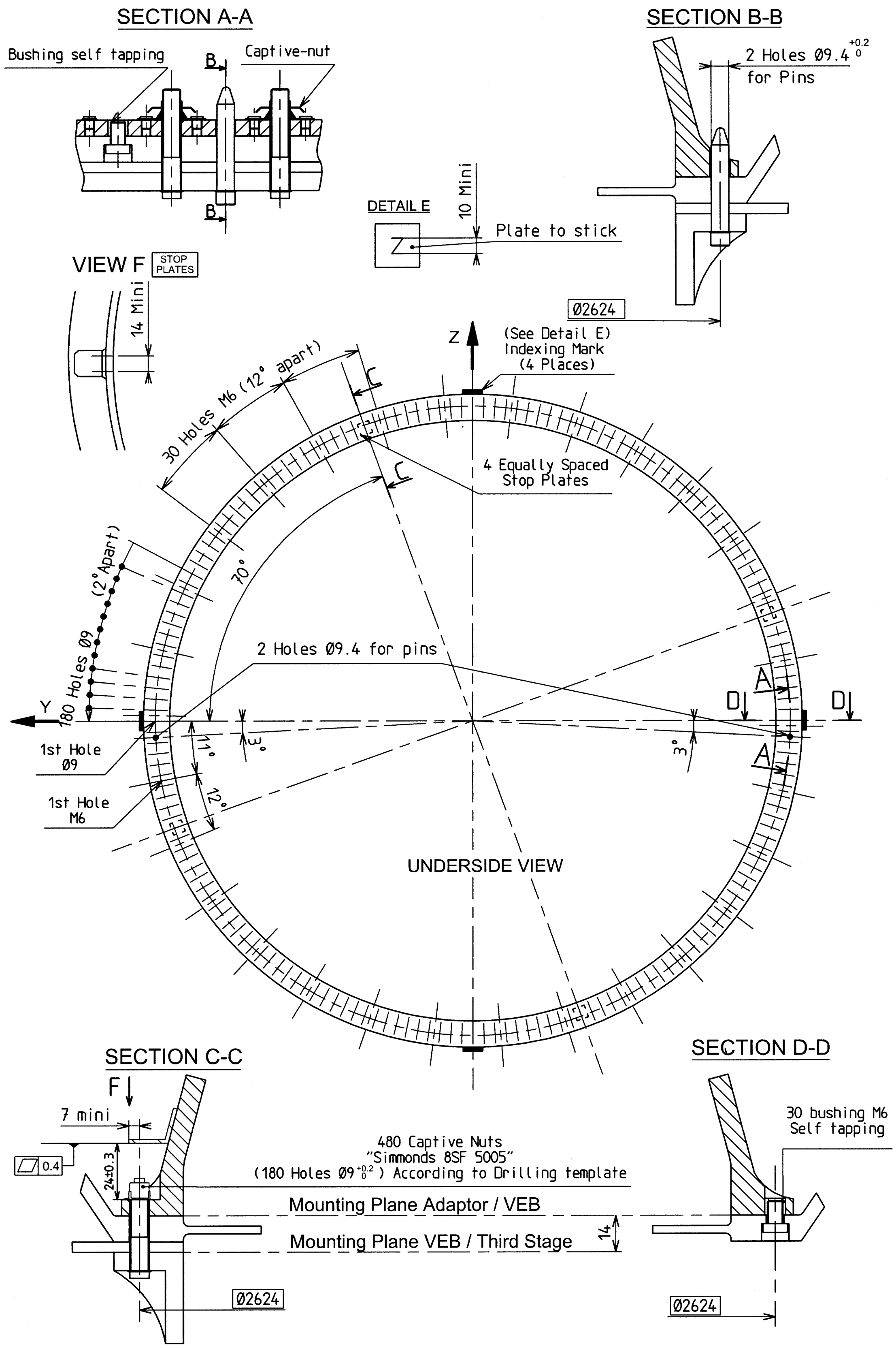


Fig. 4.3.3.4.b -∅ 2624 Interface Frame

4.4. Electrical and radio electrical interface

4.4.1. Earth-potential continuity

The spacecraft is required to have an « Earth » reference point close to the separation plane, on which a test socket can be mounted. 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.

Spacecraft structure in contact with the launch vehicle (separation plane of the spacecraft rear frame or mating surface of an user's adaptor) shall not have any treatment or protective process applied which creates a resistance greater than 10 mΩ for a current of 10 mA between the spacecraft earth reference point and that of the launch vehicle (Ariane adaptor or carrying structure).

4.4.2. Services available at the base of spacecraft adaptor

This section defines the electrical services, which the User may request at the base of the spacecraft adaptor i.e. the ∅ 1920 mm bolted interface.

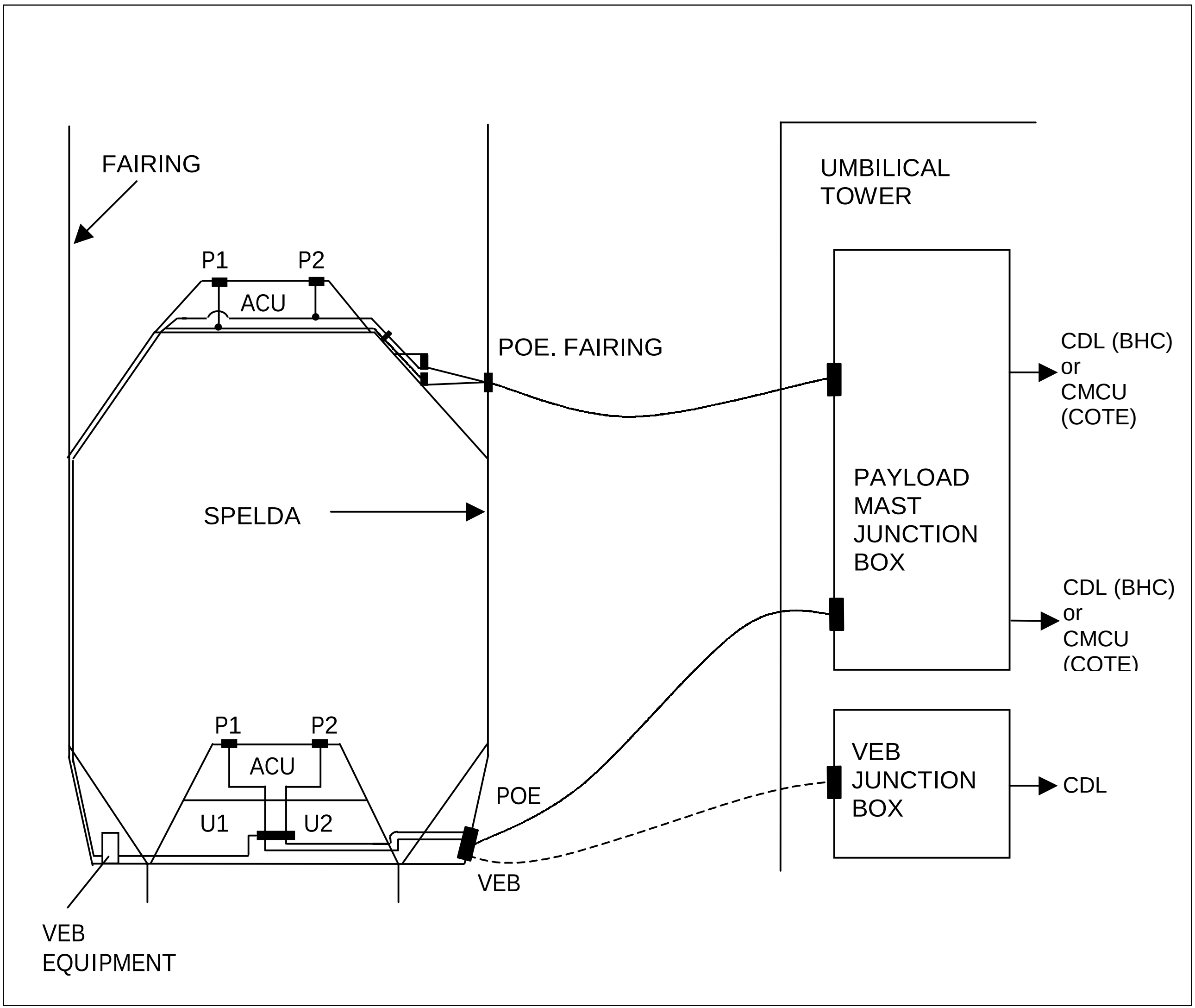
Some of these services are standard (Std), others are optional (Opt). The mode of access to the service at the interface and the related descriptive paragraph references are as follows:

Service	Interface access	Service definition	Ref. Para.
UMBILICAL LINK	2 x 37 pin connectors DBAS 74 37 OPW DBAS 74 37 OPY	74 wires (Std)	4.4.2.1
COMMANDS	2 x 37 pins connectors DBAS 74 37 OPN DBAS 74 37 OPN	Separation command (Std) Pyrotechnic command (Opt) Electrical command (Opt) Dry loop command (Opt)	4.4.2.2. 4.4.2.2. 4.4.2.3. 4.4.2.4.
TELEMETRY	1 x 19 pin connector DBAS 74 19 OPN	Separation status (Std) Power supply (Opt) In flight environment data (Std) Spacecraft data transmission (Opt)	4.4.2.5 4.4.2.7 4.4.2.6 4.4.2.6

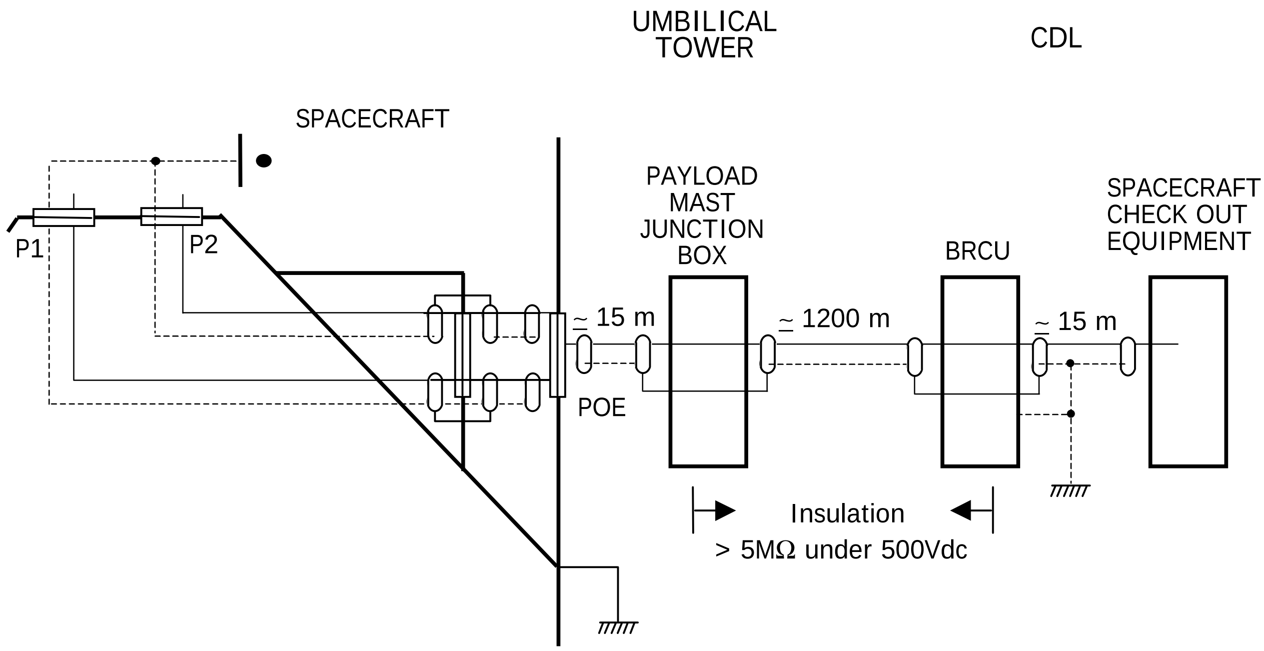
__Note :__ The above listed services are also available at the 2624 mm mounting plane. Details of interface access and other characteristics will be provided to the User, on request, within the related Adaptor Design Specification.

4.4.2.1. Umbilical Link

The principle of the umbilical links between a playload on top of Ariane and its electrical support equipment located in the ELA-2 Launch Centre is shown on fig. 4.4.2.1.a and 4.4.2.1.b.



4.4.2.1.a – Schematic of Spacecraft/Umbilical Mast Junction Box links



Ptbi	30	Screened insulated twisted pairs	0.93 mm^2
Ttbi	42	Screened insulated twisted triples	1.91 mm^2
Ttbi	30	Screened insulated twisted triples	6 mm^2
Cds	20	Insulated conductors	10 mm^2
Cds	22	Insulated conductors	16 mm^2
Coax	9	Coaxial links	KX4 – (Zo = 50 Ω)

Fig. 4.4.2.1.b – Available lines between the BRCU and the Payload Mast Junction Box to be shared between various Users

The following links are available as a standard service between each spacecraft and the umbilical mast junction box:

- 4 shielded triples,
- 30 shielded pairs,
- 2 common ground.

Characteristics of these umbilical lines are as follows:

- all wires 0.93 mm^2, resistance at + 20°C = 0.022 Ω /m
- insulation > 100 mΩ under 500 V dc.

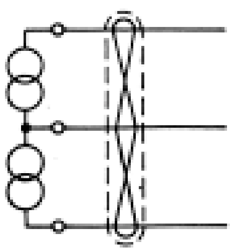
In addition, coaxial lines are available in the umbilical cable between the launch vehicle and the umbilical mast.

Operating constraints:

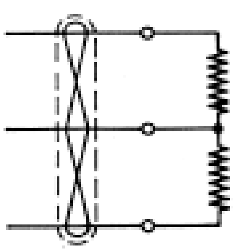
- The wired connections shall not carry current in excess of 7.5A (per wire) with a maximum voltage limitation of 150 V.

- Each pair will be used bidirectionally for one function (command, measurement, power supply, etc.):

- Each triple will be used bidirectionally (symmetrical or not) for one function (command, measurement, power supply, etc.):



or



- No current shall circulate in the screens. The User is responsible for grounding arrangements between the screens and the adaptor structure.

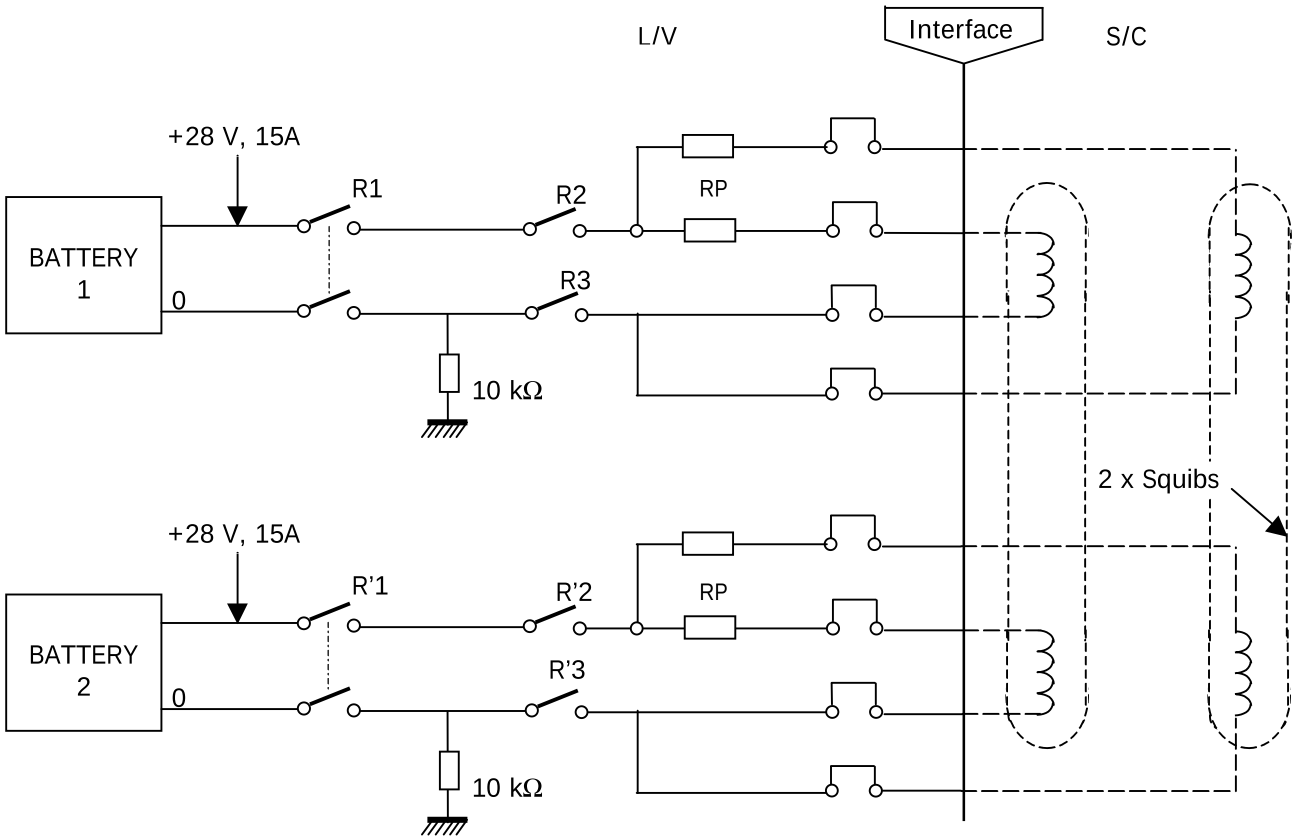
- Spacecraft wiring insulation: > 10 mΩ under 50 V dc.

4.4.2.2. Pyrotechnic command

- The pyrotechnic command for spacecraft separation is a standard service.

  User designing its own separation system should contact Arianespace to obtain details of the characteristics of this command.

- One additional command is available for a User's pyrotechnic system, as an optional service.
  This latter command can initiate 2 squibs and is fully redundant i.e. two totally separate lines provide the same command simultaneously, the power being supplied from separate batteries:

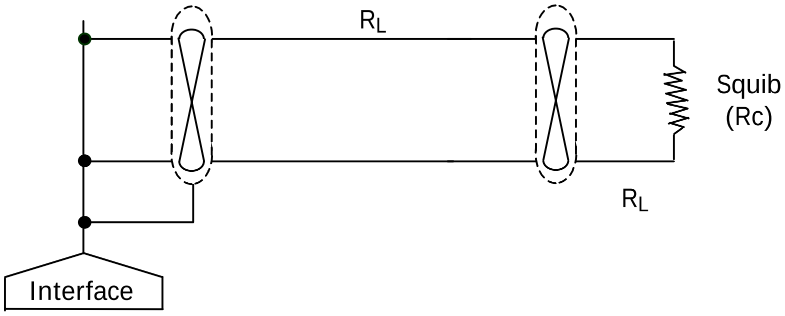


R1 & R'1 ARMING RELAYS ; R2, R'2, R3 & R'3 = COMMAND RELAYS; Rp = PROTECTION RESISTOR

Main electrical characteristics:

- Voltage (no-load): 28.8 V ± 4V.
- Pulse Width: 300 ms.
- Output insulation ≥ 100 MΩ.
- Current delivered to each of the 4 squibs: >3A for load resistor Rc = 0.95 Ω to 1.15 Ω and spacecraft line resistance R_L<0.8 Ω.
- Interval between 2 commands: the minimum time interval between the end of a transmitted command and the beginning of the following one is 20 ms.
- The execution of the pyrotechnic command (pyrotechnic voltage at sequencing unit output) is transmitted to the ground via telemetry.
- On option, 5A commands can be generated. For details users should contact Arianespace.

- Utilization constraints (S/C side)



The insulation between wires (open loop) and between wires and structure must be ≥ 10 MΩ (under 50 Vdc).

The user has to intercept the launcher command circuits (prime and redundant) in order:

- to protect the S/C equipment,
- to allow the integration check-out,

using a safety plug equipped with a shunt on S/Cside and a resistance of 10 Ω ± 1% (50 W) on the L/V side (See fig. 4.4.2.2.a).

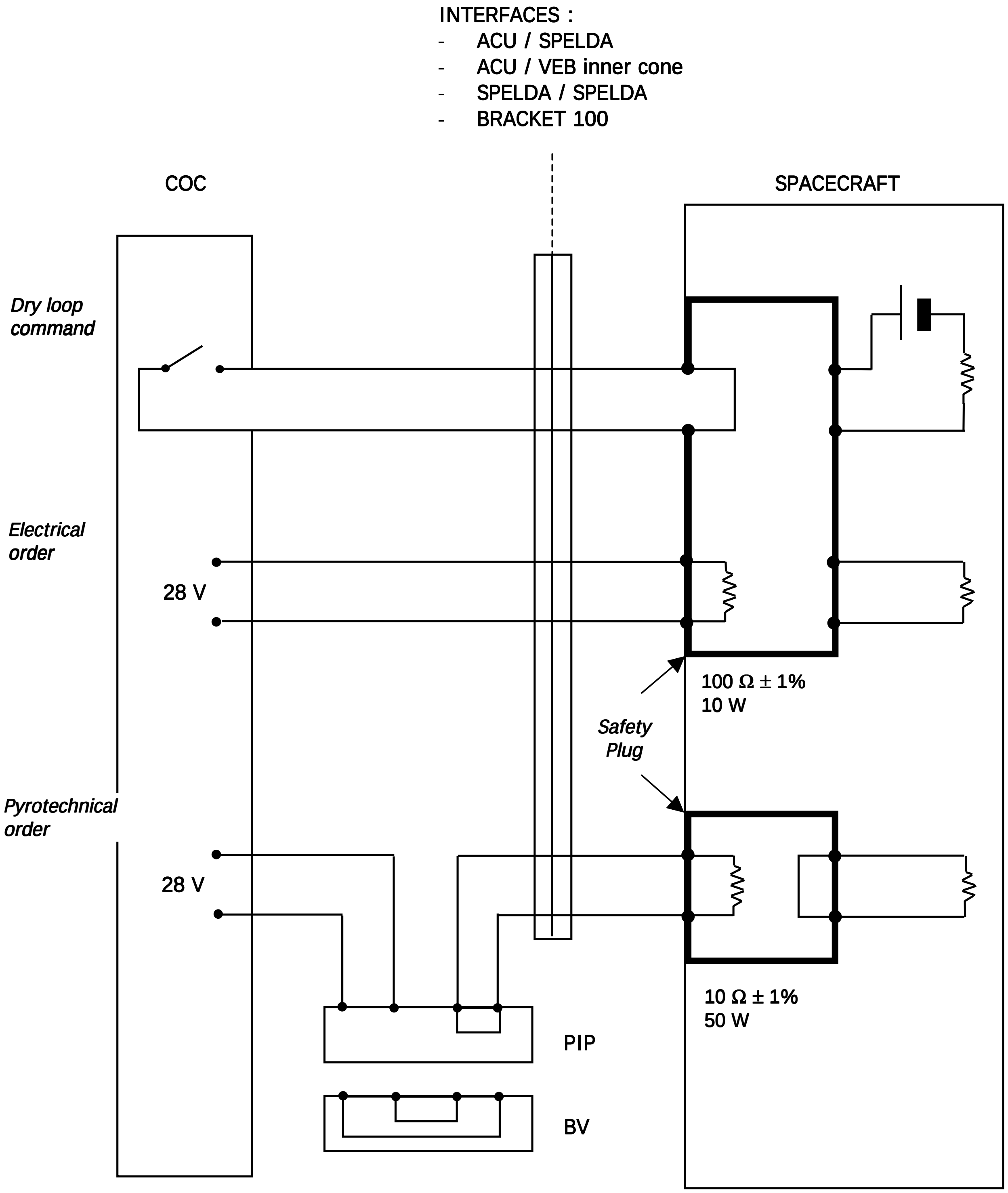


Fig. 4.4.2.2.a – Principle of S/C safety plug definition
(Nominal and Redundant Channels)

4.4.2.3 Electrical command

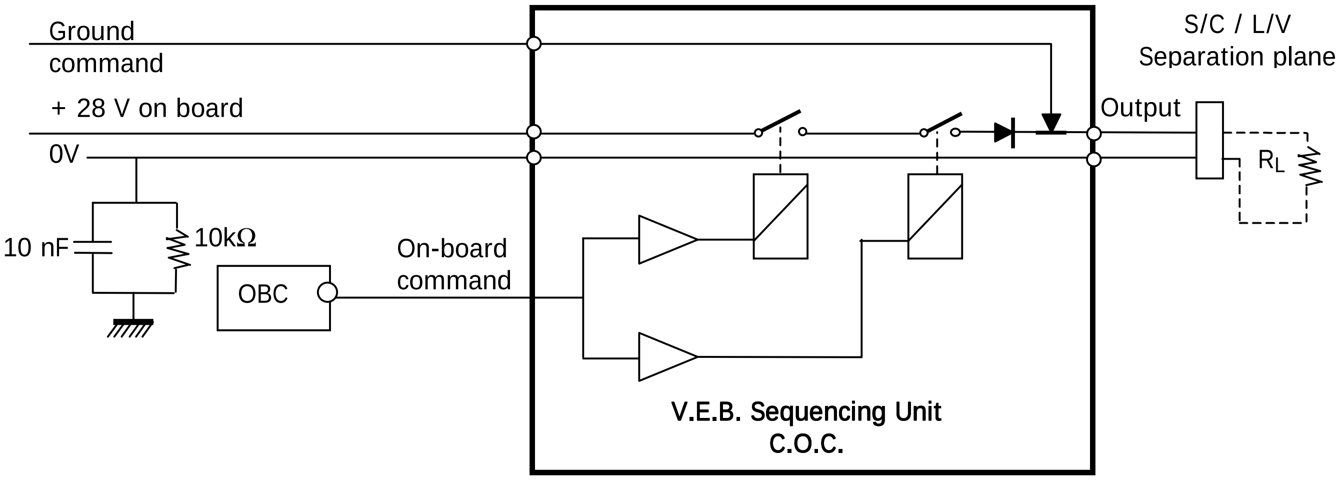
The User should contact ARIANESPACE to get detailed characteristics of this command.
One redundant electrical command is available to the User, as an optional service.

The command is initiated:

- on the ground: via the VEB umbilical plug (simulated electrical order),
- in-flight: by the flight programme.

The diagram shown below relates to one line of the redundant link.

- output voltage 28 V ± 4 V

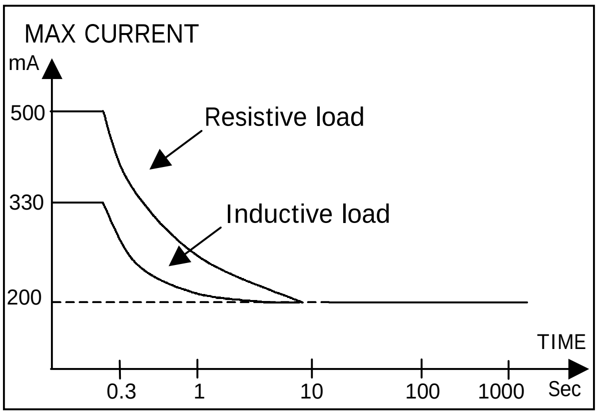


Main utilization constraints (S/C side)

- The user is required to use two independent loads, one on each redundant line. If a unique load is used, then a protection circuit is necessary up-stream of the summing-up points.

- Each circuit must be in compliance with the following constraints:

  Current/time characteristics:
  The user is required to dimension his load circuit so that the current drawn remains below the following curve.



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

  • Insulation: The insulation between the circuit load and the structure must be ≥ 10 MΩ under 50 Vdc.

  • Electromagnetic compatibilty: These links must be in compliance with the specification MIL.SDT.462 Method CE01.

- The user has to intercept the launcher command circuits (prime and redundant) in order:

  • to protect the S/C equipment,
  • to allow the integration check-out,

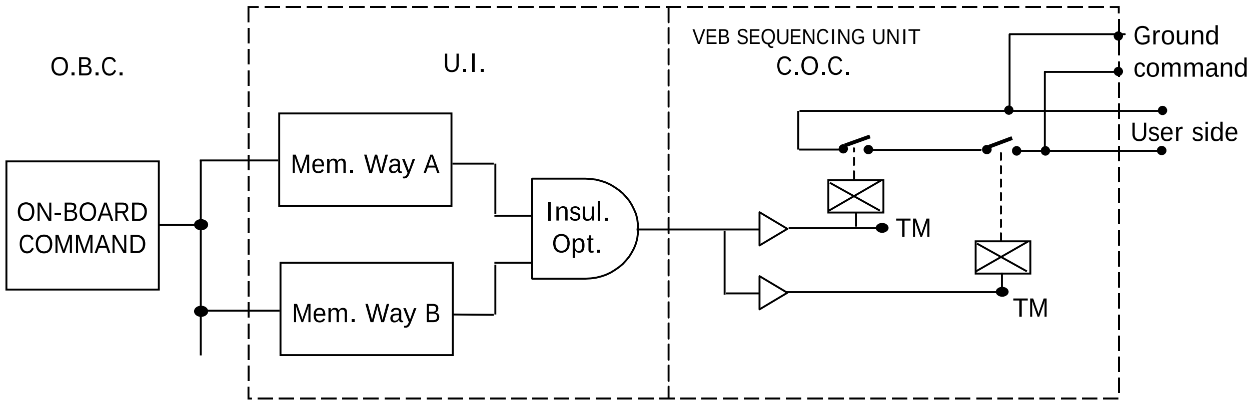
  using a safety plug equipped by the user on the S/C side and a resistance of 100 Ω ± 5% (10 W) on the L/V side (See fig. 4.4.2.2.a).

4.4.2.4. Dry loop command

The User should contact ARIANESPACE to get detailed characteristics of this command. One redundant dry-loop command is available to the User, as an optional service. The command is initiated:

- on the ground: via the VEB umbilical plug,
- in flight: via the flight programme.

The diagram shown next page relates to one line of the redundant link.



- Main electrical characteristics:
  R off ≥ 10 MΩ
  R on ground ≤ 10 Ω
  Ron board ≤ 1Ω
  On board circuit insulation: ≥ 10 MΩ under 50 Vdc
- Utilization constraints (S/C side)
  U Max ≤ 45 V
  I Max ≤ 0.5 A

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

The user has to intercept the launcher command units (prime and redundant) in order:

- to protect the S/C equipment,
- to allow the integration check-out,

using a safety plug equipped with an open circuit on the S/C side an a short circuit on the L/V side (See fig.4.4.2.2.a).

4.4.2.5. Separation Status Transmission

The VEB telemetry system sends status signals concerning satellite separation. Signals are initiated at the interface either by strap or by micro-switches.

4.4.2.6. In Flight Telemetry

Transmission of data on the in-flight dynamic environment at the interface plane with the User is performed by the VEB Telemetry system. Transducers are provided with standard Ariane adaptors.

In the case of a User's supplied adaptor, Arianespace will supply the User with transducers to be installed on the adaptor close to the interface plane.

In the case of a single spacecraft launch, the provision of facilities for the in-flight transmission of spacecraft measurements can be studied, as an optional service, on a case by case basis. A User desiring to excercise such an option should contact Arianespace for interface characteristics.

4.4.2.7. Power supply

A VEB power supply (non redundant) is available to the User, as an optional service. The characteristics are:

- voltage: 28 V ± 4 V
- current: < 0.7 A for single launch
  < 0.35 A for dual launch

Main Utilization constraints (S/C side)

The User should contact ARIANESPACE to get detailed characteristics of this optional service.

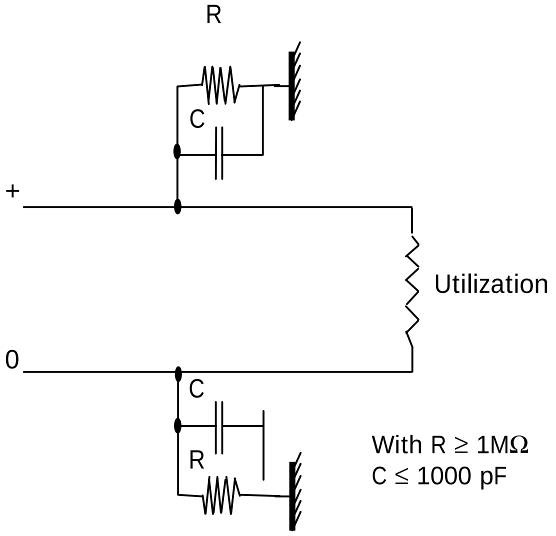
- Protection: The User is required to protect the circuit against any overload or voltage-overshoot induced by his circuits, both at circuit switchings and in the case of circuit degradation.

- In case of VEB degraded mode voltage limits can vary as much as:

    - Lower voltage: U = 24 V
    - Upper voltage: U = 32 V
    - Maximum over voltage: U = 35 V during 5 minutes

In this case the selected protection device must remain fully operational if, during 1 minute maximum, the S/C system consumption exceeds 10 % of the authorized maximum.

- Insulation: at interface level, the circuit insulation measurement should be as follows.



Note: Battery capacity mentioned above must be shared in case option described in para. 4.4.2.6 is also excercised.

- Electromagnetic compatibility: these links must be in compliance with the specification MIL. SDT. 462 Method CE01.

4.4.3. Electrical interface on standard Ariane adaptors

Standard Ariane adaptors are equipped with harness providing connections between the services available at the adaptor 1920 mm or 2624 mm mounting plane, as described in the para. (4.4.2) and the spacecraft interface connectors.

Table 4.4.3.a describes these interface connectors for each of the standard Ariane adaptors. All services available at the adaptor mounting plane (ref. para. 4.4.2.) can be routed to these interface connectors, the only limitation being the number of pins available in the interface connectors.
The mechanical interface description of these connectors is provided in the related applicable sections of Ariane standard adaptors annexes.

On the spacecraft side, the umbilical connectors brackets must be stiff enough to prevent any deformation greater than 0.5 mm under the maximum force of the connector spring.

Note: Arianespace will supply the User with the spacecraft side interface connectors with which standard adaptors have been qualified.

	Available Interface connectors	Adaptor side reference (male)	Spacecraft side reference (female)	Number of pins available to the User (3)
Adaptors 1194	2 connectors: 27 pins each or 37 pins each or 61 pins each	DBAS.025.82.1027 DBAS.025.82.1127 DBAS.025.82.1037 DBAS.025.82.1437 DBAS.025.82.1061 DBAS.025.82.1461	DBAS.7027.0SN DBAS.7027.0SW DBAS.7037.0SN DBAS.7037.0SY DBAS.7061.0SN DBAS.7061.0SY	54 see note (1) 74 see note (1) 74 see note (1)
Adaptors 937 or SYLDA	2 connectors : 27 pins each or 37 pins each	DBAS.025.82.1027 DBAS.025.82.1127 DBAS.025.82.1037 DBAS.025.82.1437	DBAS.7027.0SN DBAS.7027.0SW DBAS.7037.0SN DBAS.7037.0SY	54 see note (1) 74 see note (1)
Adaptor 1666A or Adaptor 1663SP	2 connectors: 27 pins each or 37 pins each or 61 pins each	DBAS.025.82.1027 DBAS.025.82.1127 DBAS.025.82.1037 DBAS.025.82.1437 DBAS.025.82.1061 DBAS.025.82.1461	DBAS.7027.0SN DBAS.7027.0SW DBAS.7037.0SN DBAS.7037.0SY DBAS.7061.0SN DBAS.7061.0SY	54 see note (1) 74 see note (1) 74 see note (1)

(1) Separation status is given by 2 microswitches routed separately.
(2) Two pins of one 37 pin connector are required to be used for separation status.
(3) Among these pins, two (one per connector) are to be reserved for shielding.

Note : All contacts are in 0.93 mm^2.

Table 4.4.3.a – Interface connectors of standard Ariane adaptors

4.4.4. Radio and electromagnetic constraints

4.4.4.1. Fairing, SPELDA and SYLDA transparency for spacecraft radio-communications.

- Radiotransparency of fairing and SPELDA can be achieved through radio transparent windows for fairing and through access doors for SPELDA enabling radio links to be maintained between spacecraft and ground support equipment. The insertion loss of the material used for a window will be less than 2 dB in the 2 GHz – 15 GHz frequency range.

  The locations of ∅ 250 fairing windows are given in figure 4.4.4.1.a. The locations and dimensions of SPELDA access doors are given in annex 5.

- Radio transparency through SYLDA is ensured by holes in the structure.

- A passive repeater system can also be installed inside the SPELDA or the fairing. It is composed of 2 cavity back spiral antennae and it is linked to the launch tower until lift-off.

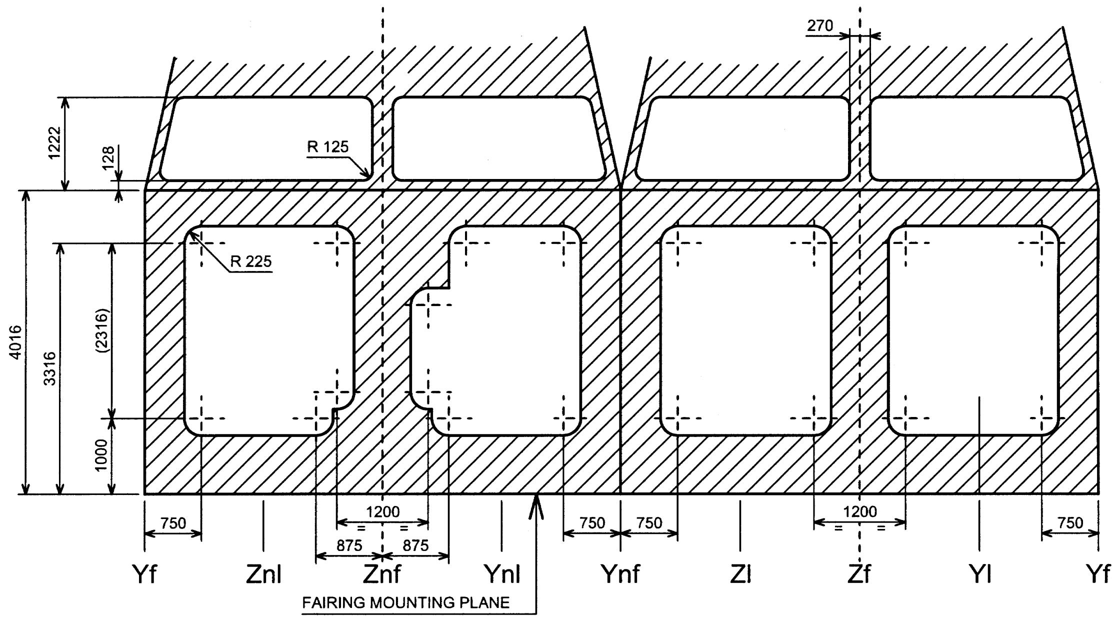
4.4.4.2. Operating constraints

- The spacecraft shall not radiate a narrow-band electrical filed at the level of the VEB exceeding the limit set in figure 4.4.4.2.a (including intentional transmission);

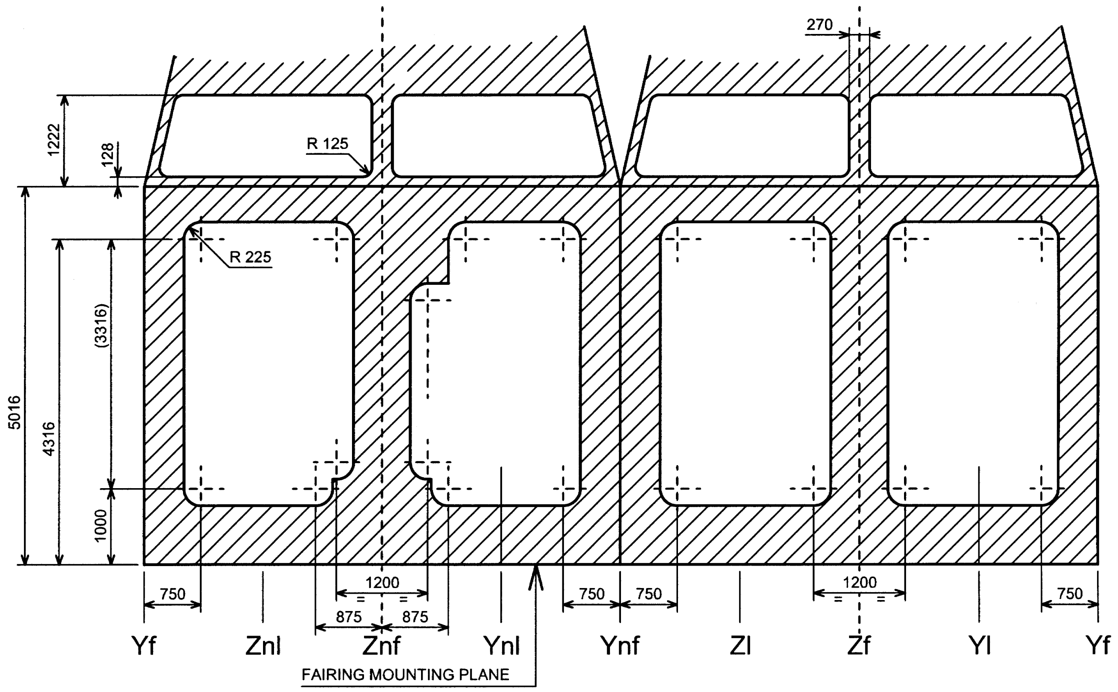
- A 20 dBµv/m level radiated by the spacecraft, in the launch vehicle telecommand receiver 400-500 MHz band, shall be considered as the worst case of the sum of spurious level over a 300 kHz bandwidth.

- Spacecraft transmitters have to meet general IRIG specifications.

FAIRING TYPE 01



FAIRING TYPE 02



Internal developped view R2000

Fig. 4.4.4.1.a – Locations of fairing radio-transparent window

4.4.5. Electrical and radio requirements for the launch phase

- Electrical requirements:

The User shall design his spacecraft so that during the final preparation phase leading up to actual launch, the umbilical cables are carrying only very low currents at the moment of lift-off, (i.e. less than 10 mA – 50 V for a resistive circuit). Spacecraft power shall be switched from external to internal and ground power supply must be switched off before lift-off.

- Radio requirements:

135 – 139 MHz band: Spacecraft transmissions are not permitted during flight until separation time on orbit plus 20 s.

In other bands there is a restriction on spacecraft transmissions up to 20 s after separation of spacecraft. Authorization to allow transmission during the countdown phase and/or flight phase and/or at spacecraft separation will be considered on a case by case basis.

The housekeeping telemetry and telecommand of the spacecraft may be subject to change on request of ARIANESPACE up to 20 months before launch.

The spacecraft telemetry frequency band must not overlap the launch vehicle bands: 2203 MHz ± 250 kHz, 2218 MHz ± 250 kHz and 2227 MHz ± 250 kHz.

Flight constraints: refer to para. 5.3.2.2.

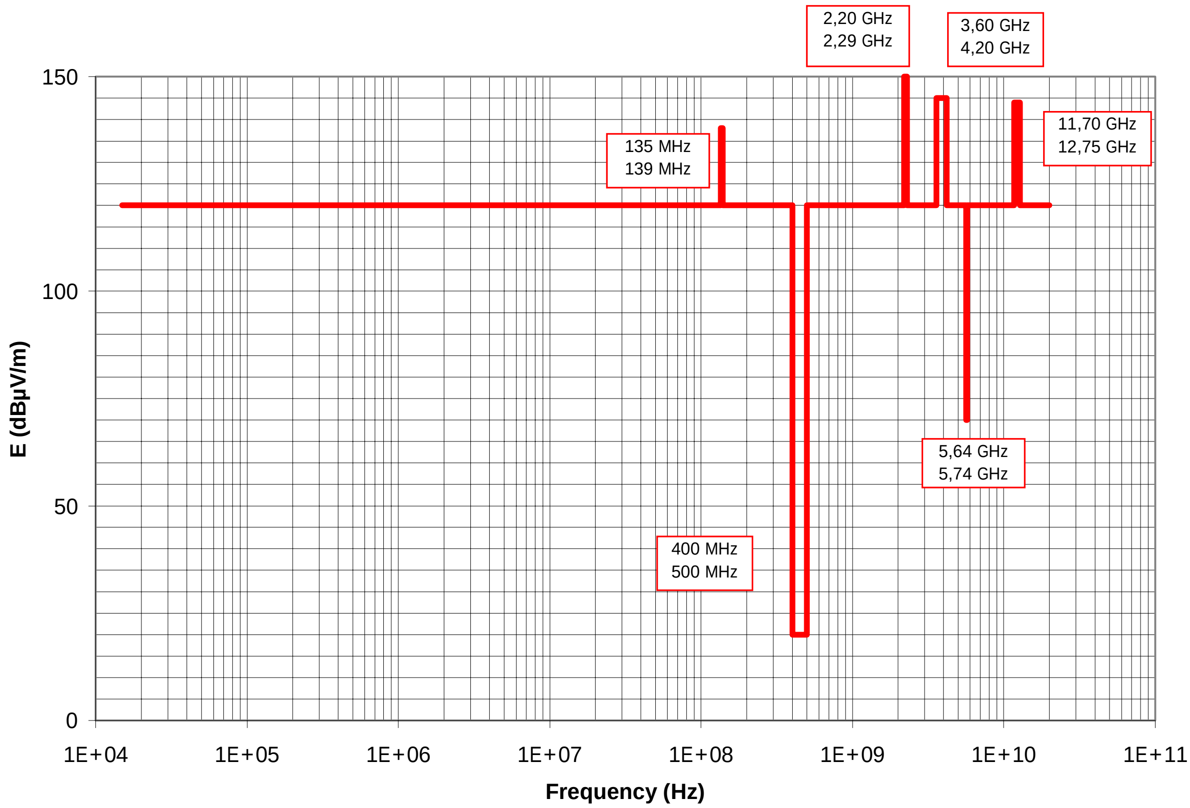


Fig. 4.4.4.2.a - Spurious radiations acceptable to launch vehicle
Narrow-band electrical field

4.5. Dimensioning

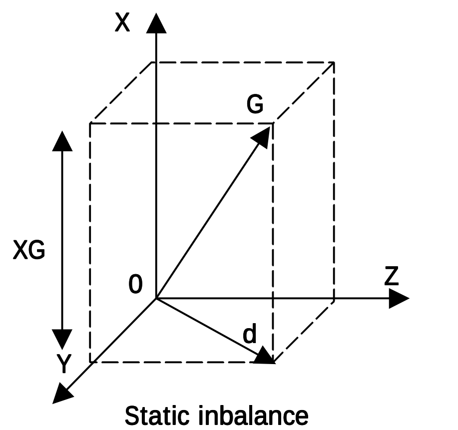
This chapter defines the spacecraft design data and dimensioning criteria that shall be taken into account by any User intending to be compatible with Ariane 4 launch vehicles.

In the following paragraphes the spacecraft station is defined as the top of the adaptor used.

4.5.1. Center of gravity limits

If Mcu < 2 000 kg and f_{longit} > 40 Hz, XG is to be < 2,2 m (w.r.t. the ∅ 1920 mm bolted interface).

For other S/C, see annexes refering to the adaptors.



Static inbalance

4.5.2. Spacecraft balancing*

4.5.2.1. Static inbalance

In the same three-axis system O, X, Y, Z, the centre of gravity of the spacecraft must rest within a distance d ≤ 10 mm from the axis OX.

4.5.2.2. Dynamic inbalance

For spacecraft which require to be spun up, the maximum dynamic inbalance ε corresponding to the angle between the spacecraft longitudinal geometrical OX-axis and the principal roll inertia axis shall be:

$\varepsilon \leq 1$ degree.

**** It should however be kept in mind that this parameter is in direct relationship with the angular transverse rate after separation.***

4.5.3. Frequency requirements

To avoid dynamic coupling between the low-frequency vehicle and spacecraft modes, the spacecraft should be designed with a structural stiffness which ensures that:

- the fundamental frequency in the lateral axis is above 10 Hz,
- the fundamental frequency in the thrust axis is above 31 Hz.

This applies for a spacecraft hardmounted at separation plane.

If the User supplies its own adaptor and separation system, the stiffness requirements defined in the design specifications for spacecraft adaptors (see para. 4.3.3.3. and 4.3.3.4.) must be fullfilled.

For a single launch, lower figures can be accepted. This should be discussed with Arianespace so that appropriate analyses can be performed, in order to define the loading conditions more precisely.

4.5.4. Dimensioning loads

During flight, low frequency dynamic and steady loads are combined to produce the quasistatic loads (QSL).

The design and dimensioning of the spacecraft primary structure must therefore allow for the most severe load combination that can be encountered at any given instant of flight. In this respect the most critical flight events are:

- during maximum dynamic pressure,
- just before thrust termination,
- during thrust tail-off.

The corresponding flight limit loads are given in the table 4.5.4.a.

These data are intended as a guide to be used for initial spacecraft design.

	Acceleration (g) Q.S.L.	
Flight event	*Longitudinal*	*Lateral axis*
Maximum dynamic pressure	*- 3.0*	*± 1.5*
Before thrust termination	*- 5.5*	*± 1.0*
During thrust tail-off	*+ 2.5 **	*± 1.0*

The minus sign with longitudinal axis values indicates compression.

Lateral loads may act in any direction simultaneously with longitudinal loads.

These loads apply uniformely all over the primary structure of a spacecraft complying with:
- the frequency requirements of para. 4.5.3,
- the static moments of para. 4.5.1.

**** For spacecraft with first longitudinal frequency above 40 Hz, the "during thrust tail-off" tension value is following:***

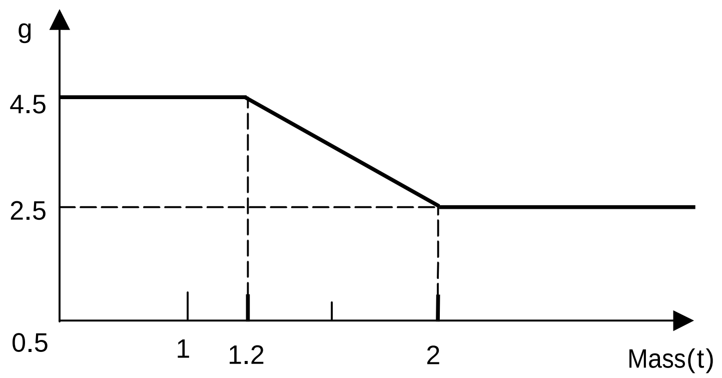


Table 4.5.4.a – Flight limits loads

Dimensioning must take into account safety factors, which are defined by the Spacecraft Authority (Arianespace requires a minimum value of 1.25 at ultimate stress and 1.1 at yield stress).

The secondary structures and flexible elements (e.g. solar panels, antennae and propellant tanks) must be designed to withstand the dynamic environment induced at the base of the spacecraft (as described in pare 3.1.) taking account of amplification resulting from the design of the spacecraft.

Coupled loads analysis:

The dynamic coupled load analysis (preliminary and final) carried out as part of the Mission Analysis studies (see Chapter 6.4) must be used as verification tools for the dimensioning of the spacecraft.

A simplified coupled load analysis based on a parametric spring-mass model of the spacecraft can be requested at short notice (about one month). Contact Arianespace for detailed specification.

4.5.5. Overflux

The maximum value of the circumferential flux, at the spacecraft / adaptor interface, induced by the spacecraft and taking into account its geometrical discontinuities (e.g. holes, stringers, ...) is allowed in local areas to be up to 10 percent over the dimensioning flux seen by the adaptor under limit loads condition.

4.6. Spacecraft compatibility tests

4.6.1. Spacecraft structural qualification and acceptance tests

The Spacecraft Authority has to prove that the spacecraft structure is capable of withstanding the Ariane flight environment based on a coupled load analysis provided by Arianespace as well as the random, acoustic and shock environment described in chapter 3.

The spacecraft mathematical model to be provided by the User to Arianespace for carrying out the launch vehicle/spacecraft coupled load analysis has to be validated by test.

The spacecraft structural capability must be proven by adequate tests. The following classification corresponds to current practice:

1) Structural Test Model approach

- static, sine and acoustic (or random) qualification tests on structural test mode,
- sine and acoustic (or random) protoflight tests (qualification level, acceptance duration) on first flight model,
- acoustic (or random) acceptance tests on subsequent flight models.

2) Protoflight Model approach

- static qualification by heritage,
- sine and acoustic (or random) protoflight tests on first flight model,
- sine and/or acoustic (or random) acceptance tests on subsequent flight models.

3) Recurrent spacecraft acceptance tests

- acoustic (or random)

or

- sine and acoustic (or random).

The mechanical environmental test plan for spacecraft qualification and acceptance shall be reviewed by Arianespace prior to implementation of the first test. The selected test must comply with the requirements presented from para. 4.6.1.1. to 4.6.1.6.

4.6.1.1. Static qualification tests

On the basis of the Ariane quasi-static loads as described in para. 4.5.4. the User determines the dimensioning load cases to which the spacecraft structure will be subjected. The static tests shall be carried out without rupture up to flight limit loads, multiplied by a factor > 1.25.

4.6.1.2. Sinusoïdal vibration tests

Sine	Frequency range (Hz)	Qualification levels (0-peak) (recommended)	Acceptance levels (0-peak)
Longitudinal	5-6 6-100	8.6 mm 1.25 g	1 g 1 g
Lateral	5-18 18-100	1 g 0.8 g	0.8 g 0.6 g
Sweep rate		2 oct./min	4 oct./min

A notching procedure will be agreed on the basis of the latest dynamic coupled load analysis available. Launch vehicle forcing function discrepancies are taken into account in the CLA results. In addition, a 1.25 safety factor will be applied in order to determine the maximum allowable notching levels for protoflight or qualification tests.

4.6.1.3. Transient Test

For detailed test definition, the User is requested to contact Arianespace.

4.6.1.4. Random vibration tests

Random	Frequency range (Hz)	Density (g^2/Hz)	rms value (g)
Qualification levels (recommended)	30 – 150 150 – 700 700 – 2000	+ 6 dB/Oct 0.09 –3 db/Oct	11
Acceptance levels	30 – 150 150 – 700 700 – 2000	+ 6 dB/Oct 0.04 –3 db/Oct	7.3

The levels are identical for longitudinal and lateral vibrations. The test duration shall be 2 minutes per axis for qualification, and 1 minute per axis for acceptance testing.

4.6.1.5. Acoustic vibration test

Octave band centre frequency (Hz)	Qualification Level (recommended)	Acceptance level (flight)	Test tolerance
	0 dB : (ref. 2 x 10^{-5} pascal)		
31.5	124	120	-2, +4
63	131	127	-1, +3
125	139	135	-1, +3
250	143	139	-1, +3
500	138	134	-1, +3
1000	132	128	-1, +3
2000	128	124	-1, +3
4000	124	120	-4, +4
8000	120	116	-4, +4
Overall level	146	142	-1, +3
Test duration	2 minutes	1 minute	

The tolerance indicated in the above table allows for standard test-equipment inaccuracy.

Fill factor correction : special consideration shall be given to spacecrafts which fill factor, calculated as the ratio of the horizontal cross area of the spacecraft main body (without appendages) over the fairing/SPELDA section (∅ 3900) is greater than 80 % : + 5 dB at 31 Hz, + 5 dB at 63 Hz, + 4 dB at 125 Hz and + 1 dB at 250 Hz.

The test levels are to be measured, in the acoustic chamber, at a minimum distance of one meter from the spacecraft. User's wishing to apply a one-third octave band spectrum are requested to submit it to ARIANESPACE for comments.

4.6.1.6. Shock test

The spacecraft and, in particular, the equipments located close to the separation interface, must withstand the shock specified in para. 3.1.7. Spacecraft dimensioning must take into account this specification to define equipment base environment. The demonstration of the spacecraft ability to withstand this shock can be made :

- either through a test and analytic demonstration performed in two steps :
  - a shock test (generating a shock at the interface, typically by means of a separation system release test), during which interface levels and equipments base levels are measured.
    This test can be performed on the STM, PFM or on the first flight model, provided that the spacecraft structure close to the interface and the equipments locations and mountings are the same as on the flight model. 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.
  - an analytic demonstration of the qualification of the equipment. This is obtained by comparing the component unit qualification levels to the equipment base levels experienced during clampband firing test (with the addition of a qualification margin, 3 dB recommended). This demonstration could be made by using equivalent rules on other environment qualification test (i.e. random or sine).

- or based on multi clampband firing heritage

4.6.2. Interface tests

At User's request, the following tests can be arranged, based on the provision of specific Ariane hardware.

4.6.2.1. Mechanical compatibility tests

Flight configuration test hardware can be provided for 937, 1194, 1663 and 1666 mm diameter interfaces, with associated separation systems and umbilical connectors, without cabling.

4.6.2.2. Fit-check

Ariane flight adaptors can be provided for 937, 1194, 1663 and 1666 mm diameter interfaces, with associated separation systems and umbilical connectors, without cabling. For a SYLDA inner spacecraft, the inner flight SYLDA adaptor (SBI) can be made available for a fit check at the spacecraft integration plant. A fit check for a SYLDA upper spacecraft with the upper SYLDA flight structure (SH) is only possible at the CSG, at the beginning of the launch campaign.

4.6.2.3. Volume compatibility test

A SYLDA 4400 volume simulator (SVS) can be made available in order to verify that the spacecraft remains within the specified static envelope. This equipment may be needed to verify the accessibility of the spacecraft through SYLDA.

4.6.2.4. Separation shock test

Flight configuration test hardware can be provided for 937, 1194, 1663 and 1666 mm diameter interfaces with the associated separation systems and consumable items.

It is recommended that this test be performed in conjunction with the mechanical compatibility check.

4.6.2.5. Acoustic test

Flight configuration test hardware can be provided for 937, 1194, 1663 and 1666 mm diameter interface for acoustic test, up to flight levels.