GALILEO GALILEI (GG)

LAUNCHER IDENTIFICATION AND COMPATIBILITY ANALYSIS REPORT

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1. INTRODUCTION

1.1 Scope

This document is submitted in partial fulfilment of Work Package 1A-ADA of the GG Phase A2 Study (DRL item DEL-44). It summarizes the identification and compatibility analysis of launcher for Galileo Galilei (GG) mission.

1.2 Background

The Galileo Galilei (GG) mission is a part of the Cosmology and Fundamental Physics project of the ASI Unit on Observation of the Universe, the purpose of which is providing support to the Italian Scientific Community in its participation in the European and worldwide development of knowledge in this field, both by independent projects and by international collaboration. Mission goal is to test the “Equivalence Principle” (EP) to 1 part in 10^{17}, more than 4 orders of magnitude better than today’s ground experiments.

The GG mission and satellite have already been studied at both scientific and industrial level. Between 1997 and 2000, a mission based on an equatorial orbit was studied under ASI contract [RD 1]. In 2001, adaptation of the mission to a sun-synchronous orbit, driven by launcher availability, was addressed [RD 2]. The successful launch of AGILE has demonstrated the feasibility for ASI of launching, at low cost, a small satellite into near perfectly equatorial orbit. Thus the equatorial orbit, which was preferred anyway because of simplicity of design and operation, can be taken again as the GG baseline.

The design orbit altitude is around 600 km, the actual value depending on the launch epoch [RD 11]. The spacecraft mass is about 400 kg including 20% margin. Launcher selection shall be done taking in account these data as references.

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2. ACRONYMS

AD	Applicable Document
ASI	Agenzia Spaziale Italiana
AVUM	Attitude and Vernier Upper Module
EP	Equivalence Principle
GG	Galileo Galilei
GTO	Geosynchronous Transfer Orbit
ISRO	Indian Space Research Organisation
LEO	Low Earth Orbit
LEOP	Launch and Early Orbit Phase
LV	Launch Vehicle
MRD	Mission Requirement Document
P/L	Payload
PSLV	Polar Satellite Launch Vehicle
QSL	Quasi Static Loads
RD	Reference Document
S/C	Spacecraft
S/S	Subsystem
SSO	Sun Synchronous Orbit
TBC	To Be Controlled
TBD	To Be Defined

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3. DOCUMENTS

3.1 Applicable Documents

[AD 1] ASI, “Progetto Galileo Galilei-GG Fase A-2, Capitolato Tecnico”, DC-IPC-2007-082, Rev. B, 10-10-2007 and applicable documents defined therein

3.2 Standards

[SD 1] ECSS-M-00-02A, Space Project Management – Tailoring of Space Standards, 25 April 2000

[SD 2] ECSS-E-ST-10C, Space Engineering - System Engineering General Requirements, 6 March 2009

[SD 3] ECSS-E-10-02A, Space Engineering – Verification

[SD 4] ECSS-Q-00A, Space Product Assurance - Policy and Principles, and related Level 2 standards.

3.3 ASI Reference Documents

[RD 1] GG Phase A Study Report, Nov. 1998, revised Jan. 2000, available at: http://eotvos.dm.unipi.it/nobili/ggweb/phaseA/index.html

[RD 2] Supplement to GG Phase A Study (GG in sun-synchronous Orbit) “Galileo Galilei-GG”: design, requirements, error budget and significance of the ground prototype”, A.M. Nobili et al., Physics Letters A 318 (2003) 172–183, available at: http://eotvos.dm.unipi.it/nobili/documents/generalpapers/GG_PLA2003.pdf

[RD 3] A. Nobili, DEL001: GG Science Requirements, Pisa, September 2008

3.4 GG Phase A2 Study Notes

[RD 4] SD-RP-AI-0625, GG Final Report / Satellite Detailed Architecture Report, Issue 1

[RD 5] SD-RP-AI-0626, GG Phase A2 Study Executive Summary, Issue 1

[RD 6] SD-TN-AI-1163, GG Experiment Concept and Requirements Document, Issue 3

[RD 7] SD-RP-AI-0620, GG System Performance Report, Issue 2

[RD 8] SD-TN-AI-1167, GG Mission Requirements Document, Issue 2

[RD 9] SD-RP-AI-0590, GG System Concept Report (Mission Description Document), Issue 3

[RD 10] SD-SY-AI-0014, GG System Functional Specification and Preliminary System Technical Specification, Issue 1

[RD 11] SD-RP-AI-0631, GG Consolidated Mission Description Document, Issue 1

[RD 12] SD-TN-AI-1168, GG Mission Analysis Report, Issue 2

[RD 13] DTM, GG Structure Design and Analysis Report, Issue 1

[RD 14] SD-RP-AI-0627, GG Thermal Design and Analysis Report, Issue 1

[RD 15] SD-RP-AI-0268, GG System Budgets Report, Issue 1

[RD 16] SD-RP-AI-0621, Technical Report on Drag and Attitude Control, Issue 2

[RD 17] TL25033, Payload Architectures and Trade-Off Report, Issue 3

[RD 18] SD-RP-AI-0629, Technical Report on Simulators, Issue 1

[RD 19] ALTA, FEEP Thruster Design and Accommodation Report, Issue 1

[RD 20] TASI-FI-44/09, Cold Gas Micro Thruster System for Galileo Galilei (GG) Spacecraft - Technical Report, Issue 1, May 2009

[RD 21] SD-RP-AI-0630, Spin Sensor Design, Development and Test Report, Issue 1

[RD 22] SD-TN-AI-1169, GG Launcher Identification and Compatibility Analysis Report, Issue 1

[RD 23] ALTEC-AD-001, GG Ground Segment Architecture and Design Report, Issue 1

[RD 24] SD-TN-AI-1218, GG Preliminary Product Tree, Issue 1

[RD 25] SD-PL-AI-0227, GG System Engineering Plan (SEP), Issue 2

[RD 26] TAS-I, Payload Development and Verification Plan, Issue 1

[RD 27] SD-PL-AI-0228, GG System Verification and Validation Plan, Issue 1

[RD 28] SD-TN-AI-1219, Report on Frequency Management Issues, Issue 1

[RD 29] SD-RP-AI-0632, GG Mission Risk Assessment And Mitigation Strategies Report, Issue 1

[RD 30] SD-RP-AI-0633, Report on Mission Costs Estimates, Issue 1

3.5 External Reference Documents

[RD 31] VEGA Users Manual, Issue 3 rev. 6, 2006, Arianespace

[RD 32] Polar Satellite Launch Vehicle User Manual, VSSC:PSLV:PM:65:87/4, Issue 3 rev. 6, 2006, ISRO

4. LAUNCHER SELECTION

4.1 Preliminary Spacecraft Design Parameters

In the selection of launcher reference is done to the current mission and spacecraft design results. The main assumptions are the following.

- The spacecraft shall be put to a near-circular and equatorial orbit at an altitude of 600 km.
- Current baseline mass of spacecraft is 400 kg. This value includes the 20% margin.
- Spacecraft body is a cylinder (see Figure 4.1-1 for external configuration view) with the following dimensions:
  - 1.4 m diameter
  - 2.2 m height
- The spacecraft shall be deployed in spinning mode, with spin axis perpendicular to the orbit plane.
- The spacecraft shall meet the launcher mechanical and stiffness requirements.

These results provide a first reference for launcher selection and confirmation of preliminary spacecraft design. If selected launcher requirements are not met by spacecraft performances in existing design, the latter shall be updated or an alternative launch vehicle shall be selected. Of course these considerations need a more detailed analysis, to be done in the study.

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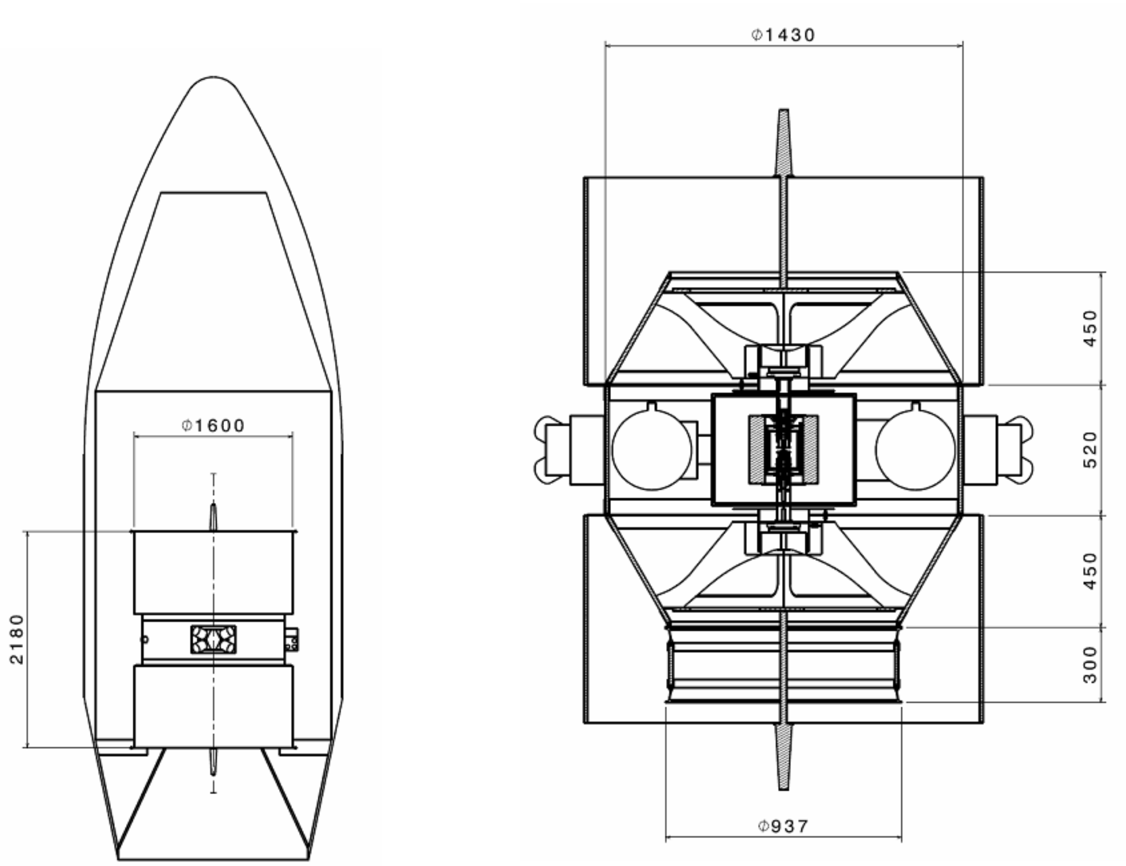

Figure 4.1-1: GG spacecraft configuration and main dimensions

4.2 Selection Criteria

The choice of the launcher will be based on the following criteria:

- Launch cost
- Launcher readiness and availability
- Launch error
- Orbit accessibility

The previous list is not exhaustive. For instance, there are launchers that may provide the necessary launch capability but are too large and expensive to represent a suitable option. These aspects will allow discarding some of the possible launchers that are mentioned in the next paragraph.

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4.3 Launchers Review

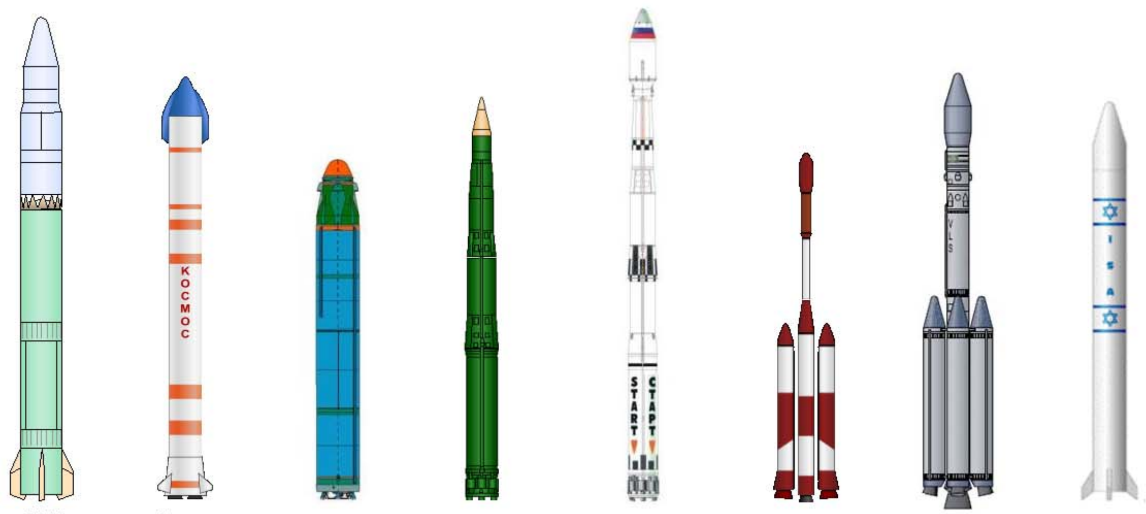
Launchers that could be considered for GG payload may be divided in two different groups:

- Small launchers
- Midi launchers

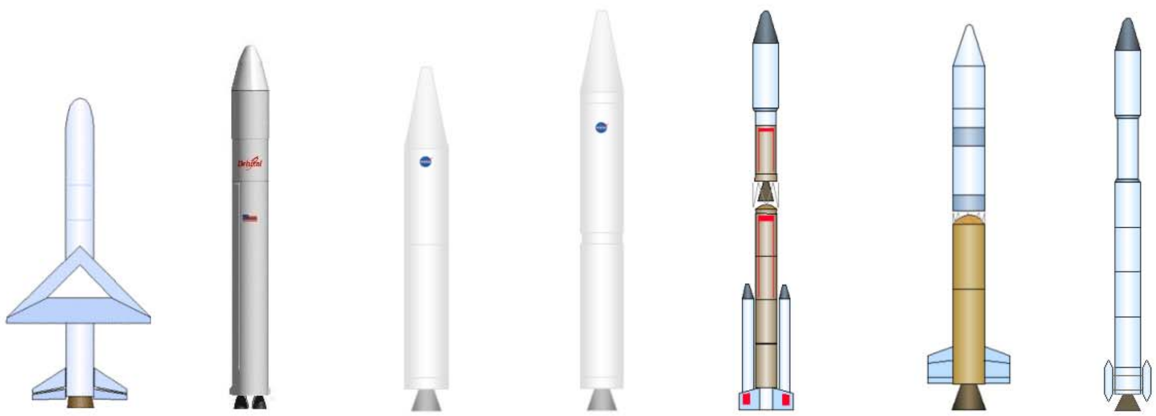
A graphic representation of launchers (not in scale) is reported in Figure 4.3-1 (small launchers) and Figure 4.3-2 (midi launchers). Of course not all the launchers are equally suitable for the mission. As remarked in the previous paragraph, launchers with too larger mass capability, like Soyuz, Delta, CZ3 and CZ4 may be discarded.

In Table 4.3-1 some characteristics of the more commonly used launchers are presented. The table reports the mass launch capability in correspondence of orbit altitude. The list mainly includes midi launchers, with the notable exception of Pegasus. In GG Phase A Study Report [RD 1] Pegasus was taken as reference launcher for the mission. It was considered that a design compatible with Pegasus could be readily adapted to other launchers providing larger margins. Detailed launch capability dependence on orbit altitude for the four more commonly used launchers (Dnepr, Eurockot, VEGA, PSLV) is shown in Figure 4.3-3.

Data shown in Figure 4.3-3 and Table 4.3-1 provide a first indication about the performance of the launchers and their suitability for the mission. However, launch into an equatorial orbit, as required by GG, needs additional investigation, specific to each launcher, since the minimum inclination is constrained by the latitude of the launch site. Therefore the preliminary conclusions reported in the present document may be updated in the study, although there is confidence that general selection results will be confirmed.



Name	CZ-1D	Cosmos	Volna	Start-1	Start	ASLV	VLS	Shavit
Nation	China	Russia	Russia	Russia	Russia	India	Brasil	Israel
Propellant	Liquid	Liquid	Liquid	Solid	Solid	Solid	Solid	Solid
P/L LEO (kg)	740	1100	-	360	570	150	350	160



Name	Pegasus	Taurus	Athena-1	Athena-2	M3S-II	M-V	J-1
Nation	USA	USA	USA	USA	Japan	Japan	Japan
Propellant	Solid	Solid	Solid	Solid	Solid	Solid	Solid
P/L LEO (kg)	460	800	800	1990	880	1800	900

Figure 4.3-1: Small launchers review (pictures not in scale)

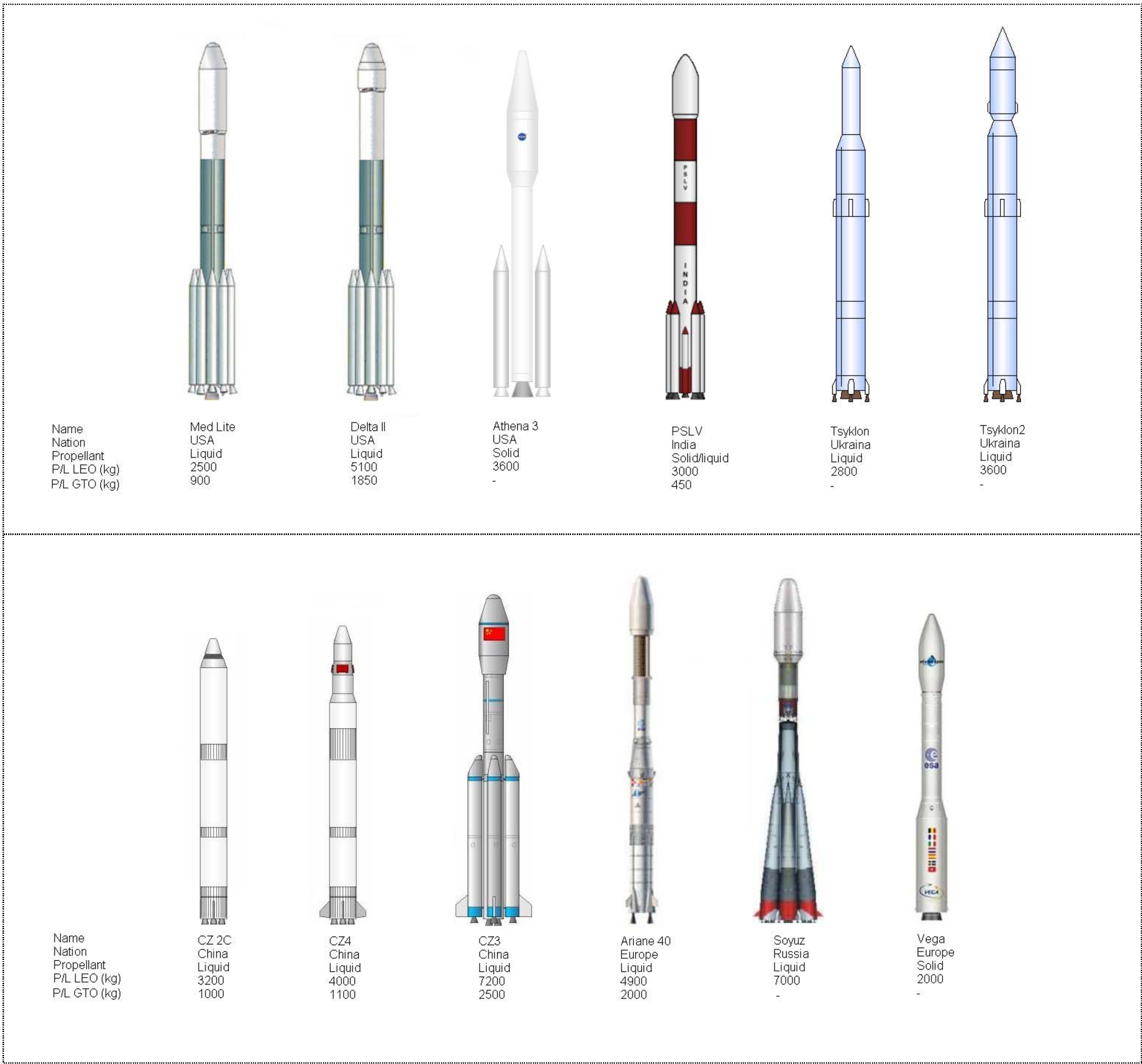


Figure 4.3-2: Midi launchers review (pictures not in scale)

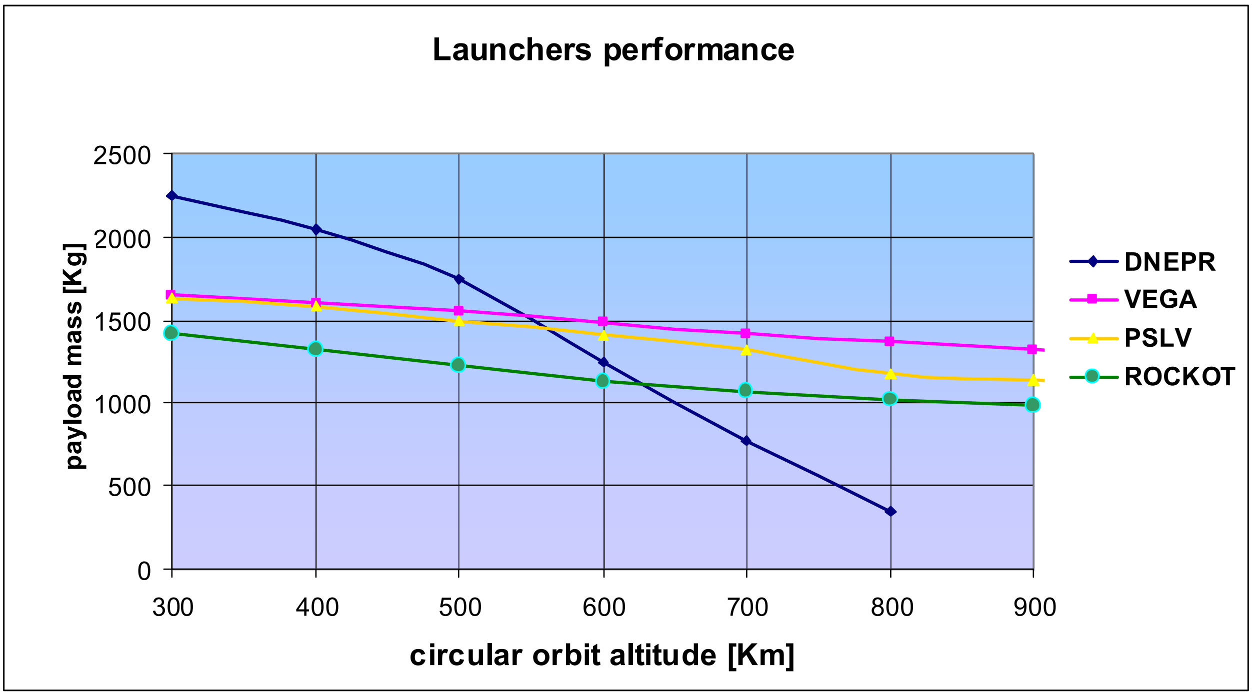


Figure 4.3-3: Payload mass capability for DNEPR, VEGA, PSLV and Eurockot

Launcher	Performance at 500 km [kg]	Fairing usable room [mm]*				QSL envelope [g/s]		Minimum Eigenfrequency [Hz]	
		Φ_1	H_1	Φ_2	H_2	Lateral	Axial	Lateral	Axial
Dnepr	1750	2700	1880	1930	1530	1.0	8.3	10	20
PSLV	1500	2900	2900	1051	2540	1.1	6.4	18	40
Vega	1550	2380	3515	1060	2000	0.9	5.0	15	20.45
Eurockot	1230	2100	3711	697	2424	0.9	8.1	15	33
CZ 2C	1000	3000	3400	TBD	TBD	0.4	6.7	TBD	TBD
Pegasus	380	1118	2139	N/A	N/A	3.5	9.0	20	N/A

Table 4.3-1: Characteristics of possible candidate launchers

**values typical for 650km÷800km SSO*

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4.4 Launchers Selection

4.4.1 Cost assessments

As reported in para.4.4.1 cost is a key parameter to select the candidate launcher. Best performing launcher allowing cost minimization shall be considered.

An indicative list of launchers' costs is reported in Table 4.4-1. Although preliminary, it provides some important indications. Most expensive launchers, like Soyuz and DELTA have been excluded. Indeed they provide a launch capability much higher than that required for GG spacecraft, although a multiple launch may be contemplated, involving other spacecrafts together with GG and therefore allowing cost reduction.

Launcher	Assumed Price (M$)	Assumed Price (M€)*
DNEPR 1	9.5	6.79
ROCKOT KM	13.5	9.64
PSLV	20	14.29
Long March 2C	22.5	16.07
VEGA	25	17.86

** $ to € change rate: 1.4*

Table 4.4-1: Preliminary launcher costs assessment

4.4.2 Availability and readiness

A further reduction of the list of launchers represented in Figure 4.3-1 and Figure 4.3-2 may be done by considering the really available launchers. The following considerations can be done:

- Some launchers, like Indian small launcher ASLV, Ukrainian Tsyklon2 and USA midi launcher Athena, have been retired. In some cases retirement is due to the development of a new generation of launchers of the same family, like Ariane 5 that will take the place of Ariane 40. However, retired launchers cannot be taken in account for a launch that will take place several years in the future.

- Although all the launchers are theoretically available for a scientific research mission, strategic and political reasons may limit the availability. For instance, recent stress in international relations between Russian Federation and western countries could do the use of a Russian launcher a less probable choice than in past years. In addition, it is realistic to assume that a scientific research mission promoted by Italian institutions of research and space agency will primarily take in account European launchers, like VEGA.

4.4.3 Launch error

Another element to take in account is the percentage of errors in past launches with the candidate launchers. Of course it is expected that failures occur in the first launches and then technical problems are solved. Indeed the percentage of errors should considered taking in account the total number of launches. For instance, Ariane 4 has accomplished 116 flights with a success rate of more than 97%. PSLV has a lower success rate (86%) but on a total of 14 launches and the two failures occurred only at the beginning of its career. The most reliable modern launcher was the Ukrainian (former Soviet) Tsyklon2, but it was retired.

4.4.4 Launch orbit

The spacecraft shall be put in an equatorial orbit at 520 km. Therefore the candidate launch vehicles must either be launched from an equatorial site, or a plane-change manoeuvre must be introduced in the ascent flight plan.

4.4.5 Conclusions

For the choice reference can be made once again to the conclusions of former GG Study Report [RD 1]. Although the study was done with reference to Pegasus launcher, its analysis has demonstrated that VEGA launcher provides larger margins not only for mass launch and fairing capability, but also for mechanical requirements. Therefore VEGA may be assumed as the baseline option for GG launcher.

As backup option, it is reasonable to consider a launcher with the same performances of VEGA and analogous characteristics for cost and availability, taking in account the considerations presented in previous paragraphs. The choice is for PSLV launcher, having a quite similar launch capability to VEGA (see Figure 4.3-3). Therefore these two launchers will be considered as reference options for GG mission.

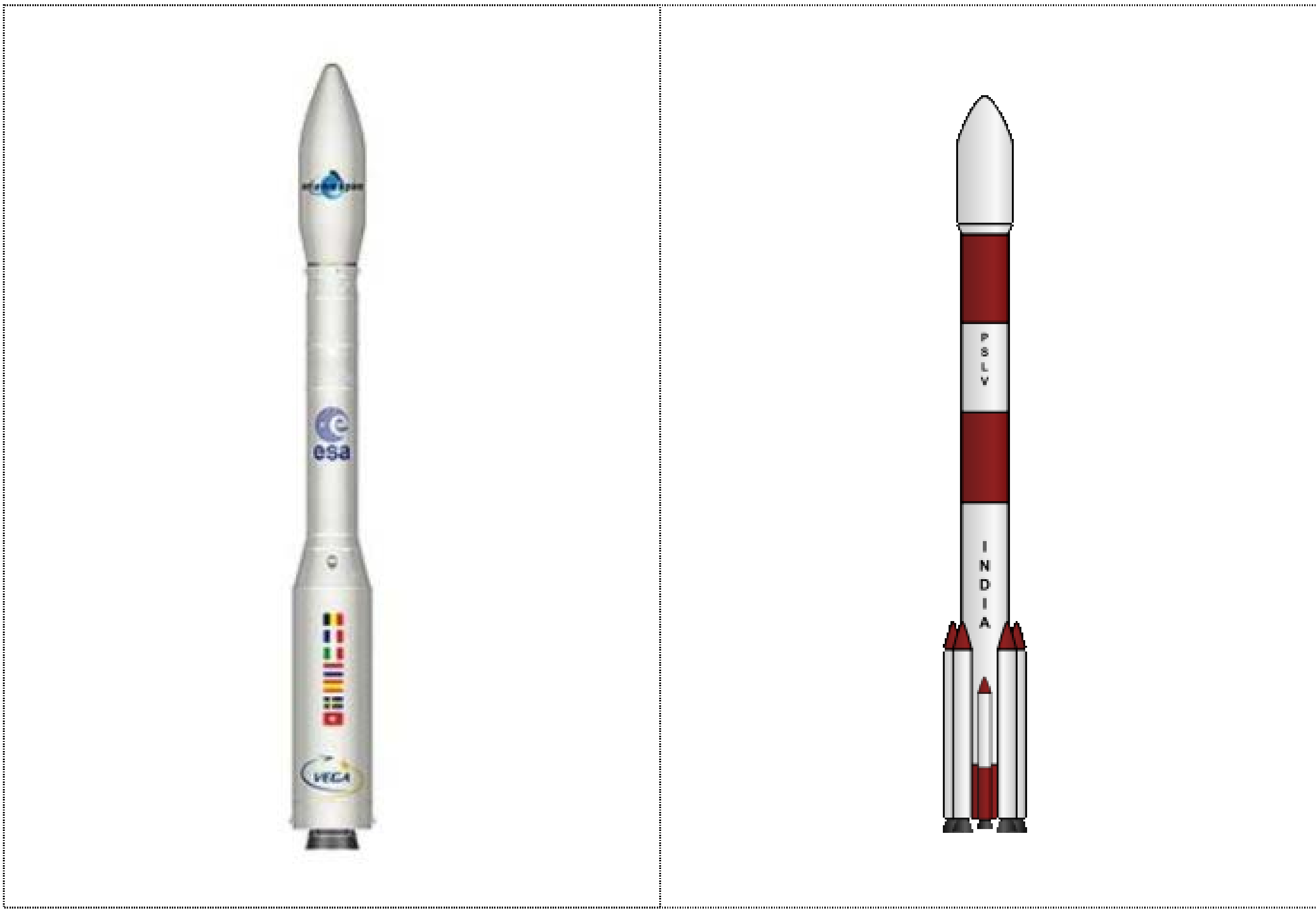

Figure 4.4-1: VEGA launcher (baseline option, at left) and PSLV (back-up option, at right)

5. CANDIDATE LAUNCHERS COMPATIBILITY ANALYSIS

5.1 Overview

The two purposed launchers have been identified following high-level considerations referred to the items listed in para.4.1 and selection criteria of para.4.2. Then a more detailed although preliminary analysis of their compatibility to mission needs can be done.

Launchers description and performance data are reported in their user manuals [RD 31] and [RD 32].

5.2 VEGA launcher (baseline)

5.2.1 VEGA overview

VEGA program has started in the early 1990s, to complement the Ariane launchers family with a small launch vehicle exploiting Ariane solid boosters' technology. Initially a fully Italian program, it has then been proposed by ASI and Italian space industry as a European project. Indeed VEGA has not yet been tested in a real operative mission. However, realistic tests have proved its functionality. As for the reliability, VEGA production benefits from reuse of already developed part in the framework of other programs as well as some off-the-shelf subsystems, components and materials. This design logic allows establishing reliability of the system at the highest level of 98% ([RD 31]).
VEGA launcher vehicle (LV) essentially consists of the following elements:

- A lower composite consisting of three solid propellant stages (from first to third) and a restartable Attitude and Vernier Upper Module (AVUM).
- An upper composite consisting of a payload fairing and a payload adapter/dispenser with separation system(s).

Baseline VEGA technical data are presented in Table 5.2-1. A graphic representation of the launcher evidencing the different stages is shown in Figure 5.2-1.
Launch ground facility is the Guyana Space Centre at Kourou. The satellite/launch vehicle integration and launch are carried out from launch sites dedicated for Ariane, Soyuz or VEGA.

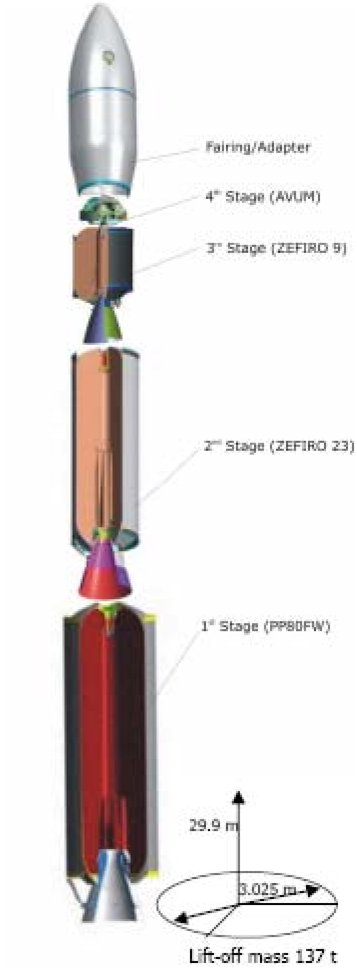

Figure 5.2-1: VEGA launch vehicle schematic architecture

PAYLOAD FAIRING	
Fairing	
Diameter:	2.600 m
Length:	7.880 m
Mass:	490 kg
Structure:	Two halves - Sandwich panels CFRP sheets and aluminum honeycomb core
Acoustic protection:	Thick foam sheets covered by fabric
Separation	Vertical separations by means of leak-proof pyrotechnical expanding tubes and horizontal separation by a clamp band

PAYLOAD ADAPTERS		
Off-the-shelf devices:	Clampband, Ø937	(60 kg);

DUAL CARRYING STRUCTURE	
Off-the-shelf devices:	Under development

MINI SATELLITE CARRYING STRUCTURE		
Off-the-shelf devices:	ASAP Plate type	(TBD kg);

AVUM UPPER STAGE	
Size:	2.18-m diameter × 2.04-m height
Dry mass:	418 kg (TBC)
Propellant:	367-kg/183-kg of N_2O_4/UDMH
Subsystems:	
Structure:	Carbon-epoxy cylindrical case with 4 aluminum alloy propellant tanks and supporting frame
Propulsion	RD-869 - 1 chamber
- Thrust	2.45 kN - Vac
- Isp	315,5 s - Vac
- Feed system	regulated pressure-fed, 87l (3,72 kg) GHe tank MEOP 310 bar
- Burn time/ restart	Up to 667 s / up to 5 controlled or depletion burn
Attitude Control	
- pitch, yaw	Main engine 9 deg gimbaled nozzle or four 50-N GN_2 thrusters
- roll	Two 50-N GN_2 thrusters
- propellant	GN_2; 87l (26 kg) GN_2 tank MEOP 6 / 36 bar
Avionics	Inertial 3-axis platform, on-board computer, TM & RF systems, Power

.	1st STAGE	2nd STAGE (CORE)	3rd STAGE
Size:	3.00-m diameter × 11.20-m length	1.90-m diameter × 8.39-m length	1.90-m diameter × 4.12-m length
Gross mass:	95 796 kg	25 751 kg	10 948 kg
Propellant:	88 365-kg of HTPB 1912 solid	23 906-kg of HTPB 1912 solid	10 115-kg of HTPB 1912 solid
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	P80FW Solid Rocket Motor (SRM)	ZEFIRO 23 Solid Rocket Motor	ZEFIRO 9 Solid Rocket Motor
- Thrust	2261 kN – SL	1196 kN – SL	225 kN - Vac (TBC)
- Isp	280 s – Vac	289 s – Vac	295 s – Vac (TBC)
- Burn time	106,8 s	71,7 s	109.6 s
Attitude Control	Gimbaled 6.5 deg nozzle with electro actuator	Gimbaled 7 deg nozzle with electro actuator	Gimbaled 6 deg nozzle with electro actuator
Avionics		Actuators I/O electronics, power	Actuators I/O electronics, power
Interstage/Equipment bay:	**0/1 interstage:** Structure: cylinder aluminum shell/inner stiffeners Housing: Actuators I/O electronics, power		
	1/2 interstage: Structure: conical aluminum shell/inner stiffeners Housing: TVC local control equipment; Safety/Destruction subsystem	**2/3 interstage:** Structure: cylinder aluminum shell/inner stiffeners Housing: TVC local control equipment; Safety/Destruction subsystem	**3/AVUM interstage:** Structure: cylinder aluminum shell/inner stiffeners Housing: TVC control equipment; Safety/Destruction subsystem, power distribution, RF and telemetry subsystems
Stage separation:	Linear Cutting Charge/Retro rocket thrusters	Linear Cutting Charge/Retro rocket thrusters	Clamp-band/ springs

Table 5.2-1: VEGA technical data (source: [RD 31])

5.2.2 *VEGA orbital delivery performances*

VEGA has the capability of delivering payloads into the sun synchronous orbits (SSO), polar circular orbits, or circular orbits of different inclination. This confirms VEGA suitability for GG mission, involving an equatorial circular orbit. The launcher may also deliver payloads in elliptical orbits delivering in elliptical orbits.

Payload mass delivery performances vs. orbit inclination for circular orbits at different altitudes are shown in Figure 5.2-2. It is evident that VEGA provides a huge margin for GG spacecraft and the possibility of multiple launches, involving other payloads together with GG, exists. This represents an advantage for the possible cost reductions.

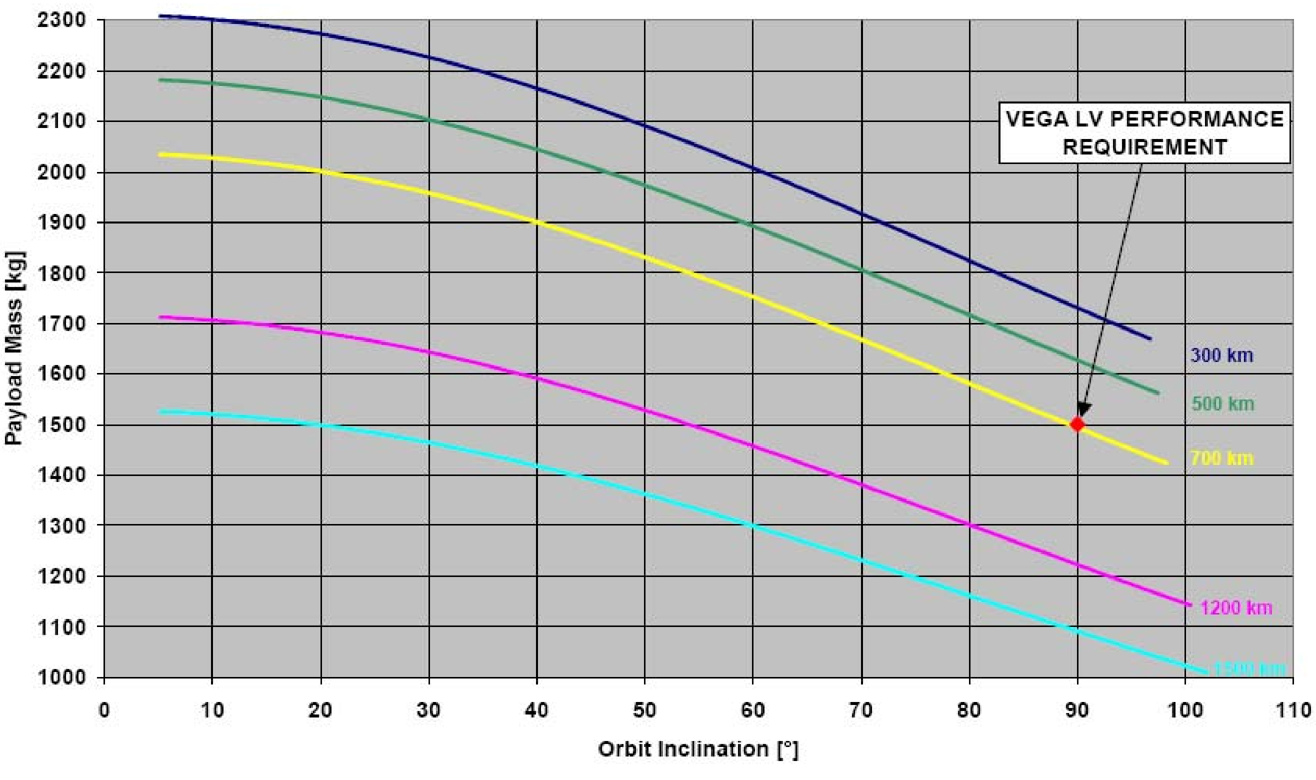

Figure 5.2-2: VEGA performances for circular orbits (source: [RD 31])

5.2.3 VEGA attitude performances for separation conditions

On-board launcher attitude control equipment allows achieving fixed attitude or spun mode pointing. According to these two different situations, spacecraft separation can be done with the two following modes:

- Three-axes stabilized mode
- Spin stabilized mode

Typical VEGA performances for attitude accuracy in the two modes are listed in Table 5.2-2 (3σ values). For the spin stabilized mode values corresponding to 30 deg/s spin are reported. Of course all these values are only indicative. To retrieve the real pointing performance parameters for GG mission, analysis taking in account GG spacecraft mass and real orbit parameters is required.

Injection accuracy for a circular orbit at 700 km (as reported in [RD 31]) corresponds to a possible altitude error of 5 km and inclination error of 0.05 deg (1σ values). Of course these data are not representative of all the possible missions, although they give an indication of system performances. GG mission-specific injection accuracy will be calculated as part of the mission analysis.

Three-axes stabilized mode	**3σ value**
Longitudinal axis depointing	≤ 1 deg
Transversal axis depointing	≤ 1.5 deg
Angular tip-off rates along longitudinal axis	≤ 0.6 deg/s
Angular tip-off rates along transversal axis	≤ 1.0 deg/s
Spin stabilized mode (30 deg/s spin)	**3σ value**
Spin rate accuracy	≤ 1 deg/s
Transverse angular tip-off rates	≤ 0.6 deg/s
Nutation, half angle	≤ 5 deg
Injection accuracy (700 km orbit)	**1σ value**
Apogee/perigee error	TBD
Altitude error	5 km
Inclination error	0.05 deg

Table 5.2-2: VEGA attitude and orbit injection accuracy parameters

5.2.4 *Requirements for launched spacecraft*

VEGA user manual [RD 31] reports in detail several requirements for the spacecraft to be launched. Mainly interesting requirements for GG mission are related to spacecraft mechanical and stiffness properties. They are briefly illustrated in the following.

Spacecraft mechanical stiffness
To prevent dynamic coupling with LV fundamental modes, the spacecraft structural stiffness shall fulfil the following requirements for fundamental mode frequencies (that apply to hard-mounted spacecrafts with mass lower than 2500 kg and with an off-the-shelf adapter):

- Lateral axis: F ≥ 15 Hz
- Longitudinal axis: 20 Hz ≤ F ≤ 45 Hz

Mechanical design load factors
They are represented by the quasi-static g-loads (QSL) i.e. the more severe combinations of dynamic and steady-state accelerations encountered at any instant of the mission (ground and flight operations). QSL flight limits for a spacecraft launched on VEGA and complying with frequency requirements and static moment limitations are reported in Table 5.2-3.

Load Event	QSL (g) (+ = tension; – = compression)					
	Longitudinal			Lateral		
	Static	Dynamic	Total	Static	Dynamic	Total
Lift-off phase	– 1.5	± 2.0	Min – 3.5 Max + 0.5	-	-	± 0.9
Flight with maximum dynamic pressure (Qmax)	– 2.5	± 0.5	Min – 3.0 Max – 2.0	-	-	± 0.9
First-stage flight with maximal acceleration	– 4.5	± 0.5	Min – 5.0 Max – 4.0	-	-	± 0.5
Third stage maximal acceleration	-4.5	± 0.2	Min – 4.7 Max – 4.3	-	-	± 0.2
Stages ignition		-5 3	Min – 5.0 Max + 3.0	-	-	± 0.2

Table 5.2-3: VEGA flight limit levels- QSL

5.3 PSLV launcher (alternative option)

5.3.1 PSLV overview

The Polar Satellite Launch Vehicle (PSLV) is a four-stage launch vehicle designed and developed by Indian Space Research Organisation (ISRO) with the participation of Indian industries and institutions. The four stages use solid and liquid propulsion systems alternately. The first and third stage use solid propellant and the second and fourth stage use liquid propellant. Although it was primarily designed to inject 1000 kg class spacecraft into a 900 km SSO, PSLV can also perform launches to Low Earth Orbits (LEO) as well as Geo-synchronous Transfer Orbits (GTO).

PSLV has repeatedly proved its reliability and versatility by launching 30 spacecrafts (14 Indian and 16 for international customers) into a variety of orbits so far. Most recent launch has been done on October 22nd 2008, in the frame of first Indian Moon exploration mission.

Cooperation with European space agencies is well consolidated. PSLV has been employed to launch PROBA3 and AGILE satellites, as well as a plurality of small and mini-satellites, including German, Dutch and Danish ones in a single launch on April 28th 2008. In particular, AGILE was a scientific satellite by ASI for the observation of X and Gamma rays, slightly heavier than GG.

A schematic representation of PSLV internal architecture, with the indication of vehicle parts and components, as reported in [RD 32], is presented in Figure 5.3-1.

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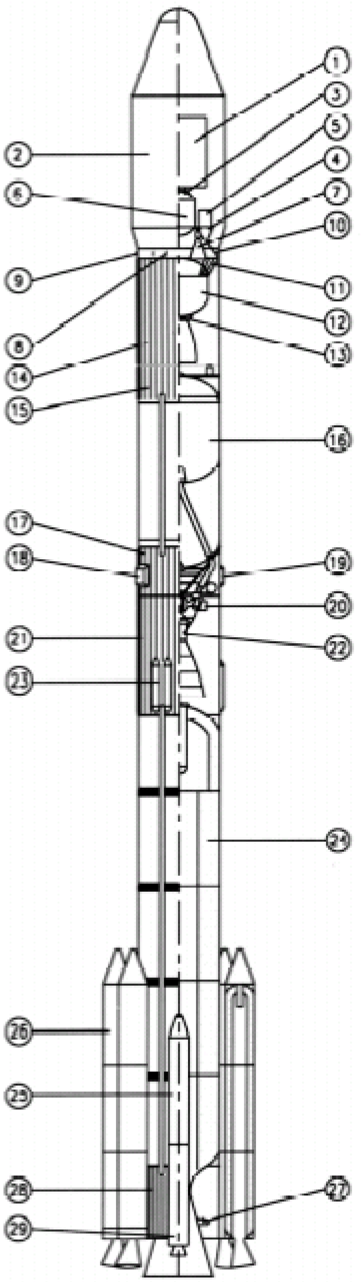

1. PAYLOAD
2. HEATSHIELD
3. PAYLOAD ADAPTOR
4. EQUIPMENT BAY
5. AUXILIARY PAYLOAD
6. FOURTH STAGE PROPELLANT TANK
7. FOURTH STAGE ENGINE (2)
8. ANTENNAE
9. REACTION CONTROL THRUSTER (6)
10. INTERSTAGE 3/4
11. THIRD STAGE ADAPTOR
12. THIRD STAGE MOTOR
13. FLEX NOZZLE CONTROL SYSTEM
14. INTERSTAGE 2/3U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

Figure 5.3-1: PSLV launch vehicle schematic architecture (source: [RD 32])

5.3.2 PSLV orbital delivery performances

PSLV payload mass delivery performances vs. orbit inclination for circular orbits at different altitudes are shown in Figure 5.3-2. As anticipated, its launch capability is enough large to allow accommodating more small and medium payloads, so the possibility of a multiple launch exists, like for VEGA. Indeed the performance profiles in [RD 32] have not been traced for inclination values down to 0 deg i.e. equatorial orbit conditions, but it may be assumed that launch capability corresponds to that at 30 deg.

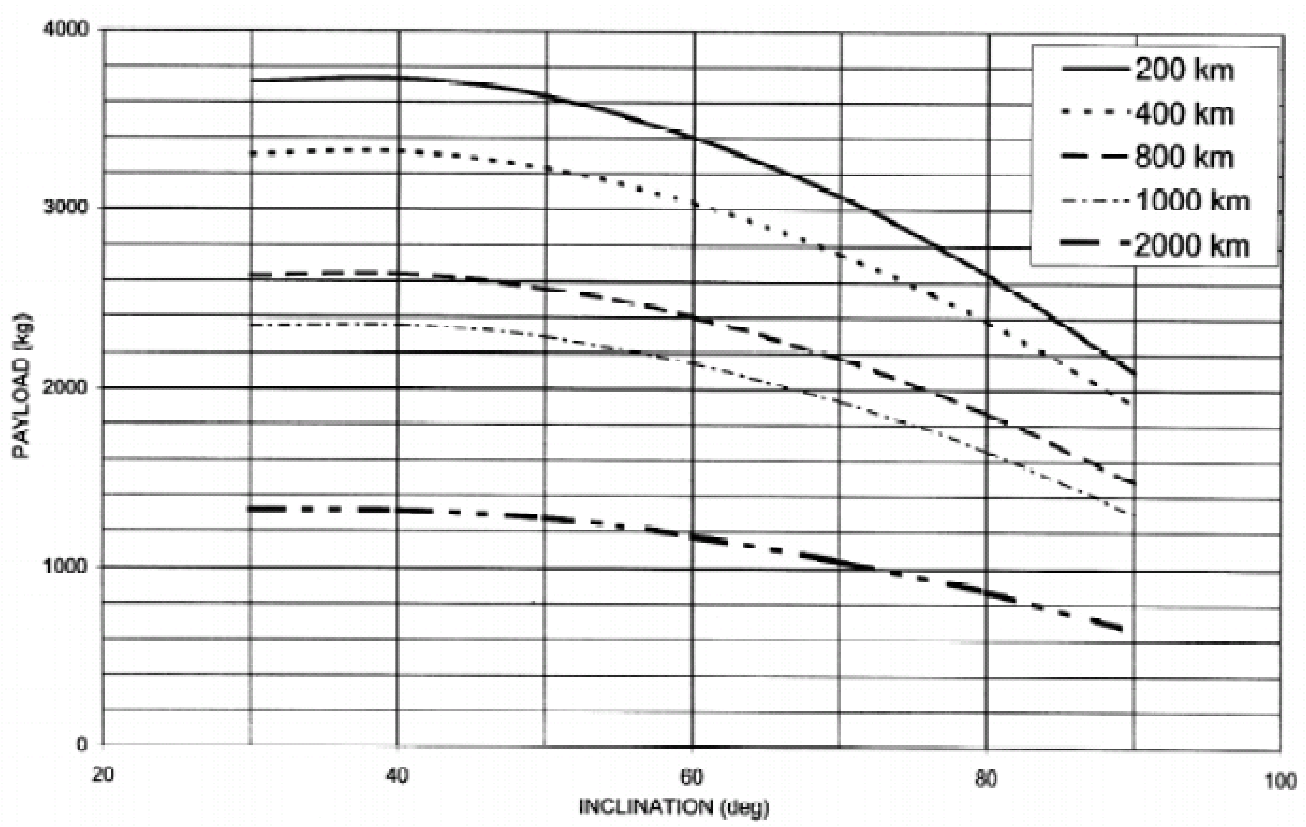

Figure 5.3-2: PSLV performances for circular orbits (source: [RD 32])

5.3.3 PSLV attitude performances for separation conditions

Spacecraft injection in the specified orbit principally occurs in three axes stabilised mode. For a SSO orbit at 817 km altitude the orbital parameters errors (apogee/perigee and inclination) are those reported in Table 5.3-1. The pointing accuracy value depends on the attitude stabilization and control equipment of the launcher. In the table there is also the separation disturbance on the spacecraft, as reported in [RD 32] to characterise the performances of spacecraft separation system.

Three-axes stabilized mode	3σ value
Longitudinal axis depointing	≤ 0.5 deg
Transversal axis depointing	≤ 0.5 deg
Angular tip-off rates along longitudinal axis	≤ 2 deg/s
Angular tip-off rates along transversal axis	≤ 2 deg/s
Spin stabilized mode	**3σ value**
Spin rate accuracy	TBD
Transverse angular tip-off rates	TBD
Nutation, half angle	TBD
Injection accuracy (817 km orbit)	**1σ value**
Apogee/Perigee error	± 35 km
Altitude error	TBD
Inclination error	± 0.2 deg

Table 5.3-1: PSLV attitude and orbit injection accuracy parameters

5.3.4 Requirements for launched spacecraft

As for VEGA launcher, PSLV requirements for launched spacecraft of main interest deal with mechanical design and stiffness. They are briefly summarised in the following.

Spacecraft mechanical stiffness
To avoid dynamic coupling between the low frequency excitation and spacecraft modes, the stiffness of the spacecraft structure shall be designed to keep the fundamental frequencies (hard-mounted at launcher interface) in the following values ranges:

- Lateral axis: F ≥ 20 Hz
- Thrust (longitudinal) axis: F ≥ 35 Hz

Mechanical design load factors
The maximum static and dynamic accelerations occurring at spacecraft interface during each stage are defined as follows:

- Longitudinal: 7 g / - 2.5 g
- Lateral: ± 1.5 g
- Ultimate load factor: 1.25

These loads are to be taken for design, analysis and structural qualification tests for spacecraft to be launched with PSLV.

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6. COMPATIBILITY ASSESSMENT

6.1 Overview

In the assessment of compatibility of GG spacecraft to the two candidate launchers some preliminary considerations may be expressed. First of all, the selection has been done taking in account current GG spacecraft design as reference for the comparison to available spacecraft envelope and launcher accommodation capability of candidate launchers. The mass and size of the spacecraft has allowed discarding the largest launchers and identifying medium size launchers as possible candidates. Then reliability, cost and availability considerations have allowed limiting the choice to two possible options, VEGA (baseline) and PSLV (back-up) launchers. Further considerations about compatibility between GG spacecraft and purposed launchers are illustrated in the following.

6.2 Compatibility Assessment Approach

Several aspects must be taken in account to assess the compatibility between GG spacecraft and selected launchers:

- Mass and size, as defined in current design, are compliant with the corresponding launcher requirements for spacecraft. Indeed multiple launches may be recommended, because of the potential cost reductions, given the limited size and mass of the GG spacecraft.

- The compatibility to launcher mechanical environment requires dedicated analysis. The current evaluation of GG spacecraft mechanical and stiffness performances is reported in [RD 13]. This analysis indicates full compliance with the VEGA launcher requirements.

- Another potentially critical aspect is the possibility of orbital delivery of a spinning spacecraft, as required by the experiment. The current satellite design is compatible with both spinning and 3-axis stabilized release, and includes the capability to reach the required spin rate and direction of the axis by the spacecraft's own systems. Should any of the envisaged launchers possess the capability to deliver the spacecraft with the required pointing and spin rate, this would constitute an asset for the launcher under consideration.

Further verification of the above mentioned issues is recommended for Phase B.

7. CONCLUSIONS

In the present document a review of possible candidate launchers for GG mission has been done. Two candidates have been selected: the baseline candidate (European VEGA) and a back-up launcher (Indian PSLV). Both the launchers present characteristics suitable to the needs of the mission. Based on the available information, compatibility of either launcher with the GG spacecraft is assured.

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