CAS-ESA Call for Mission Proposals
Technical annex

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1 INTRODUCTION

1.1 Background

This document is an Annex to the ESA-CAS Call for Proposals for a small class mission that would be jointly developed by ESA and CAS for a launch in 2021.

The Annex provides technical background information for supporting the scientists in Europe and China in building their proposals, including:

- Some technical boundaries (e.g. on mass, power etc) that are deemed necessary for satisfying the Call programmatic constraints and having a mission that could actually be implemented. It is strongly recommended to the proposers to meet these boundaries, although a departure from this constraints can be tolerated with due justification.
- Background information on potential launchers,
- Examples of mission profiles,
- General technical inputs that may be useful for the proposers, e.g. on data rates, space debris mitigation.

Additional technical information on ESA science spacecraft in orbit or under development and on study cases (Cosmic Vision) can be found at the following link: http://sci.esa.int/home/51459-missions/

1.2 Reference documents

[1] LM-2C User's manual, issue 1999
[2] Vega User's manual, issue 4.0, 2014
[3] Soyuz User's manual, issue 2.0, 2012
[4] Requirements on space debris mitigation for ESA projects, IPOL(2008)2 Annex 1
[5] ISO/CD 16290, Space system – Definition of the TRL and their criteria of assessment 2012
[6] ESTRACK facilities manual (EFM), issue 1.1, 2008

1.3 List of acronyms

AOCS	Attitude and Orbit Control System
ASAP	Arianespace System for Auxiliary Payloads
AVUM	Attitude and Vernier Upper Module (Vega's upper stage)
BER	Bit Error Rate
CAS	Chinese Academy of Science
CHEOPS	Characterising Exoplanet Satellite
EoL	End of Life
ESTRACK	ESA tracking stations network
GEO	Geo-stationary Earth Orbit
GPS	Global Positioning System
GSC	Guiana Space Centre
GTO	Geo-stationary Transfer Orbit
I/F	Interface
ISO	International Organization for Standardization
ITAR	International Traffic in Arms Regulation
JSLC	Jiuquan Satellite Launch Center
LEO	Low Earth Orbit
LM	Long March
LV	Launch Vehicle
MEO	Medium Earth Orbit
MOC	Mission Operations Centre
NSSC	National Space Science Centre
P/L	Payload
PM	Propulsion Module
QUESS	Quantum Experiments at Space Scale
RAAN	Right Ascension of Ascending Node
rms	root mean square
SOC	Science Operations Centre
S/C	Spacecraft
SPELDA	Structure Porteuse Externe pour Lancement Double Ariane
SPELTRA	Structure Porteuse Externe pour Lancement Multiples
SSO	Sun Synchronous Orbit
TRL	Technology Readiness Level
TSLC	Taiyuan Satellite Launch Center
VESPA	Vega Secondary Payload Adapter
XSLC	Xichang Satellite Launch Center

2 GENERAL CONSIDERATIONS

2.1 Mass and power

The Call targets a “small class” mission. As general guidelines, the following boundaries will be considered:

- Spacecraft wet mass: ≤ 300 kg
- Payload mass: ≤ 60 kg
- Payload average power consumption: < 65 W (peak < 100 W)

The spacecraft wet mass is the full space segment mass excluding:

- The launch adapter
- The propulsion module if any for the orbit transfer.

2.2 Schedule

A relatively fast implementation is contemplated:

- Mission selection in 2015
- Joint definition phase < 2 years
- Space segment development < 4 years
- Launch in 2021
- Operational lifetime: 2-3 years

2.3 Work breakdown and share

Any of the mission elements below can in principle be provided by China, Europe, or jointly. The work share shall preserve clean and manageable interfaces.

- Mission Architect
- Platform
- System Integration and testing
- Launch services
- Spacecraft operations (MOC)
- Ground stations
- Science operations (SOC)
- Science exploitation

The following elements are expected to be jointly achieved:

- Overall mission management
- Science Management and exploitation
- Science payload

2.4 Space debris mitigation

The mission concept has to ensure that no additional orbital debris will contaminate the protected regions (Figure 1). The practical consequence is the need to implement a propulsion subsystem, even when using low-Earth orbits, for either moving the S/C into graveyard orbits at its end of life, or to ensure its re-entry in the atmosphere within a specified maximum duration of 25 years.

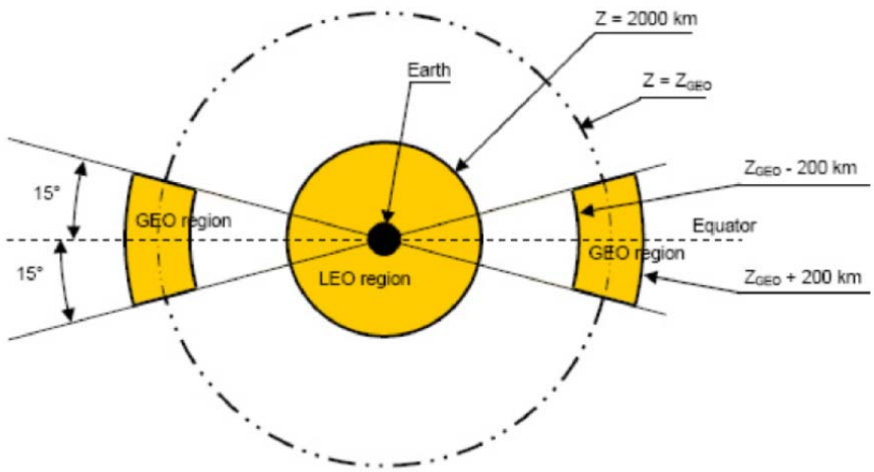


Figure 1: LEO and GEO protected regions [4].

When fragments of the S/C may survive the re-entry (typically for large missions), a controlled re-entry manoeuvre has to be performed to mitigate the risk of ground casualty. This is typically the case for medium to large (typically > 1000 kg) LEO missions. For small missions as is the case here, an un-controlled re-entry is acceptable, as long as it happens within 25 years. This requirement applies to the S/C, but also to any other debris generated by the mission, such as LV upper stages, multi-S/C adapters, ejectable covers etc.

Further details are provided in Appendix C.

2.5 Technology Readiness

The spacecraft development schedule (including the payload) shall be achievable in ~ 3.5 years maximum. This requires the instrumentation and the platform to rely on available technologies (ISO TRL ≥ 6 for the payload and ≥ 7 for the platform elements, see definition in Appendix E). This means the payload must rely on existing heritage, although it could be a new development, while the platform is expected to be adapted from existing ones with flight qualified equipment.

Care must be taken with potential obsoleteness of components and subsystems for proposed platforms and payload when referring to heritage.

2.6 Export control constraints

Proposed missions have to take export control constraints into consideration. In particular, the spacecraft and its payload shall be free of any export control constraints from European countries –including those resulting from US ITAR regulations - for enabling the spacecraft launch from China.

3 POTENTIAL LAUNCH VEHICLES AND EXAMPLE OF MISSION PROFILES

The Chinese and European launchers that can reasonably be considered for this mission are the following:

- Long March launchers LM-2C or LM-2D, both launched from China,
- European launchers, namely Vega, Soyuz and Ariane 5, launched from the Guyana Space Centre (CSG in Kourou). Note that for Soyuz or Ariane 5 cases, only an auxiliary/piggy-back passenger launch can be envisaged.

Launcher performance curves are provided in Appendix A. Further details are available in the respective user manuals [1 to 3] (for European launchers, see www.arianespace.com).

When combined with the launcher performance, the mass constraint given in section 2.1 implies that the two following conditions must be satisfied:

- $M_{S/C\_wet} + M_{PM\_wet} + M_{adapter} \leq M_{launch}$
- $M_{S/C\_wet} \leq 300$ kg (mass limit given in section 2.1)

where:

- M_{launch} is the launcher performance (Table 1)
- $M_{adapter}$ is the launcher adapter mass
- M_{S/C_wet} is the spacecraft's wet mass (excluding the adapter)
- M_{PM_wet} is the (potential) propulsion module wet mass (also excluding the adapter)

It is important to note that an auxiliary passenger launch adds constraints on the potential orbits that can be reached, since the main passenger dictates the launcher ascent profile and burns. Therefore, highly specific and specialized orbits are unlikely to be reached with an auxiliary passenger launch, while more common orbits could be envisaged (e.g. SSO or GTO).

A summary of launcher performance is given in Table 1 for some typical science orbits. The Venus case is provided as an illustrative example of a mission to a nearby planet. Table 2 provides additional useful information. Note that exact performance figures will be determined with specific mission analysis activities depending on the proposed mission concepts.

	Vega [with bi-liquid propulsion module]	**Soyuz**	**LM-2C [LM-2C/CTS]**	**LM-2D [LM-2D/TY-2]**
LEO	~ **2.300 kg** @ 300 km (i=5°) **1.480 kg** @ 400 km SSO **1.140 kg** @ 1000 km SSO	**4500 kg** @ 700 km SSO	**1850 kg** @ 400 km SSO [**1650 kg** @ 700 km SSO with LM-2C/CTS]	**2200 kg** @ 400 km SSO [**1550 kg** @ 700 km SSO with LM-2D/TY-2]
HEO	**1.963 kg** @ 200 x 1500 km [~ **650 kg** @ 300 x 36000 km] (both at equatorial i=5.4°)	**3250 kg** in GTO	**3350 kg** @ 200x1000 km (i=29°) [**1250 kg** in GTO with LM-2C/CTS]	**3700 kg** @ 200x1000 km (i=28.5°)
Sun Earth L1/L2 (C3 = 0 km^2/s^2)	[~ **420 kg**]	**2160 kg**	[**820 kg** with LM-2C/CTS]	[**380 kg** with LM-2D/TY-2]
[1]Heading/trailing heliocentric orbits and [2]Sun-Earth L4/L5 (C3 > 0 km^2/s^2)	[1][≤ **400** kg] [2][~ **230-350 kg** for L5 depending on transfer time]	< **2160 kg**	[< **820 kg** with LM-2C/CTS]	[< **380 kg** with LM-2D/TY-2]
Venus, before orbit insertion (C3 ≈ 7.5 km^2/s^2)	[≤ **340 kg**]	**~1780 kg**	[< **420 kg** with LM-2C/CTS]	[< **200 kg** with LM-2D/TY-2]
Venus, after insertion into 2-day HEO	[≤ **240kg** after insertion]	**~1250 kg**	[< **290 kg** with LM-2C/CTS]	[< **140 kg** with LM-2D/TY-2]
Earth escape / interplanetary transfers	See performance as a function of C3 in Appendix A.			

Table 1: Summary of potential LVs and their performance to different orbits.

Orbit	Typical transfer duration	Typical science TM data rates	Power
LEO	< 1 day	X band: 20 – 200 Mbps S band: ~600 kbps	@ 1 AU Solar radiation: ~1300 W/m^2 Cosine loss for 36° off-pointing: 80% Cell efficiency: 28% System losses: 85% Cell packaging ratio: 70% Ageing: 86% (@ 3.75%/year for 4 years) ~150 W/m^2 at EoL
HEO			
Sun Earth L1/L2	~1 month	X band: 5-10 Mbps Ka band: 75 Mbps	
Heading/trailing heliocentric orbits and Sun-Earth L4/L5	14 – 50 months (in increments of 1 year, see details in section A.3)	Ka band: 150 kbps	
Venus	100 – 180 days (conj. transfer) 350 – 450 days (1.5 revolution transfer)	X band: 63 – 228 kbps (superior vs. inferior conjunction)	Approximately 1.9 times the value at Earth Higher temperatures may further reduce the solar cell efficiency.

Table 2:Transfer times, data rates and power generation for the different orbits.

For the specific case of Venus, the following table shows examples of the ΔV required for insertion into a 2-sol orbit, and the resulting inserted S/C mass. The example is based on the Vega case with the bi-liquid propulsion stage, but the mass ratios are independent and can be re-used with any other launcher (as done in Table 1).

Launch date	06/11/2021	07/06/2023
Esc. Velocity [km/s]	3.608	3.127
Esc. Declination[degree]	5	-3.3
S/C wet mass [kg]	**292**	**324**
Venus arrival	24/02/2022	26/10/2023
Venus Orbit Insertion (including gravity losses) [m/s]	879	863
S/C mass in 2 day HEO [kg]	**180**	**240**

Table 3: Venus orbit insertion mission examples with Vega and a bi-liquid propulsion module.

Note: the ~2 day HEO Venus orbit is defined as 300 x 123863 km.From this HEO orbit, reducing the apogee until circularization into a 300 km altitude orbit around Venus will require an additional ΔV increment. This is shown inFigure 2, along with the resulting S/C wet to dry mass ratio.

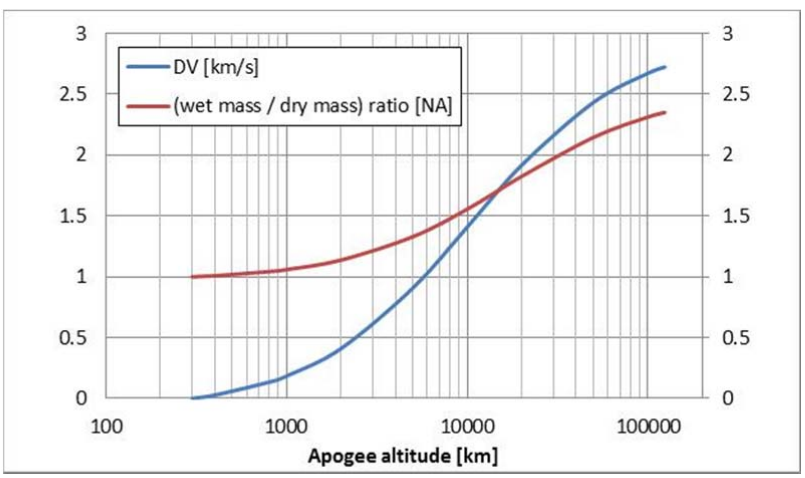


Figure 2: ΔV required to reduce the apogee of a HEO orbit around Venus down to a 300 km altitude circular orbit (blue curve), and resulting wet to dry mass ratio of the S/C to perform this ΔV assuming Isp = 325 s (red curve).

Alternatively, the orbit circularisation could be achieved with aerobraking (as demonstrated by Venus Express), saving a large fraction of the ΔV for Venus orbit insertions. Note that at 300 km altitude, atmospheric drag is non-negligible and needs to be taken into account.

4 SPACECRAFT RELATED ASPECTS

4.1 Spacecraft propulsion subsystem

The ΔV contributors to the spacecraft propulsion subsystem are briefly discussed:

- Orbit transfer and/or insertion

The orbit transfer and insertion are often achieved by the launcher upper stage. However, depending on the launcher capability and the mission case (e.g. planetary missions), it may be necessary to achieve part of the ΔV with the spacecraft propulsion system (e.g. the final insertion), and sometimes to consider a dedicated propulsion module for the orbit transfer and/or for the insertion. The orbit transfer and insertion ΔVs are generally high (a fraction of km/s to a few km/s) and mission dependent.

- Launcher dispersion correction manoeuvres

Correction manoeuvres may be needed following the launcher insertion for compensating for the launcher insertion inaccuracies. Typical ΔV values are in the range of 10-30 m/s.

- Orbit maintenance and specific manoeuvres

Once the spacecraft is on its selected orbit, and depending on the mission tolerance to a drift of the orbit parameters, some ΔV may be required for the orbit maintenance - e.g. for the correction of gravity field perturbations, atmospheric drag– and for collision avoidance manoeuvres. For LEO above 600 km altitude, typical ΔV values range from a few m/s to 10 m/s. Orbits around L1/L2 are known to be unstable and require a maintenance ΔV of 2-8 m/s/year depending on the orbit amplitude. Additional ΔV may be required depending on the specific mission needs. For example, if the Attitude Control System is using reaction wheels, a ΔV may be required for de-saturating the reaction wheels from time to time.

- Spacecraft disposal at End of Life (EoL)

The spacecraft disposal must satisfy Space Debris mitigation requirements.

LEO case is discussed in Appendix C. The EoL ΔV depends on the spacecraft orbit, with typical figure of 40-50 m/s for a circular orbit at 700 km, and 80-100 m/s for a circular orbit at 800 km. The EoL ΔV is often driving the spacecraft propulsion design in LEO. For higher altitudes (> ~1500 km), it is more advantageous to raise the orbit above 2000 km beyond the protected zones (see Appendix C for further details)

For orbits at L1/L2, if no measure is taken, the spacecraft would leave L1/L2 area in an uncontrolled manner, with a ~50% probability of returning to Earth and crossing the protected regions. Analysis performed for previous missions (Herschel/Planck, Gaia) resulted in a 50 m/s allocation to de-orbit the S/C at End of Life, with a 99.8% probability of not returning to Earth in the following 100 years.

4.2 Radiation environment

Models of the space environment and its effects can be found at https://www.spenvis.oma.be/.The energetic particle radiation environment consists of trapped charged particles in the Van Allen radiation belts (electrons and ions trapped by the Earth magnetic field), Solar particles(mainly protons) and Galactic Cosmic Rays. The impact on performance and shielding requirements are generally mission dependent.

As an example of the variation of the radiation environment as a function of orbit and lifetime, Figure 3 shows the Total Ionising Dose (TID) for missions operating in 2 different orbits:

- The HEO orbit is 1.8 x 7 R_{Earth}^2, with an argument of perigee of 270 degree, an inclination of 63.4 degrees, and a 3 year lifetime.
- The orbit around L2 has a high amplitude of about 1.5 Mkm, is attained with a direct transfer strategy from launch, and also has a 3 year lifetime.

For an Al shielding thicknesses < 6 mm, the radiation environment in the HEO orbit exceeds the one in the L2 orbit by 1 to 2 orders of magnitude. This is because the HEO orbit passes through both radiation belts 4 times every day (significant contribution from the trapped protons), while the L2 orbit is dominated by the contribution from Solar particles only. Such an orbit would require significant enhancements on the S/C's radiation tolerance.

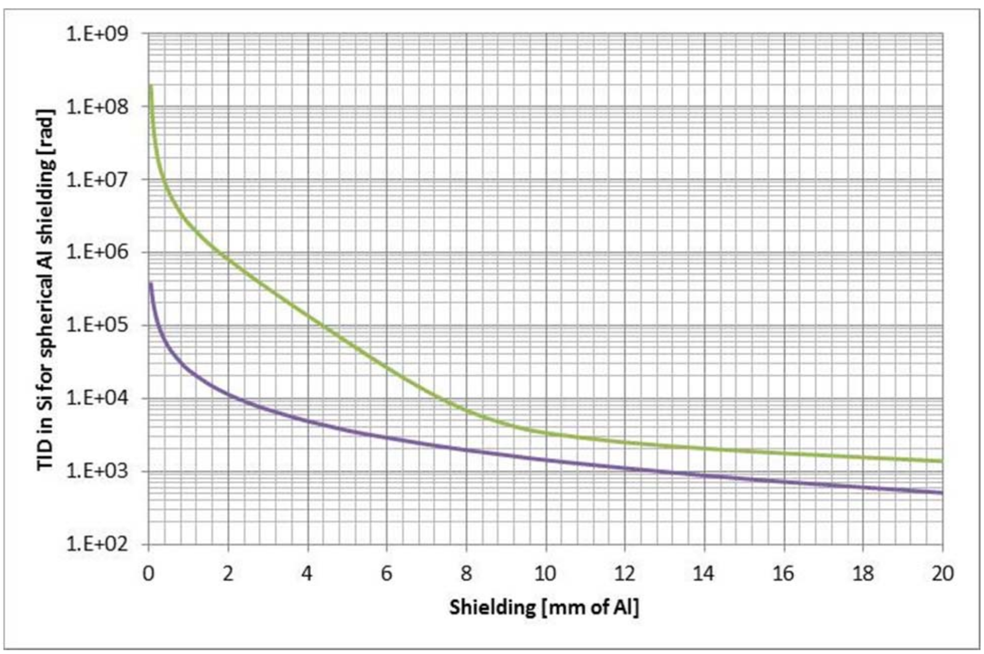


Figure 3: TID for missions in HEO (green curve) and in an L2 orbit (purple curve).

4.3 Data rate aspects

The achievable data rate is an important parameter for science missions. Examples of data rates achieved with previous spacecraft designs are provided:

- For small platforms in LEO/HEO: data rates ~ 20 to 200 Mbit/s achieved in X band, < 10 cm patch, horn, helix or isoflux Low Gain Antenna (LGA), ≤ 10 W output power, and 3 to 15 m ground antenna.
- L2 orbit in X band: 10 Mbps achieved with 30 cm High Gain Antenna (HGA), 30 W output power, and 35 m ESTRACK ground antenna.
- L2 orbit in Ka band: 75 Mbps achieved with 50 cm HGA, 35 W output power, and 35 m ground antenna.
- Planetary mission at 1 AU, in Ka band: 150 kbps achieved with 1.1 m HGA, 35 W output power, and 35 m ground antenna.
- Planetary mission at Venus, in X-band, 1.3 m HGA, 65 W output power, 35 m ground antenna, data rates 63 – 228 kbps (superior – inferior conjunction). Note: at inferior conjunction, the potential data rate achievable is higher and largely exceeds the need of the mission. Therefore the power was reduced (48 W instead of 65 W) and a maximum limit was imposed by the Command and Data Management Subsystem.

The communication link budget is primarily a function of the communication subsystem output power and of the emitting and receiving antennae diameters. For given receiver noise, coding and Bit Error Rate performance, the data rate scales as:

$$\text{Data rate} \propto P_t.(D_t/\lambda)^2.\,(D_r/\lambda)^2.(\lambda/r)^2$$

where:

- P_t is the communication subsystem transmitter power
- D_t (resp. D_r) is the diameter of the transmitting (resp. receiving) antenna
- λ is the communication wavelength
- r is the distance between the spacecraft and the ground station

The proposers can use the above formula for a preliminary evaluation of the achievable data rates, by using the above examples or other spacecraft cases. The actual data volume must take into account the ground station visibility (6 to 8 h per day for a spacecraft at L2 and a single ground station). For example, taking the above mention L2 orbit case in X band, Figure 4 shows how the data rate scales as a function of the S/C to ground station distance, the transmitter power and the transmitting antenna diameter.

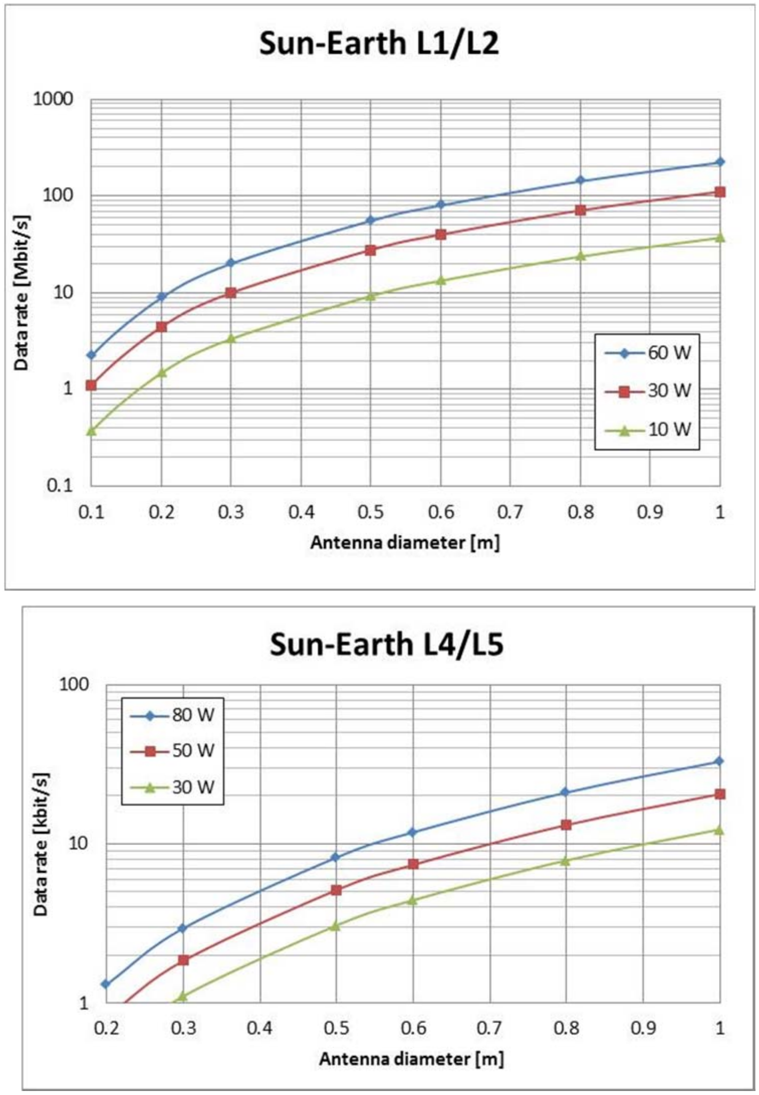


Figure 4: Evaluation of X band data rates as a function of transmitting power and antenna size from different orbits, scaled from the above mentioned L2 orbit case in X band: 10 Mbps achieved with 30 cm High Gain Antenna (HGA), 30 W output power, and 35 m ESTRACK ground antenna.

Appendix A. Performance of launch vehicles

A.1 LM-2C

The Long March 2C launcher (CZ-2C for Chang Zheng 2C in Chinese) is part of the Long March LV series. It is a two stage LV mainly used for LEO missions < 500 km. CTS is a third upper stage that can be added on the LM-2C, mainly for missions ≥ 500 km.

The main launch site for LM-2C is in JSLC (40.96°N – 100.29°E), but it can also be launched from XSLC (28.20°N - 102.02°E) and TSLC (38.50°N - 111.36°E).

The performance of LM-2C to circular LEO from JSLC is given in Figure 5.

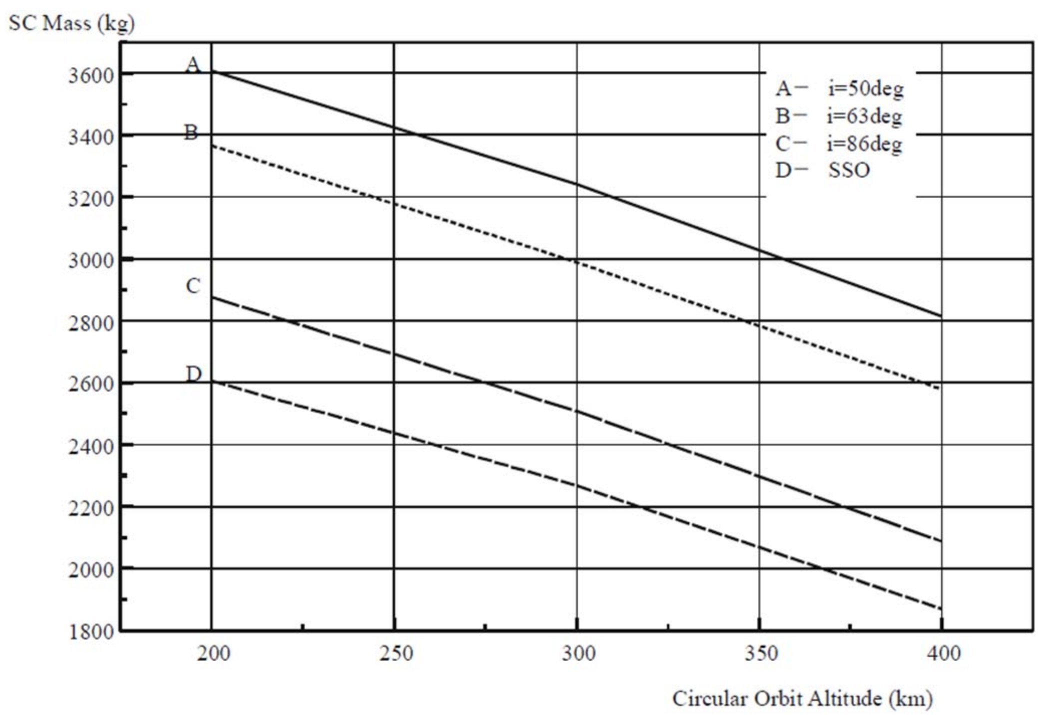


Figure 5: LM-2C performance to circular LEO from JSLC [1].

The performance of LM-2C/CTS to circular LEO is given in Figure 6 and Figure 7 for different launch sites. Note that:

- The curves labelled "de-orbit" refer to the possibility to de-orbit the CTS upper stage, in which case the LV performance is slightly degraded.
- "Hp" refers to the altitude of the perigee for elliptical orbits.

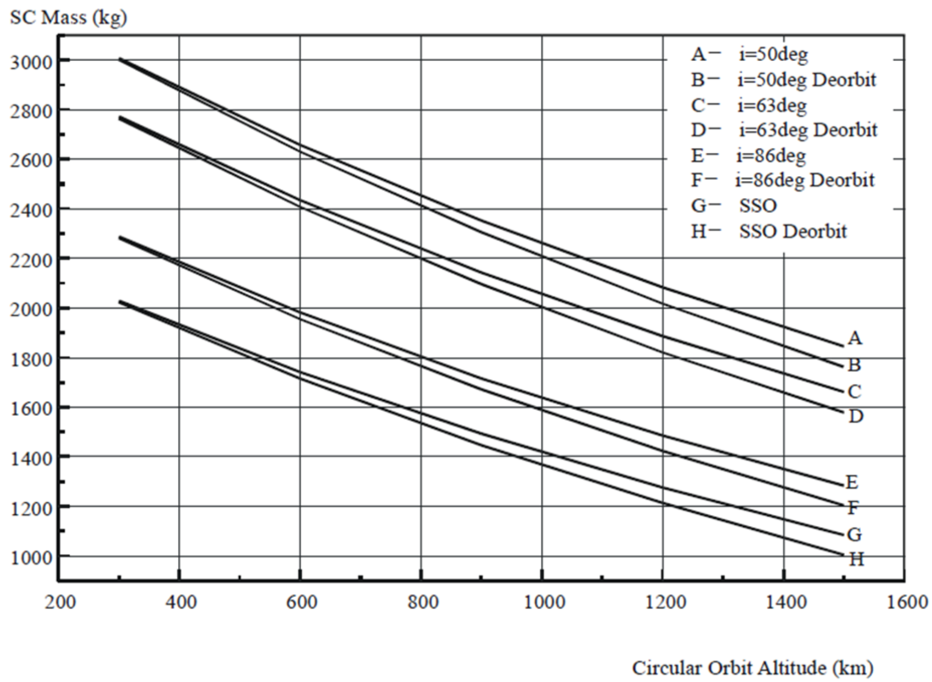


Figure 6: LM-2C/CTS performance to circular LEO from JSLC [1].

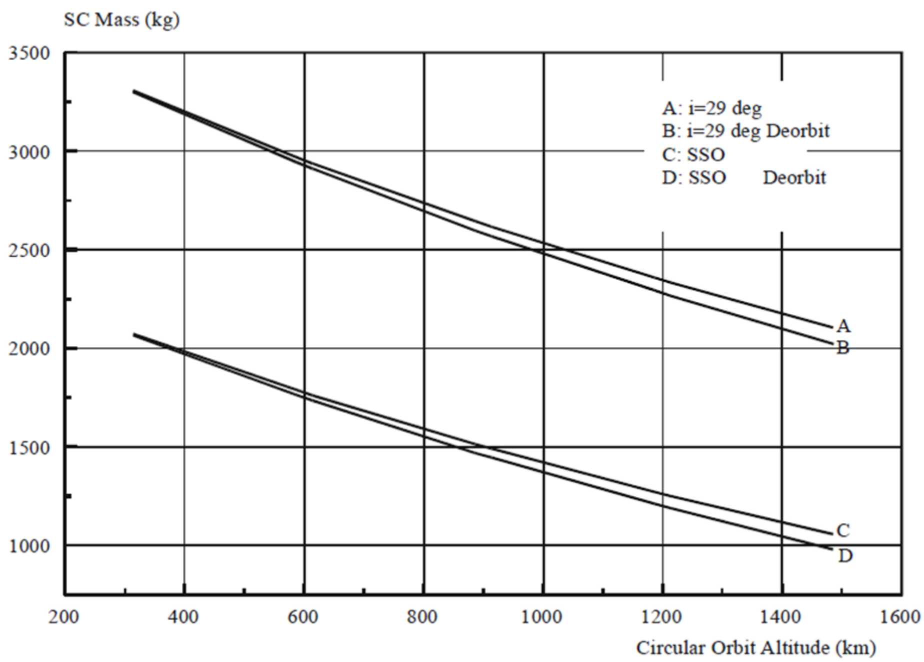


Figure 7: LM-2C/CTS performance to circular LEO from XSLC [1].

The performance of LM-2C to elliptical orbits is given in Figure 8 and Figure 9 for different launch sites.

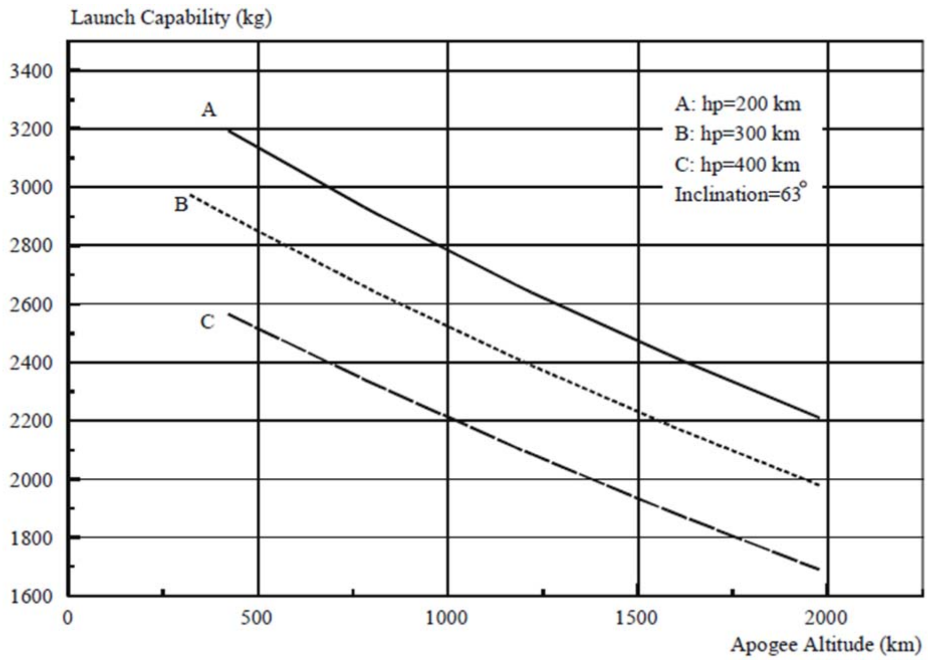


Figure 8: LM-2C performance to elliptical orbits from JSLC [1].

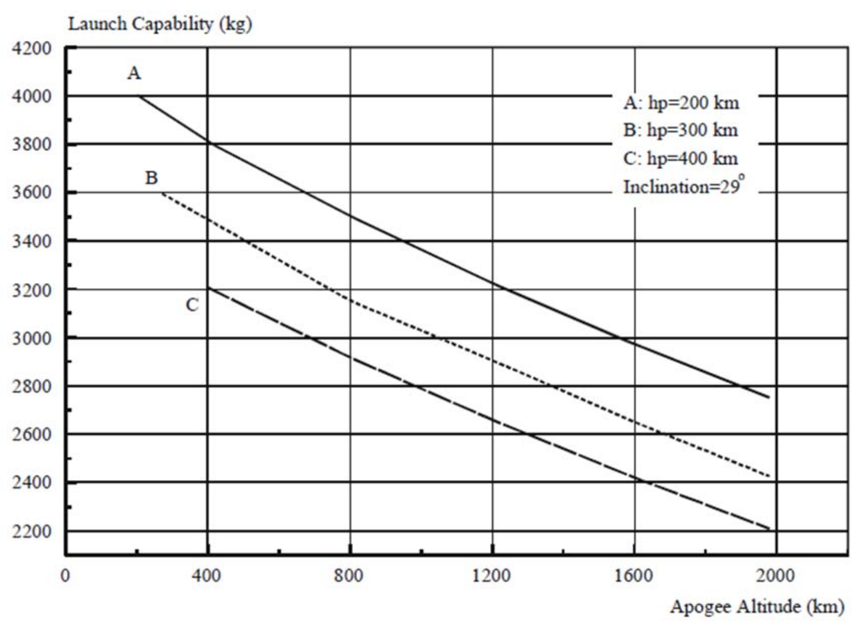


Figure 9: LM-2C performance to elliptical orbits from XSLC [1].

The performance of LM-2C/CTS to large elliptical orbits is given in Figure 10.

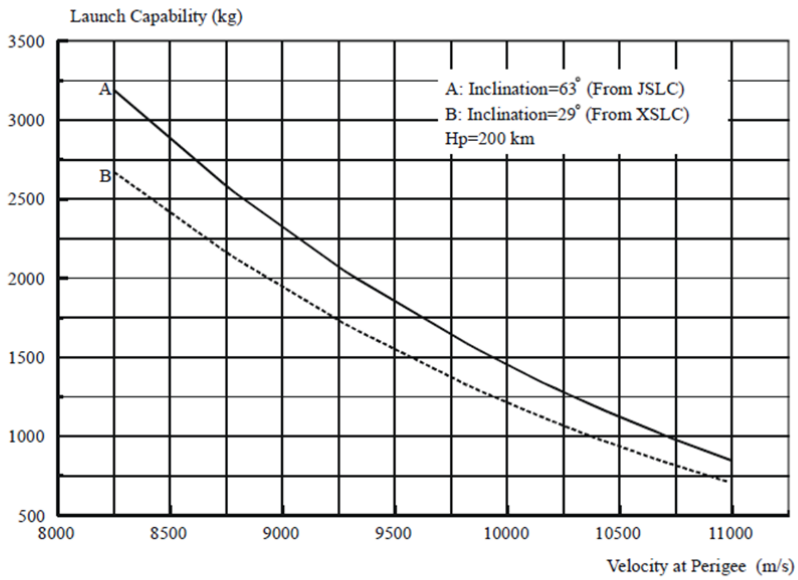


Figure 10: LM-2C/CTS performance to large elliptical orbits as a function of velocity at perigee [1].

The perigee velocities given in Figure 10 result in the apogees given in Figure 11 (with the perigee altitude of 200 km).

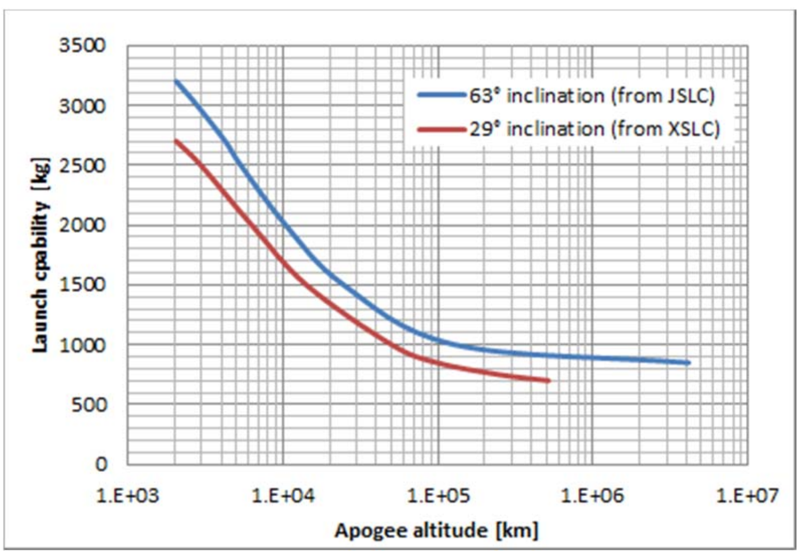


Figure 11: LM-2C performance to large elliptical orbits as a function of apogee altitude.

In this case, an initial LEO parking orbit is used first, following which an ignition of the upper stage injects the S/C into the desired orbit. This is suitable for e.g. GTO, L2 or even Earth escape orbits with the CTS upper stage. For Earth escape orbits, the launch capability of LM-2C/CTS is given in Table 4 as a function of the velocity at a 200 km perigee (and the C3 parameter) and for an inclination of 29°.

Velocity at perigee [m/s]	11044	11125	11210	11300	11396	11500	11608
C3 [km^2/s^2]	0.78	2.57	4.47	6.50	8.68	11.06	13.55
Launch capability [kg]	600	550	500	450	400	350	300

Table 4: LM-2C/CTS performance for Earth escape orbits as a function of velocity at a 200 km altitude perigee (and the C3 parameter) and for an inclination of 29°.

The data from Table 4 is illustrated in Figure 12.

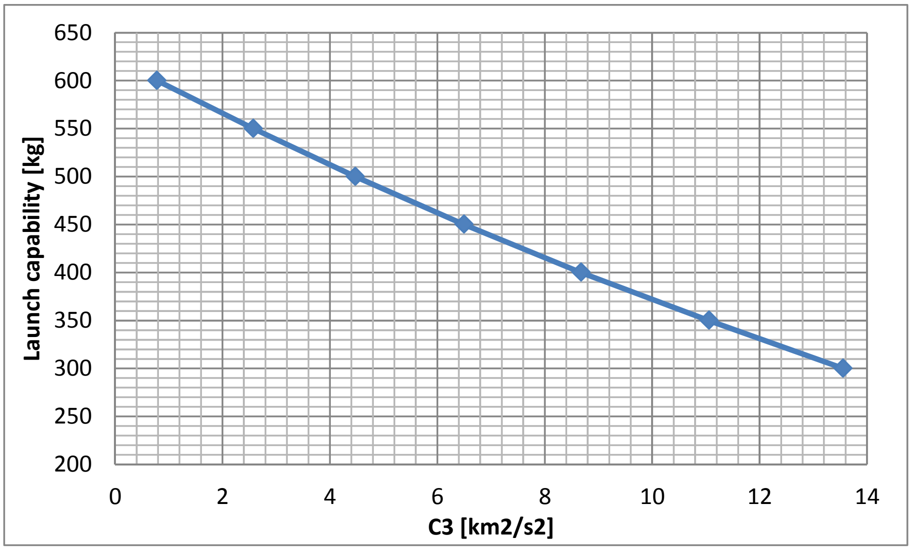


Figure 12: LM-2C/CTS performance for Earth escape orbits as a function of the C3 parameter, from a 200 km perigee altitude and for an inclination of 29°.

Note: At an altitude of 200 km, the Earth escape velocity, at which the orbit changes from elliptical (C3<0) to parabolic (C3=0) or even hyperbolic (C3>0), is 11.008 km/s.

A.2 LM-2D

LM-2D is another two stage LV for LEO and SSO missions. It can also incorporate an additional liquid upper stage TY-2 for an enhanced performance.

The performance of LM-2D to circular LEO is given in Figure 13.

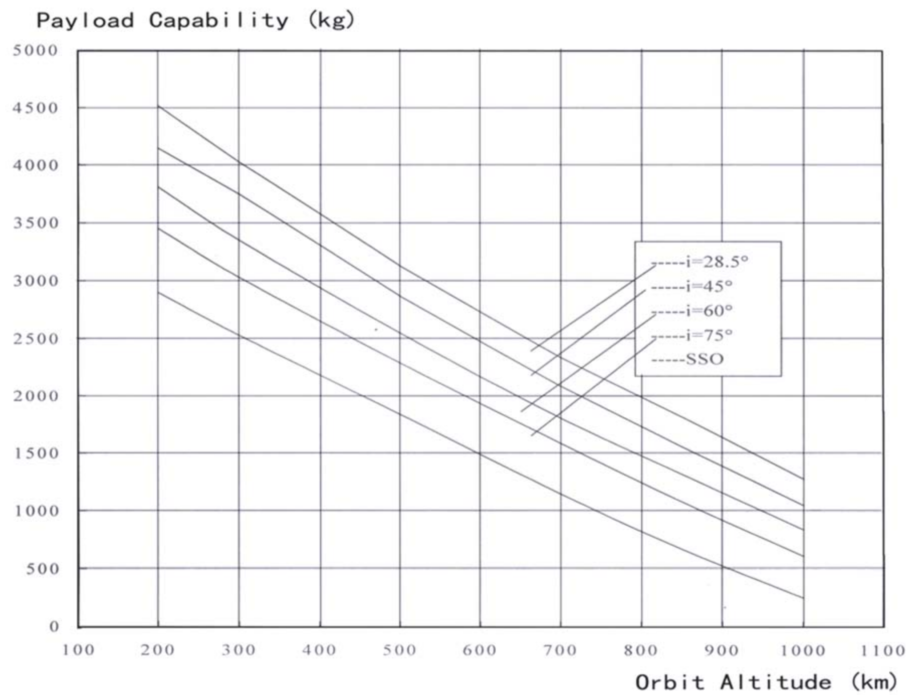


Figure 13: LM-2D performance to circular LEO.

The performance of LM-2D to elliptical orbits is given in Figure 14, Figure 15 and Figure 16 for different inclinations.

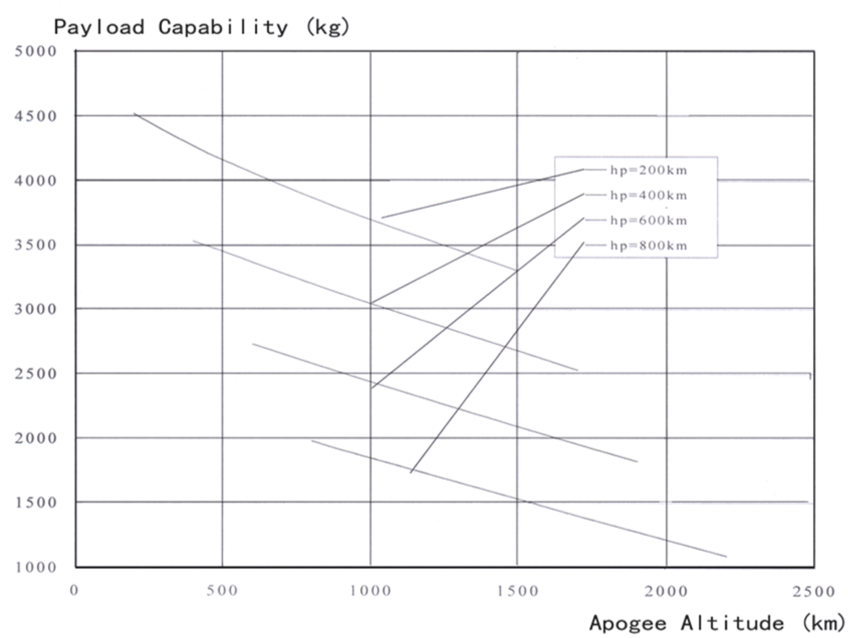


Figure 14: LM-2D performance to 28.5° inclination elliptical orbits.

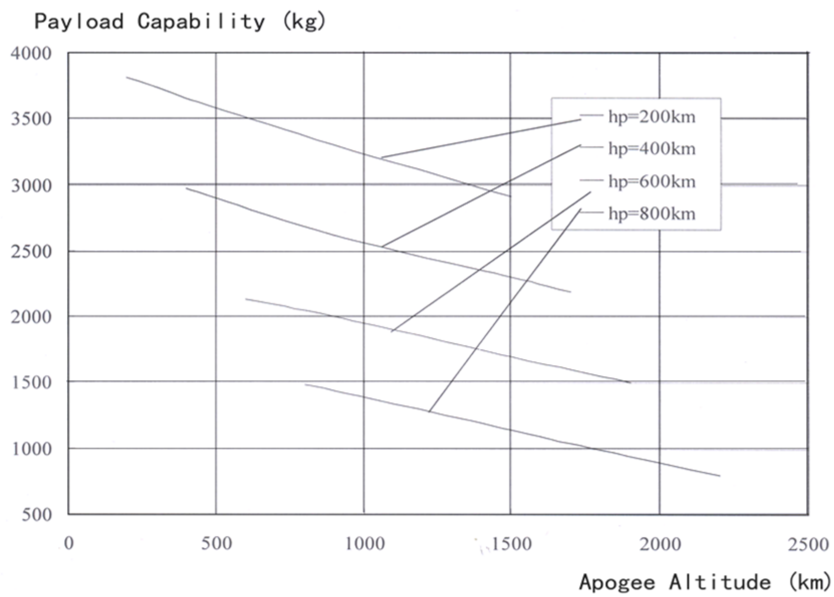


Figure 15: LM-2D performance to 60° inclination elliptical orbits.

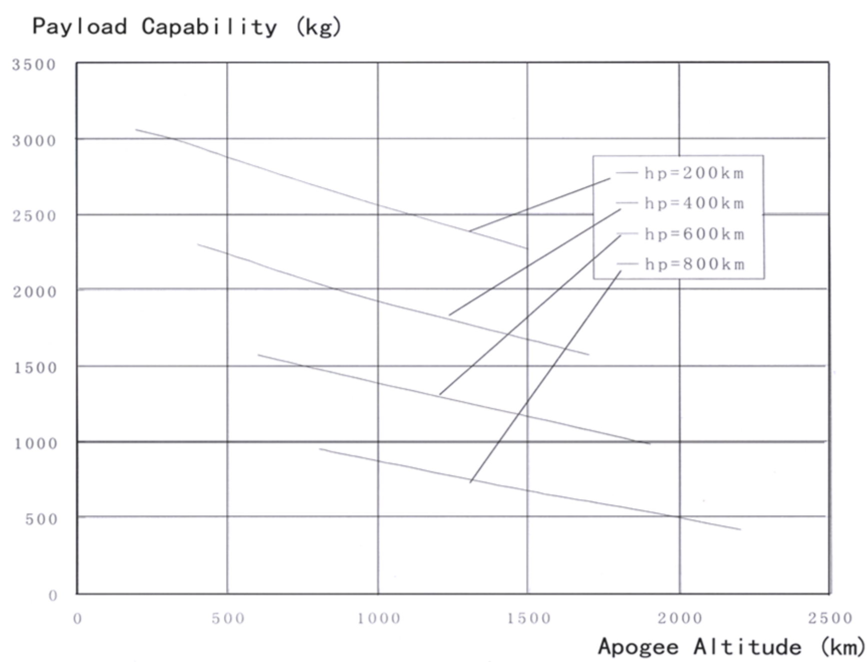


Figure 16: LM-2D performance to 90° inclination elliptical orbits .

The addition of the TY-2 upper stage also allows LM-2D to insert payloads into Earth escape orbits. This is illustrated in Figure 17.

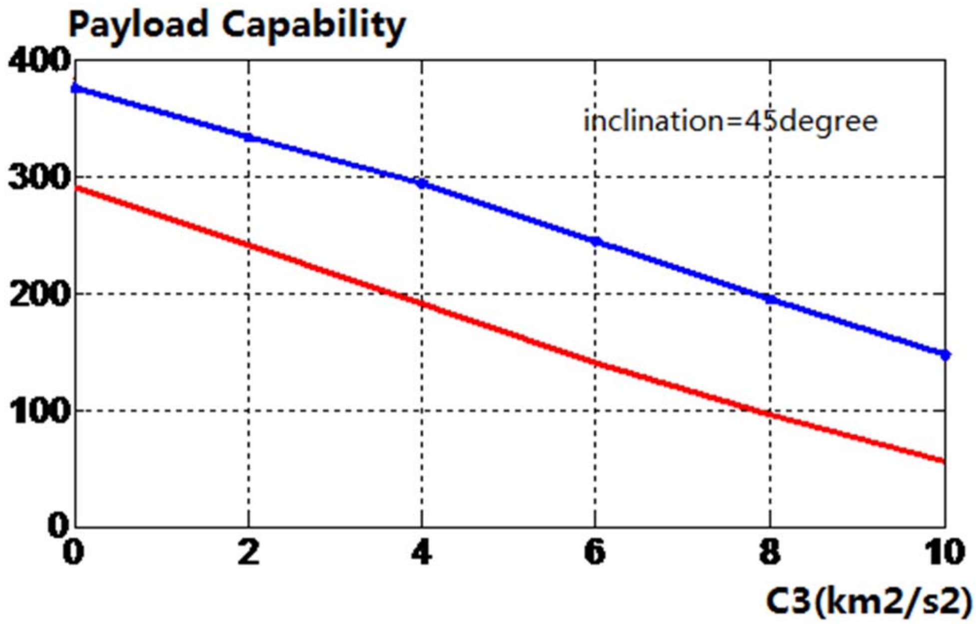


Figure 17: LM-2D/TY-2 performance for Earth escape orbits. The blue curve is escaping from a 200 x 900km parking orbit, and the red curve is escaping from a 200 km circular parking orbit.

A.3 Vega

Vega is a European launch vehicle that is launched from Kourou, French Guiana (Guiana Space Centre, 5.13°N – 52.45°W). It is part of the Arianespace LV family, with Soyuz and Ariane 5. The first flight took place in 2012. It is used for small missions to LEO, mainly Earth observation satellites in an SSO.

The performance of Vega to circular LEO is given in Figure 18. The user manual also gives the performance into the low-eccentricity, near equatorial orbit planned for the upcoming ESA mission LISA-Pathfinder (1963 kg at 200 x 1500 km, i = 5.4°).

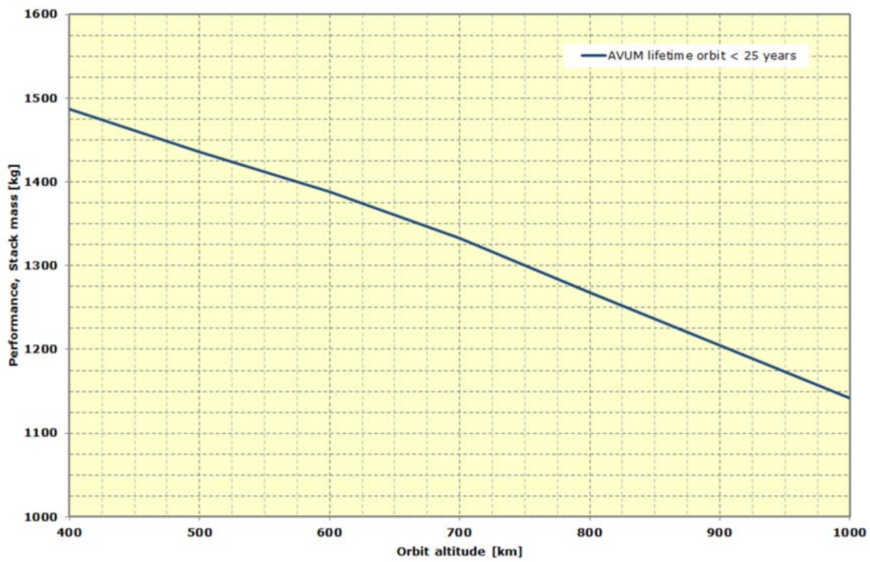


Figure 18: Vega performance to SSO [2].

A.4 Vega with a bi-liquid propulsion module

Based on the LISA-Pathfinder launch approach using a propulsion module, one can estimate the mass capability with a bi-liquid propulsion module and Vega (Figure 19). A generic low-inclination 300 km LEO is assumed for the Vega insertion, from which escape can be achieved in any direction through multiple burns for apogee raise, followed by a final burn for eventual insertion in an Earth escape hyperbola trajectory. The figure provides the spacecraft mass capability, represented as a function of the escape orbit C3 parameter. The yellow curve uses the Eurostar 2000 "short" tank (from LISA-Pathfinder), while the red curve shows a possible extension to a longer tank with a higher capacity (the tank exists, but a modification of the LPF propulsion module would be required). The "short" tank has a 1200 kg propellant capacity, which does not take advantage of the full Vega performance of ~2300 kg in the 300 km circular low-inclination orbit (which is why LPF is actually launched from a specific eccentric orbit). Since the longer tank takes full benefit of the Vega capability, the associated performance (red curve on the graph) is deemed to provide the correct order of magnitude for the Vega performance with a bi-liquid propulsion module, although the actual mission profile (and possibly the propulsion module) may differ following mission-specific optimisation. In particular, at near-zero escape velocity (C3 = 0), the escape mass is around 420 kg, which is in good accordance with the LPF case for which the mission profile was extensively optimised.

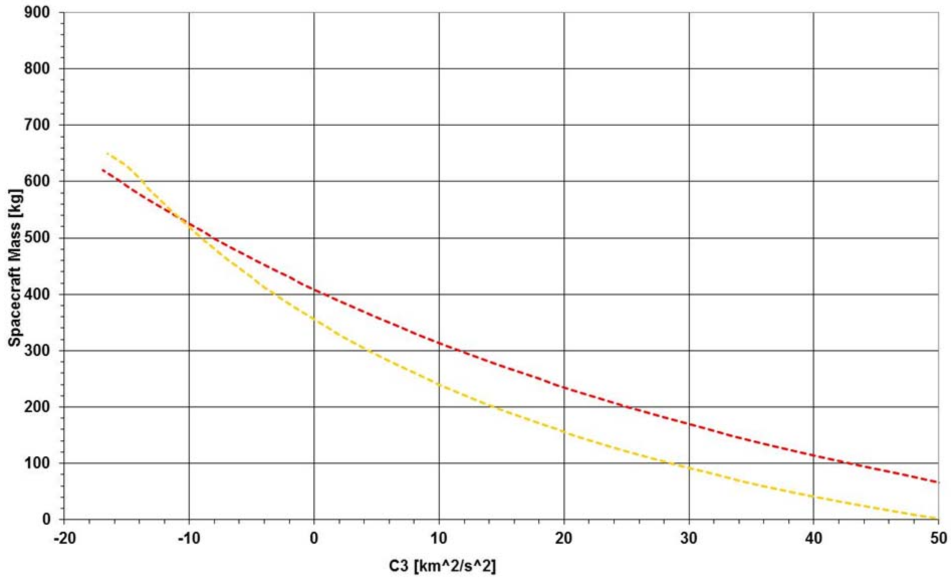


Figure 19: Approximate escape performance for a Vega launch with a bi-propellant stage, assuming launcher insertion in a 300 km LEO low-inclination orbit. The yellow line assumes a 2000 short tank (corresponding to the LISA-PF propulsion module). The red line assumes a 2000 long tank and takes full benefit of Vega's performance. The red line should be assumed as estimate of the Vega performance with a bi-liquid propulsion module, although the actual mission profile may eventually be different.

Specific C3 values for specific Earth-bound destinations are: C3 ~-16 km^2/s^2 for GEO orbit; and C3 ~0 km^2/s^2 for L1/L2. For Earth escape missions, more details on the C3 are given in Appendix F.

The L4/L5 Sun-Earth Lagrange points can be reached using the approach detailed above. The propulsion module is jettisoned after escape, and a final insertion manoeuvre is needed at arrival at L4/L5, which is assumed to be achieved by the spacecraft's on-board propulsion system (e.g. possibly with the spacecraft control thrusters).

The Sun-Earth L5 point is found to be less demanding to reach than the L4 point (L5 requires the period of the orbital transfer to be above 1 year, while L4 requires a less costly orbital transfer period, shorter than 1 year) and offers the added advantage of allowing observations of the situation on the solar surface before the observed regions will have rotated onwards so they can affect the Earth.

The fuel demands for reaching L4/L5 can be lowered by increasing the transfer time, as illustrated in Table 5. Transfers are possible in discrete intervals, the shortest of which is 14 months. The next one is 26 months and offers significant benefits both in terms of escape C3 and the ΔV applied at arrival. Longer transfers lead to further, though not significant savings.

Transfer duration [months]	Escape from 300 km LEO [km/s]	Departure C3 [km²/s²]	Estimated spacecraft mass into heliocentric orbit incl. prop system for final insertion [kg]	Arrival manoeuvre [km/s]	Prop. fraction for arrival manoeuvre [%]
14	3.292	2.016	~ 230	1.419	37
26	3.227	0.582	~ 310	0.763	22
38	3.213	0.272	~ 335	0.521	16
50	3.207	0.157	~ 350	0.396	12

Table 5: Approximate Sun-Earth L5 transfers. The last column indicates the propellant needed to execute the arrival manoeuvre assuming an Isp of 317 s, expressed as a fraction of the S/C wet mass.

For drifting, Earth leading/trailing orbits, there are no constraints such as discrete transfer intervals and no arrival manoeuvre is required. The only ΔV to consider is the one required to reach Earth escape velocity, with a $C3 \geq 0$ km²/s².

A.5 Soyuz

The performance of Soyuz from GSC (Kourou) is illustrated from Figure 20 to Figure 23:

- Figure 20: LEO circular Sun Synchronous Orbits (SSO)
- Figure 21 and Figure 22: Elliptical and High Elliptical Orbits
- Figure 23: Earth escape missions

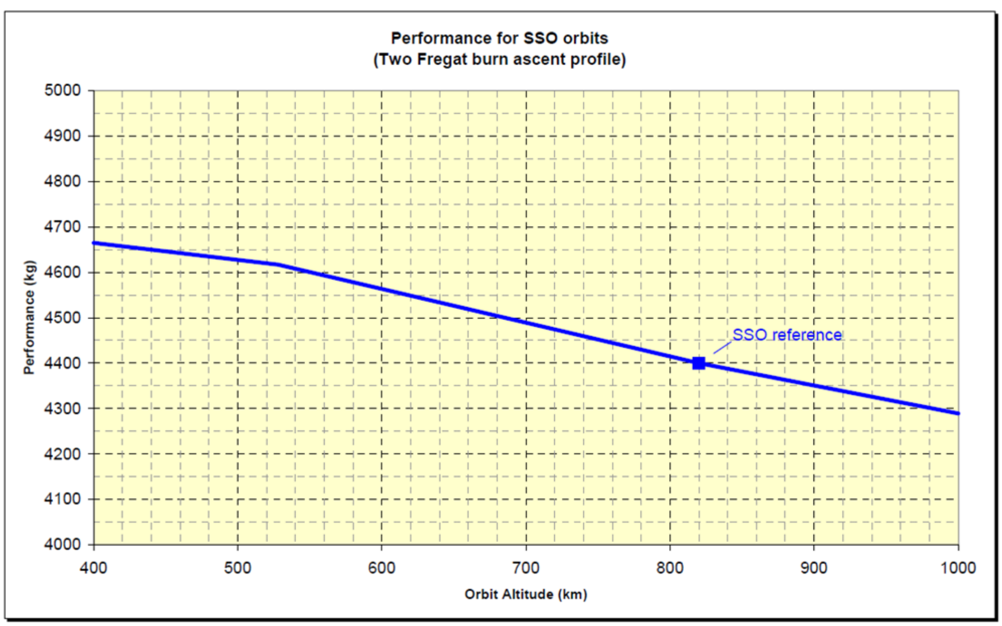


Figure 20: Soyuz performance to circular SSO [3].

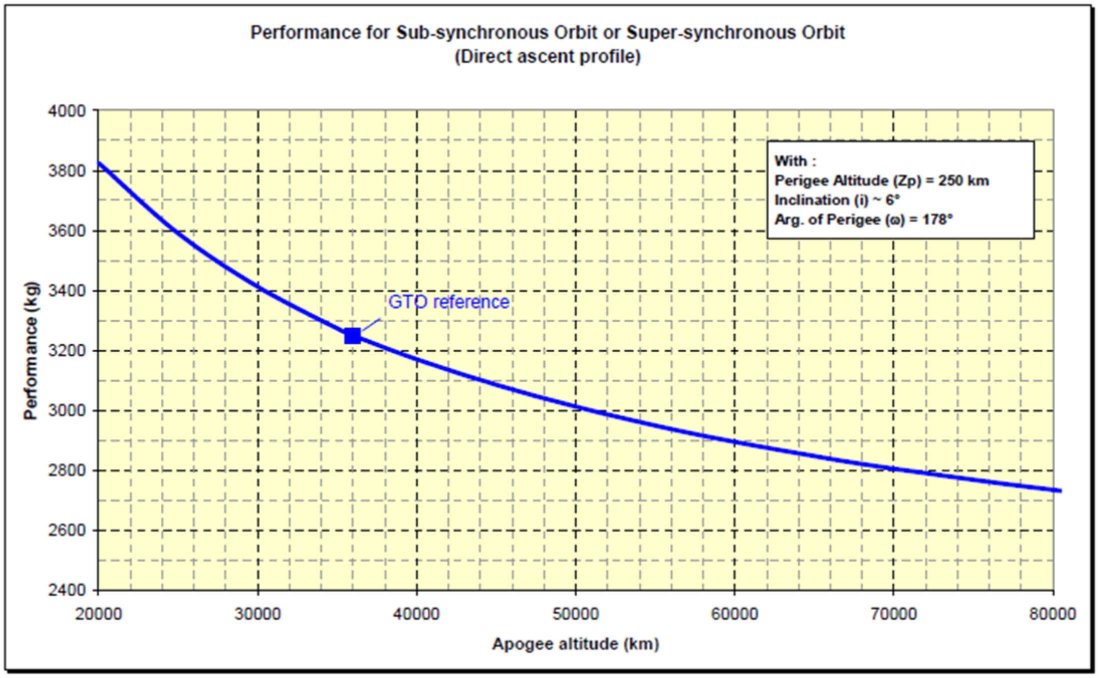


Figure 21: Soyuz performance to elliptical orbits including GTO [3].

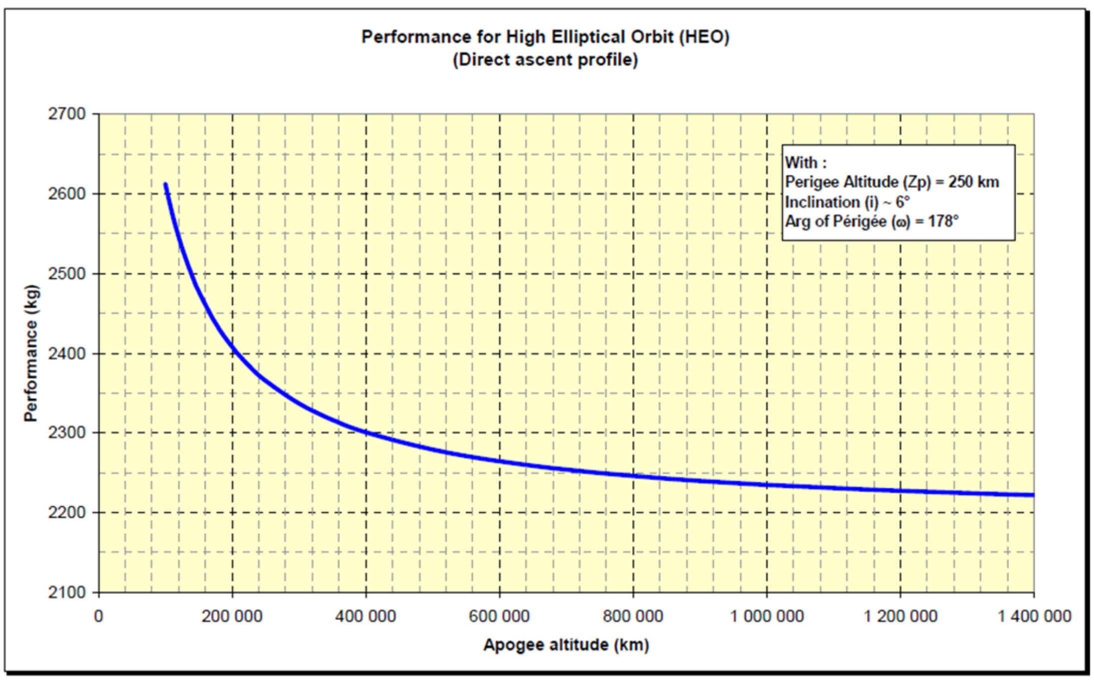


Figure 22: Soyuz performance to high elliptical orbits [3].

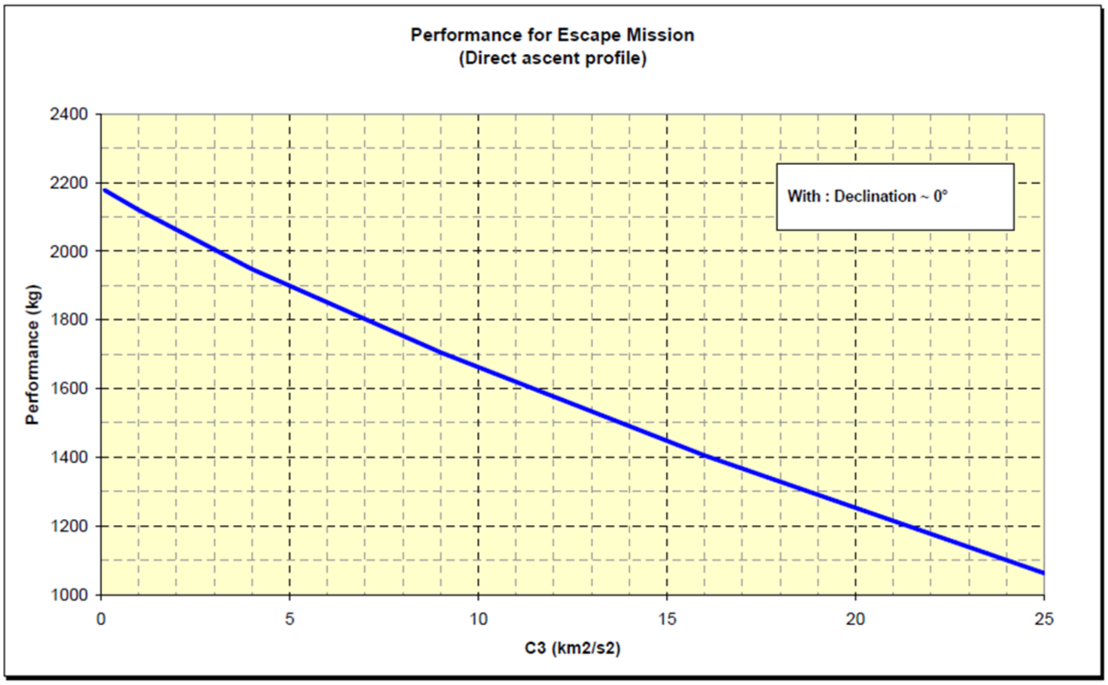
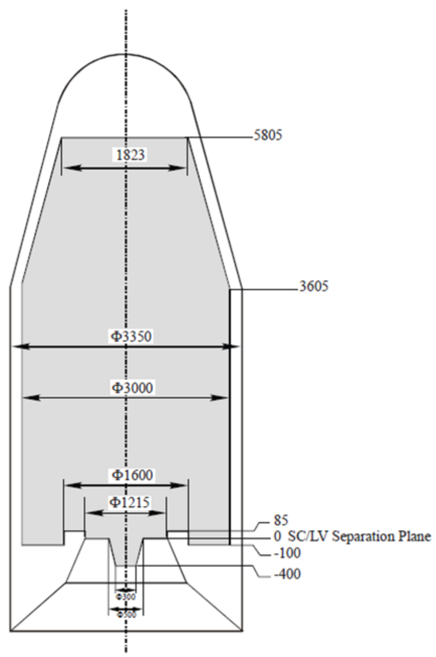
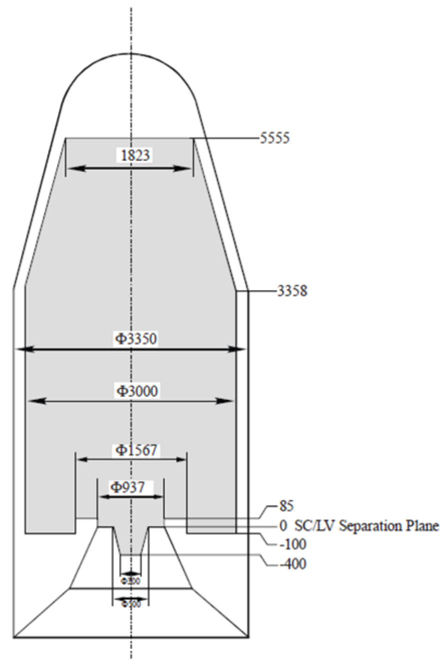


Figure 23: Soyuz performance for Earth escape missions [3].

Appendix B. Launch vehicle fairings and adapters

B.1 LM-2C





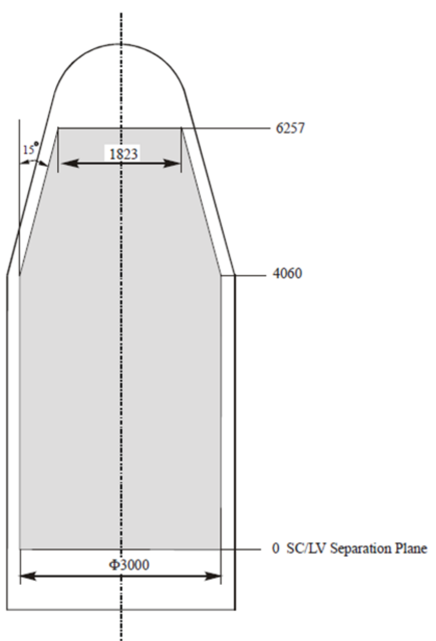


Figure 24: (up left) Fairing static envelope of LM-2C with 1194A interface, (up right) LM-2C with 937B interface and (bottom) LM-2C/CTS with explosive bolt interface [1].

B.2 LM-2D

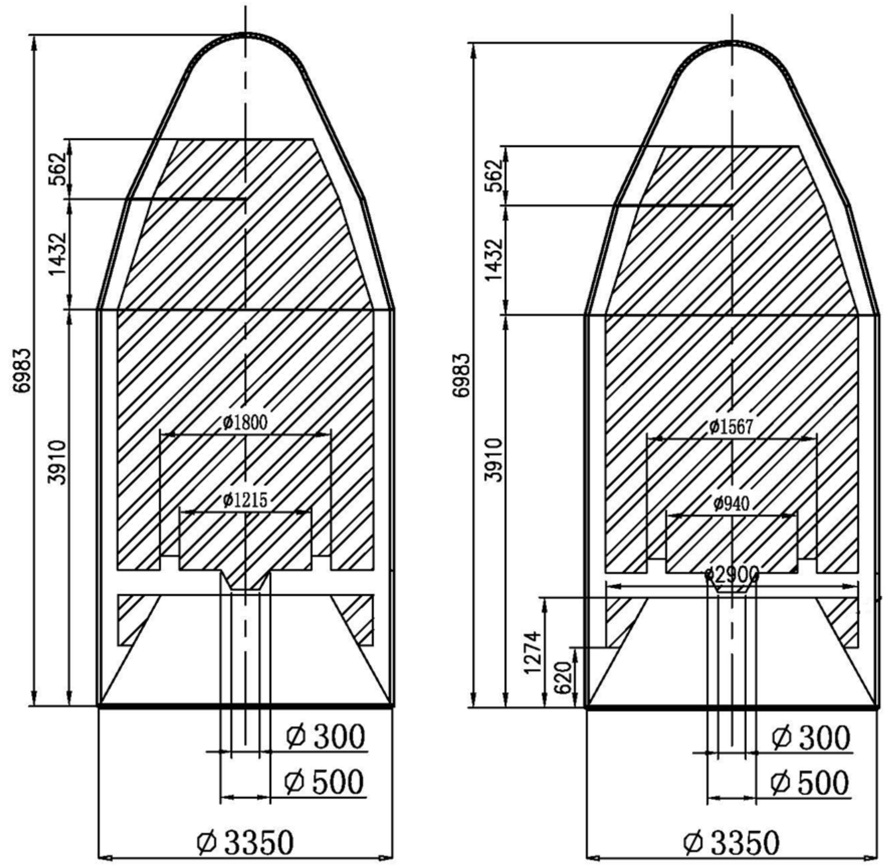


Figure 25: LM-2D fairing static envelope with 1194 (left) and 937 (right) interfaces.

Four standard mechanical interfaces are available with LM-2D: 660, 937, 1194 and 1194B.

B.3 Vega

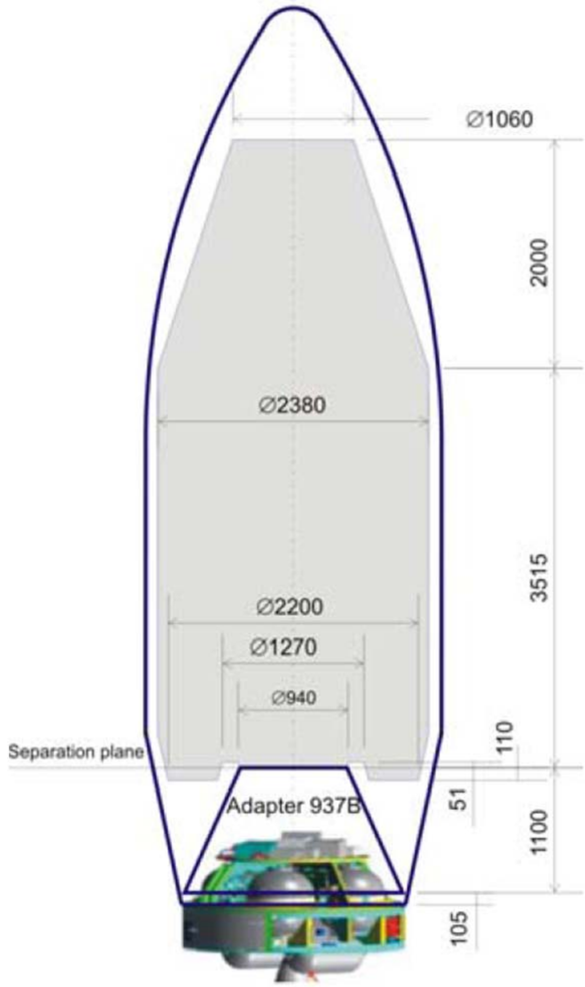


Figure 26: Fairing dimensions of Vega [2].

The Vega specific VESPA adapter [2] is available for dual missions. The upper position allows passengers up to 1000 kg, while a 600 kg S/C can be accommodated inside the VESPA cavity.

B.4 Soyuz

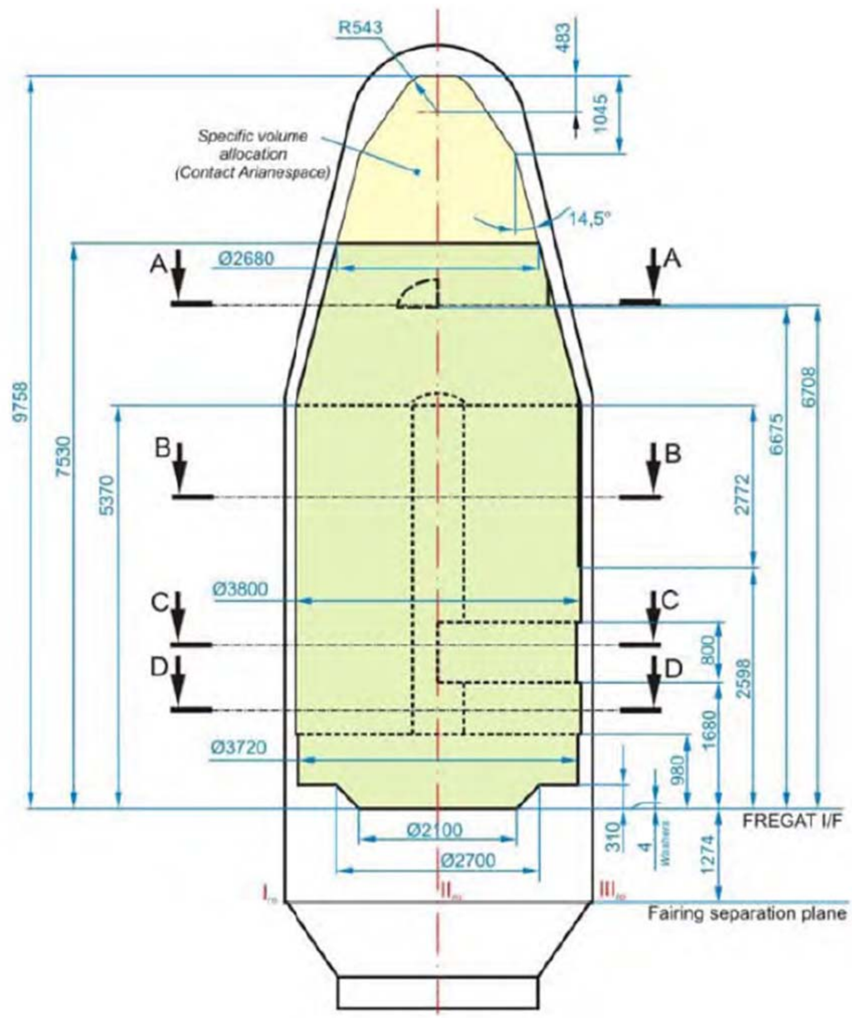


Figure 27: ST fairing of Soyuz [3].

Soyuz proposes standard adapters for multiple S/Cs. Those include adapters designed for the Globalstar 2 and Galileo missions, but also the SYLDA-S (under qualification) and the ASAP-S (designed for 1 main S/C of ~400 kg in the central position, and 4 external satellites in the 200 kg class).

Appendix C. Space debris mitigation

This appendix builds on the space debris mitigation requirements detailed in section 2.4.

The lifetime of missions in LEO is given in Figure 28, as a function of the S/C mass to area ratio (i.e. the ballistic coefficient).

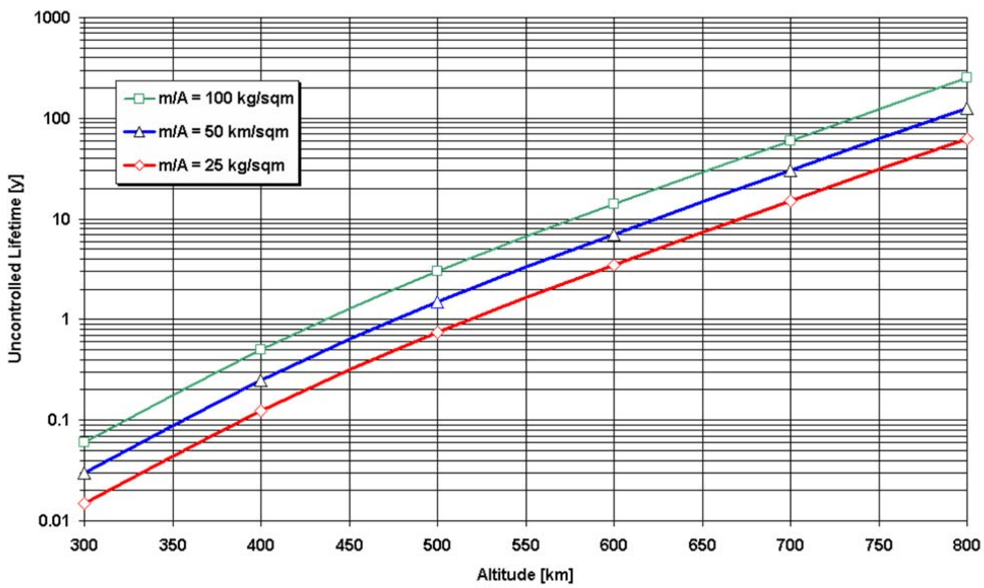


Figure 28: Lifetime in years for circular LEO orbits, with a medium Solar activity, as a function of the mass to area ratio of the S/C.

For the small-class mission under consideration in this paper, a worst case mass to area ratio of ~300 kg for 1 m² is possible, meaning the LEO lifetime could be as high as 3 times the green curve in Figure 28. With these characteristics, such a mission would re-enter within 25 years as long as it is in a circular orbit below ~550 km.

Based on this, several strategies can be adopted for orbital debris mitigation:

- Lower the altitude below 550 km, from which an un-controlled re-entry will follow within 25 years. This is achieved with a first manoeuvre to reduce the perigee, followed by a second manoeuvre to circularise the orbit.
- De-orbit to an eccentric orbit with a 25 years lifetime. Unlike the first option above, the second manoeuvre to circularise the orbit is not necessary: keeping the S/C in an eccentric orbit should ensure de-orbiting, as long as the perigee is low enough (lower than the 550 km for the circular orbit in the first option, see Figure 30).
- Raise the altitude above 2000 km, outside of the LEO protected region. This is beneficial in terms of ΔV only if the initial altitude is already high enough.

For these 3 options, the required ΔVs are shown in Figure 29. One can see that the eccentric orbit solution is more advantageous for orbits with altitudes below 1400 km (and re-entry will occur in less than 25 years if the mass to area ratio of the S/C is equal to or lower than 250 kg for 0.8 mm²), while raising the altitude above 2000 km is more advantageous for altitudes above ~1400 km. When retaining the most favourable case, the ΔV ranges from 20 m/s to ~260 m/s.

For information, the perigees of the eccentric orbits required to de-orbit within 25 years (corresponding to the blue curve in Figure 29) are given in Figure 30, for an average Solar activity (as the time scale spans over 2 Solar cycles).

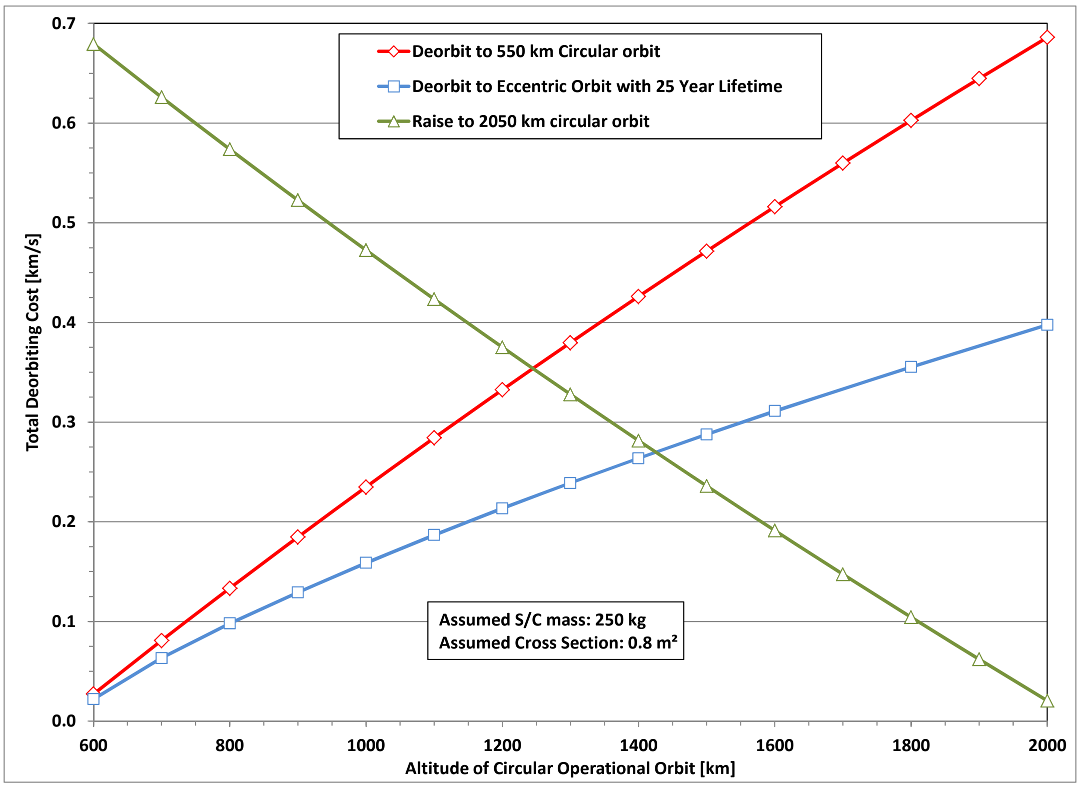


Figure 29: ΔV required to reduce the altitude of a circular orbit down to 550 km (red curve), to raise it to 2050 km (green curve), or to go to an eccentric orbit with a low enough perigee to ensure de-orbiting within 25 years (blue curve).

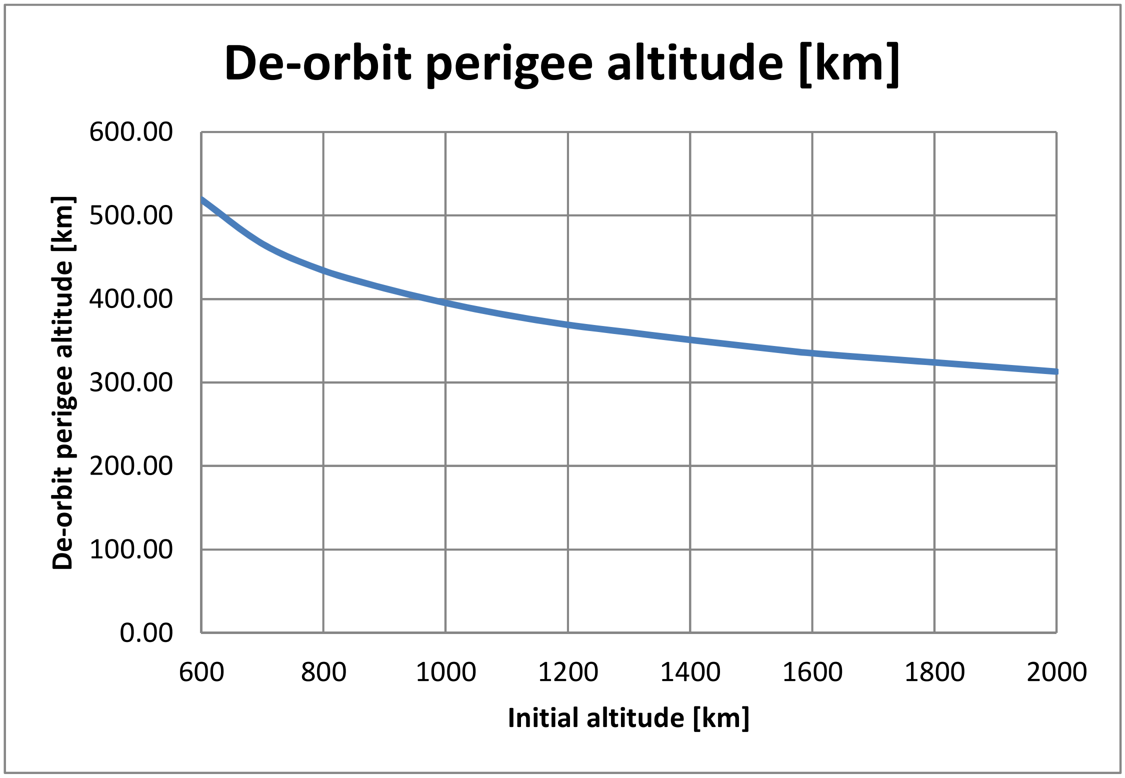


Figure 30: Perigee of eccentric orbit required to de-orbit within 25 years, with aS/C mass to area ratio of 250 kg for 0.8 m².

Appendix D. Examples of small missions

Mission	Launch	Operational orbit	Launch mass [PL mass]	Total power	Propulsion	Downlink	Pointing
Smart-1	2003 Shared Ariane 5 with ASAP	Polar elliptical Moon orbit (transfer from GTO)	367 kg [19 kg]	1765 W cruise mode 225 W science mode	3.9 km/s 82 kg Xe Solar Electric Propulsion	65 kbit/s S band + X/Ka band demonstration	APE = 15’
CHEOPS	2017 Shared launch (compatible with passenger to Soyuz, Vega, and other launchers)	LEO SSO, dusk-dawn (650-800 km)	280 kg [60 kg]	200 W nominal 60 W allocated to the instrument	17 kg Mono-propellant	1.2 Gbit/day S band	APE 4’’ rms (telescope used as a Fine Guidance Sensor)
Double Star	2003/2004 2x LM-2C launches	1 equatorial: (570x78970 km, 28.5°) 1 polar: (700x39000 km, 90°)	330 kg [80 kg] x2 (each)	260 W (BoL)	None for orbit control	Equatorial: 1.4 GBit/day Polar: 2.66 Gbit/day S band	Spinner S/C: 15 rpm

Table 6: Examples of small science missions.

Appendix E. ISO TRL table

Technology Readiness Level	Milestone achieved for the element	Work achievement (documented)
TRL 1: Basic principles observed and reported	Potential applications are identified following basic observations but element concept not yet formulated.	Expression of the basic principles intended for use. Identification of potential applications.
TRL 2: Technology concept and/or application formulated	Formulation of potential applications and preliminary element concept. No proof of concept yet.	Formulation of potential applications. Preliminary conceptual design of the element, providing understanding of how the basic principles would be used.
TRL 3: Analytical and experimental critical function and/or characteristic proof-of-concept	Element concept is elaborated and expected performance is demonstrated through analytical models supported by experimental data/characteristics.	Preliminary performance requirements (can target several missions) including definition of functional performance requirements. Conceptual design of the element. Experimental data inputs, laboratory-based experiment definition and results. Element analytical models for the proof-of-concept.
TRL 4: Component and/or breadboard functional verification in laboratory environment	Element functional performance is demonstrated by breadboard testing in laboratory environment.	Preliminary performance requirements (can target several missions) with definition of functional performance requirements. Conceptual design of the element. Functional performance test plan. Breadboard definition for the functional performance verification. Breadboard test reports.
TRL 5: Component and/or breadboard critical function verification in a relevant environment	Critical functions of the element are identified and the associated relevant environment is defined. Breadboards not full-scale are built for verifying the performance through testing in the relevant environment, subject to scaling effects.	Preliminary definition of performance requirements and of the relevant environment. Identification and analysis of the element critical functions. Preliminary design of the element, supported by appropriate models for the critical functions verification. Critical function test plan. Analysis of scaling effects. Breadboard definition for the critical function verification. Breadboard test reports.
TRL 6: Model demonstrating the critical functions of the element in a relevant environment	Critical functions of the element are verified, performance is demonstrated in the relevant environment and representative model(s) in form, fit and function.	Definition of performance requirements and of the relevant environment. Identification and analysis of the element critical functions. Design of the element, supported by appropriate models for the critical functions verification. Critical function test plan. Model definition for the critical function verifications. Model test reports.
TRL 7: Model	Performance is demonstrated for the	Definition of performance requirements, including

Technology Readiness Level	Milestone achieved for the element	Work achievement (documented)
demonstrating the element performance for the operational environment	operational environment, on the ground or if necessary in space. A representative model, fully reflecting all aspects of the flight model design, is built and tested with adequate margins for demonstrating the performance in the operational environment.	definition of the operational environment. Model definition and realization. Model test plan. Model test results.
TRL 8: Actual system completed and accepted for flight ("flight qualified")	Flight model is qualified and integrated in the final system ready for flight.	Flight model is built and integrated into the final system. Flight acceptance of the final system.
TRL 9: Actual system "flight proven" through successful mission operations	Technology is mature. The element is successfully in service for the assigned mission in the actual operational environment.	Commissioning in early operation phase. In-orbit operation report.

Table 7: Summary definition of the ISO TRL levels (Courtesy from ISO. For further details, the reader is invited to refer to the ISO document 16290 [5]).

Appendix F. C3 definition

In the two-body Newtonian gravitation approximation, the orbital velocity is defined as:

$$V = \sqrt{\mu.\left(\frac{2}{r} - \frac{1}{a}\right)}$$

where:

- V is the orbital velocity
- r is the distance from the centre of the celestial body to the S/C
- $\mu/r$is the gravitation potential
- a is the semi-major axis of the orbit (assumed to be a conic, with the convention a < 0 for the hyperbolic case)

The orbit parameter C3 is defined as:

$$C_3 = -\frac{\mu}{a} = V^2 - \frac{2.\mu}{r}$$

C3/2 is the specific energy of the orbit: therefore, C3<0 for elliptical orbits, C3 = 0 for the parabolic orbits and C3>0 for hyperbolic orbits.

For hyperbolic orbits, we also have $C_3 = V_\infty{}^2$, where $V_\infty = \lim_{r\to\infty} V$ is the velocity at infinity ($V_\infty = 0$ for the parabolic limit). Therefore, when applying the above formulas to the two-body system defined by the Earth and the spacecraft, C3 provides the escape velocity in the Earth referential frame. For obtaining the spacecraft velocity in the heliocentric referential frame, the Earth orbital velocity must be added to V_∞. When considering a direct interplanetary transfer based on the well-known Hohmann elliptic transfer from Earth orbit to some other planet of our solar system, V_∞ can be viewed as the velocity change ΔV_1 for leaving the Earth orbit to the targeted planet, and the insertion in the targeted planet orbit requires a second velocity change ΔV_2 to be provided at the planet arrival.

With the above formulas, one can calculate the order of magnitude of the C3 parameter for direct interplanetary Hohmann transfer, by neglecting the orbit inclinations and within the two-body approximation. The result is illustrated in Figure 31 and Figure 32. Exact C3 calculation must take into account the orbit inclinations and the actual arrival date.

Note that for Mercury, Jupiter and beyond, typical transfers will involve gravity assists manoeuvres (e.g. JUICE and BepiColombo missions), to reduce the ΔV budget for the space segment.

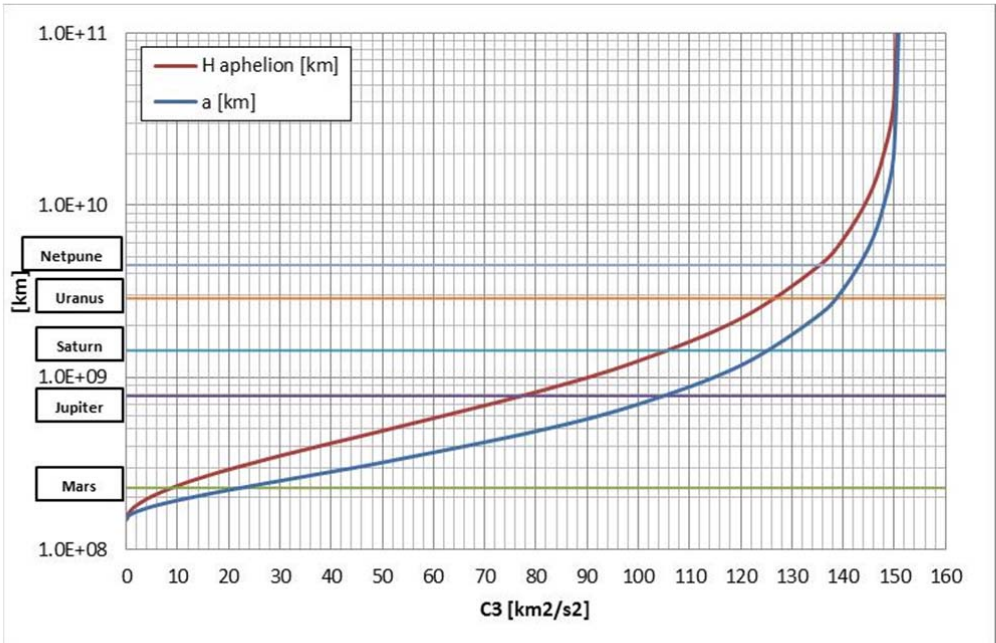


Figure 31: C3 values required to reach the external planets, assuming a direct Hohmann transfer. The semi-major axes of the orbit of the external planets are indicated. H and a are respectively the aphelion and semi-major axis of the transfer ellipse.

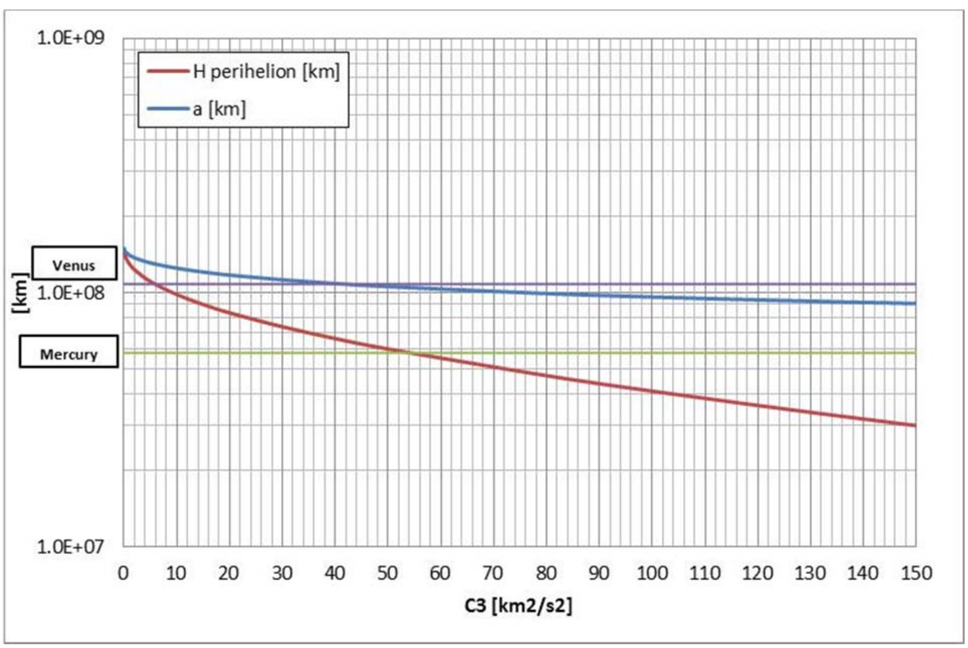


Figure 32: Same as Figure 12 for the inner planets, with H being now the perihelion of the transfer orbit.