DOCUMENT

Cosmic Vision M4 Call
Technical Annex

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1 INTRODUCTION

1.1 Scope of document

This Annex provides technical inputs for the preparation of the proposals in answer to the Cosmic Vision M4 Call. Its main objective is to provide technical information to enable proposers to define their mission concept to the level required to enable the evaluation of the mission's technical feasibility, as required by the Call for Missions.

Reference information from previous ESA missions, relevant to the preparation of the proposals as technical heritage, can be found at: http://sci.esa.int/home/51459-missions/.

1.2 Reference documents

[1] Soyuz User's Manual, issue 2.0, 2012

[2] Vega User's Manual, issue 4.0, 2014

[3] Venus probe CDF report, CDF-106(A), 2010

[4] INSPIRE CDF study report, CDF-124(A).2011

[5] Requirements on space debris mitigation for ESA projects, IPOL(2008)2 Annex 1

[6] ISO/CD 16290, Space system – Definition of the TRL and their criteria of assessment 2012

1.3 List of acronyms

APE	Absolute Performance Error
ASAP	Arianespace System for Auxiliary Payload
AU	Astronomical Unit
AVUM	Attitude and Vernier Upper Module
CaC	Cost at Completion
CDMS	Command and Data Management Subsystem
CHEOPS	Characterizing Exoplanet Satellite
CPM/S	Chemical Propulsion Module/Stage
DHS	Data Handling System
DV	Delta V (also ΔV)
EChO	Exoplanet Characterisation Observatory
EoL	End of Life
ESA	European Space Agency
FGS	Fine Guidance Sensor
GL	Gravity Loss

GEO	Geostationary Earth Orbit
GTO	GEO Transfer Orbit
HEO	High Elliptical Orbit
HGA	High Gain Antenna
ISO	International Organisation for Standardisation
LEO	Low Earth Orbit
LISA	Laser Interferometry Space Antenna
LOFT	Large Observatory For X-ray Timing
LPF	Lisa PathFinder
MEX	Mars Express
MOC	Mission Operations Centre
MOI	Mars Orbit Insertion
PAS	Payload Adapter System
PL	Payload
PLATO	Planetary Transits and Oscillations of stars
PLM	Payload Module
RF	Radio Frequency
RPE	Relative Performance Error
S/C	Spacecraft
SEP	Solar Electric Propulsion
SM	Solid Motor
SOC	Science Operations Centre
SolO	Solar Orbiter
SSO	Sun Synchronous Orbit
SVM	Service module
SYLDA	Systeme de Lancement Double Ariane
TBC	To Be Confirmed
TM	Telemetry
TRL	Technology Readiness Level
VESPA	Vega Secondary Payload Adapter
VEX	Venus Express
VOI	Venus Orbit Insertion

2 GENERAL GUIDELINES

The M4 Call is targeting a Cost at Completion (CaC) to ESA of 450 M€ (e.c. 2014) and a relatively fast development schedule (~10-11 years from the Call to launch). The purpose of this technical annex is to support the proposers in defining mission profiles that are compatible with the above programmatic boundaries.

Table 1 provides a set of high-level constraints, the range within which an ESA-only M4 mission should be affordable within the 450 M€ CaC ceiling. Without knowing the mission's specificities it is of course not possible to provide a priori detailed technical constraints that guarantee the achievement of the cost targets. For example, while there is a correlation between the spacecraft's mass and its cost, a lightweight spacecraft with high complexity may result in a cost and development schedule well above the targeted boundaries, while a heavy spacecraft with low complexity may be implement at a low cost. Therefore, the recommended constraints provided here must not be interpreted as a set of rigid requirements but as general guidelines. This said, experience shows that space science missions generally exhibit a roughly comparable level of complexity, so that the guidelines below should be viewed as recommendations helping the proposers to achieve the M4 cost and schedule targets for their proposal. Again, exceptions exist, so that proposers are of course free to depart from one or the other recommendation below, in which case however they are advised to justify their proposed mission profile in more detail.

Table 1: Parameter envelope suggested for the M4 mission (see main text for details).

Element	Recommended values	Comment
Spacecraft dry mass (including payload and propulsion system(s))	< ~ 800 kg	Upper limit, excluding the launcher adapter. Applies to both Vega and Soyuz launchers. A lower mass figure may be needed for fitting launcher capability (see below).
Payload mass	< ~ 300 kg	Also to be interpreted as an upper limit. For planetary missions, it is recommended to limit the science instrumentation mass to 80 kg.
Technology Readiness	TRL > 5-6 (ISO scale) For all the spacecraft elements (including the payload).	See Appendix A for TRL definition. The payload can be a new development but must rely on available technologies for all the instrument elements. Some limited delta-developments or verifications can be envisaged prior to the mission adoption (must be achievable in 2-2.5 years)
In-orbit operations	< 3-3.5 years	Nominal lifetime, excluding possible extensions.
Launcher	Vega or Soyuz	See Section 0 for possible mission profiles.

- **Schedule and technology readiness.**

The target launch for the M4 mission is 2025, and the mission preparation phase is planned to be completed in ~3.5 years, i.e. in a shorter time than for previous M missions. As a consequence the spacecraft development should be achievable in 6-6.5 years. As for all M missions, the schedule requires reliance on available technologies (TRL > 5-6). No basic development should be envisaged for the payload (including the detection subsystem) and the spacecraft bus. Nevertheless, the payload can include the development of new instruments, provided they are relying on available technologies. Some specific delta-developments or verifications can still be envisaged prior to the mission adoption if they can be achieved in ~ 2-2.5 years.

The definition of Technology Readiness Levels is provided in Appendix A.

- **Space segment constraints**

The CaC target implies an ESA industrial contract in the range of ~ 200 M€ (e.c. 2014) for the space segment, including any ESA provision of payload elements. For comparison, the industrial cost (actual or estimated, depending on the mission's status) for previous M missions - Euclid (M2), PLATO (M3) and all other M3 candidates (MarcoPoloR, LOFT and EChO) - is in the range 320-360 M€, leading to an ESA CaC of around 600 M€. Therefore, the space segment for the M4 mission must be significantly "smaller and simpler" than for either Euclid or PLATO (both being ~ 2000 kg spacecraft at L2) or other M3 candidates. This leads to a suggested limit for spacecraft dry mass of 800 kg, including the structural mass for the propulsion system(s) and excluding the launcher adaptor. The wet mass can of course be larger, within the selected launcher capability.

Proposers are advised not to make a priori assumptions about possible industrial set-ups for the procurement of the spacecraft in their proposal, e.g., by claiming drastic reductions of spacecraft development costs through heavy reliance on existing developments. While this may appear tempting to minimise industrial costs for ambitious missions that would not otherwise fit within the programmatic constraints, experience shows that this is likely to underestimate the actual spacecraft cost for several reasons. These include the fact that 1) recurring costs are valid only if a full and true recurrence is reached for the product, while experience shows that in practice science missions almost invariably require specific adaptations, 2) the industrial landscape can evolve over several years and invalidate the assumptions underlying recurring costs, and 3) component obsolescence over the timescale of the implementation of the M4 mission will inevitably impose re-design and non-recurring costs. Additionally, a pre-defined industrial organisation scheme may not be compatible with the Science Programme constraints at the time of the M4 implementation. While the Agency will explore in due time all means to minimise the spacecraft development costs, including the use of recurring developments, the mission proposals should be robust against implementation schemes.

The payload mass is a key parameter, since the payload measurement capability drives the science case. The proposed figure – 300 kg – is a figure obtained by scaling the payload mass of previous missions such as Euclid or PLATO. The value applies to missions for which the spacecraft remains in Earth-bound orbits (in LEO/HEO or at L1/L2) and involving a small to moderate propulsion subsystem. For planetary and deep-space missions, the propulsion subsystem is generally much more demanding, as are other major spacecraft subsystems (e.g. power, communications, and possibly thermal control). Therefore, the instrumentation mass is recommended to be below 80 kg for such missions, unless duly justified by the proposers.

- **Mission operations**

The detailed mission operation costs (Mission Operations Centre and Science Operations Centre) will be mission-dependent. Planetary missions tend to have higher MOC costs and lower SOC costs than astrophysics missions. A reasonable cap for the operations cost is 15-20% of the CaC (including margins). The nominal in-orbit lifetime is recommended to be in the range of 3-3.5 years.

- **Launcher**

European launchers should be assumed. Currently, the two launchers that can reasonably be envisaged for M4 are Vega and Soyuz. Both launchers feature a re-ignitable upper stage (AVUM for Vega, Fregat for Soyuz). It is assumed that any evolution of European launchers within M4 timescale would provide equal or better performance than with Soyuz/Vega for all aspects. Shared Ariane 5 launches (as it was the case for SMART-1) are not excluded, but these would carry a number of additional constraints (e.g., the availability of a suitable launch opportunity at the desired time).

Vega is the less expensive European launcher and it is likely to be the appropriate choice for M4 where its performance will be adequate. The launcher is most appropriate for LEO/SSO orbits. Vega can also reach other orbits (e.g. L1/L2, or escape orbits for planetary missions) by including a jettisonable propulsion module providing the additional necessary ΔV capability, as demonstrated by LISA Pathfinder. However, in that case, the launch capability is generally well below that achievable with a Soyuz launcher, which allows access to a larger variety of orbits with the Fregat upper stage. Considering that the cost of a propulsion module will typically be in the range of 10-20 M€, the estimated cost of a Vega with a propulsion module becomes comparable – although still cheaper – than that of a Soyuz launch. Therefore, for the case of planetary and deep space missions, the choice between a Soyuz launch and a Vega launch is to be made on a case-by-case basis.

The guidelines in Table 1 apply for both Soyuz and Vega missions. As a consequence, depending on the actual space segment mass and on the selected orbit, some launch capability may be available for a passenger, which would reduce the actual M4 launch cost to ESA. The proposers may identify dual launch possibilities, which will anyhow be explored by ESA where relevant, but should avoid building the mission case on such dependence.

The Soyuz and Vega User Manuals [1, 2] provide a comprehensive description of the launchers and can be downloaded from the Arianespace web site (www.arianespace.com). Section 0 provides specific examples of mission profiles for each launcher to support the proposers in elaborating their mission case.

3 ANALYSIS OF POTENTIAL MISSION PROFILES

3.1 Introduction

This section provides examples of the launch capability for Vega and Soyuz launchers and for a variety of orbits that are likely to be of interest to science missions. The intention is to provide sufficient background and good order of magnitude values for the various mission parameters to enable proposers to design their mission concepts. In all cases, a specific mission analysis will be carried out for selected missions for optimising and fine-tuning the launcher performance to the specific mission needs.

All mass performance figures refer to the total launch mass and must therefore include the mass of the launcher adapter. Arianespace proposes standard (and qualified) adapters for both launchers [1, 2], with a mass of ~80 kg for Vega (77 kg for the adapter with a clamp-band diameter 937 mm) and ~ 100 kg for Soyuz (e.g. for the clamp-band diameter 1666 mm).

The mass performance figures report the launcher's intrinsic capability, regardless of the mission's cost. They thus do not take into account the constraints (related to the mission's cost) indicated in Table 1. For a given proposed mission to be feasible it must then meet both the affordability constraints (resulting in the indicative mass constraints reported in Table 1) *and* it must meet the launcher performance requirements. Thus, the two following conditions must be satisfied:

$M_{wet} + M_{adapter} < M_{launch}$ (launcher performance) and $M_{dry} < 800$ kg (mass limit in Table 1)

where:

M_{launch} is the launcher performance; $M_{adapter}$ is the launcher adapter mass; M_{wet} is the space segment wet mass (excluding the adapter); M_{dry} is the space segment dry mass (also excluding the adapter).

3.2 Vega launch

3.2.1 *Vega performance to LEO*

Vega is best adapted to circular, or near-circular low-Earth orbits. The standard launch orbits are Sun Synchronous Orbits (SSOs) between 400 and 1000 km of altitude. Figures 1 and 2 show the performance for various orbits. Some example orbit parameters are given in Table 2.

Orbit	Lift mass
700 km circ., i=90°	1430 kg
400 km SSO, i=97.03°	1480 kg
700 km SSO, i=98.19°	1325 kg
1000 km SSO, i=99.48°	1140 kg

Table 2: Vega performance for SSO. 700 km circular, i=90° (performance comes from [2], other values are extracted from Figure 1).

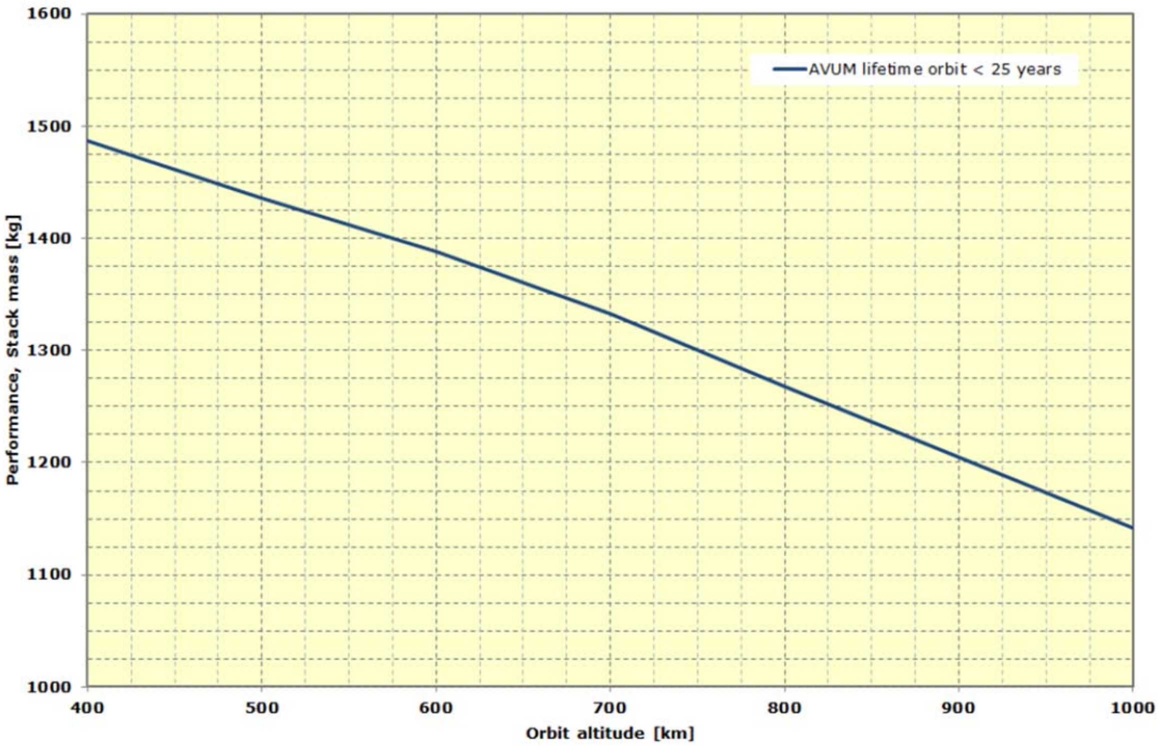

Figure 1: Vega performance for SSO orbits (with two AVUM boosts, from [2]).

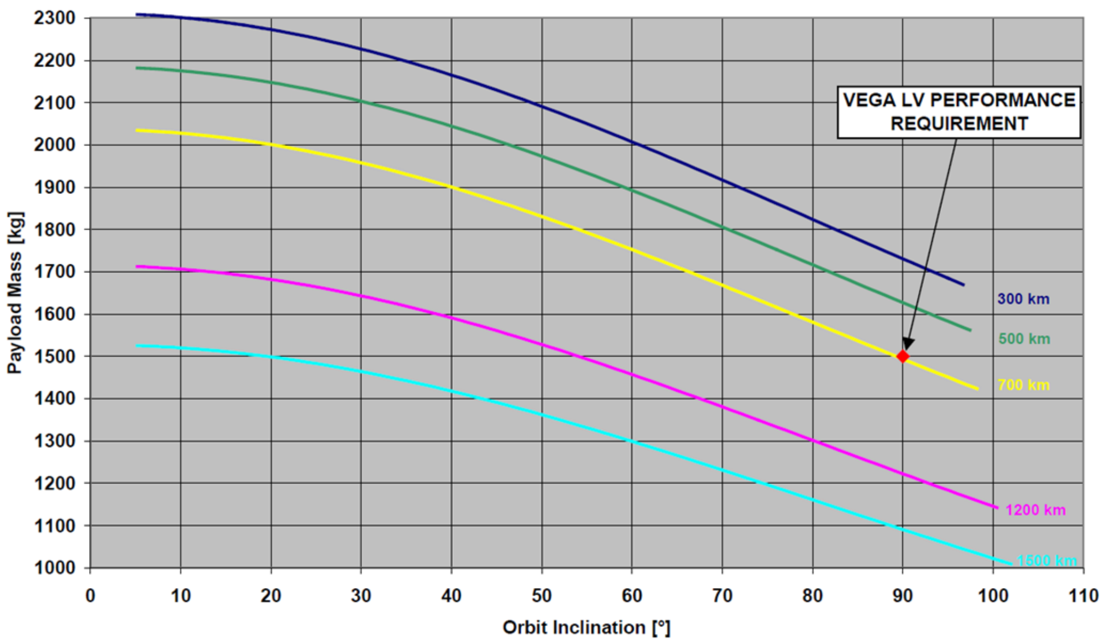

Figure 2: Vega typical performance for inclined circular orbits. It is recommended to include a mass margin of 5% with respect to these curves.

3.2.2 *Vega launches with a dedicated propulsion module*

Vega can provide access to high altitude orbits, such as missions to the Lagrange points or even interplanetary transfer missions, by making use of a dedicated propulsion stage, as illustrated by the case of the LISA Pathfinder mission which will be positioned in an L1 orbit and is briefly described below.

3.2.2.1 Orbits at the L1/L2 Lagrange points

LISA Pathfinder is planned for a launch in 2015, by using a Vega launcher and including a dedicated propulsion module for reaching its target L1 orbit. The launcher insertion orbit is a low-inclination elliptic orbit of 200 km x 1500 km; the key parameters of the LISA Pathfinder launch are summarised in Table 3.

LISA Pathfinder launch parameters	
Launch and orbit characteristics	
Launcher	Vega
Insertion orbit	Apogee 1500 km, perigee 200 km, inclination 5.4 deg
Launcher mass capability to insertion orbit	> 1963 kg, including launcher adapter
Final orbit	Large, eclipse-free Lissajous orbit around the L1 Lagrange point
Propulsion module	
Propulsion engine	Main engine: 450 N thruster, bi-propellant, re-ignitable, Isp = 320 s. Auxiliary engines for attitude control: Four pairs of 10 N bi-propellant thrusters.
Propulsion module dry mass	214 kg
Propellant mass	1250 kg
Propulsion module total mass	1464 kg
Total ΔV	3190 m/s, achieved in ~15 burns
Spacecraft	
Nominal in-orbit lifetime	6 months
Dry mass / Wet mass	428 kg / 435 kg
Platform mass	257 kg, including 51 kg for the Cold Gas micro-propulsion system
Payload mass	178 kg

Table 3: Key figures for the LISA Pathfinder launch (Vega launch with a propulsion module for reaching the final orbit at L1 Lagrange point).

Figure 3: View of LISA Pathfinder spacecraft mounted on its propulsion module.

3.2.2.2 Orbits at the L4/L5 Lagrange points and trailing orbits

The L4/L5 Lagrange points can be reached using the LISA Pathfinder approach, although with slightly higher fuel demands than for L1/L2. The analysis shown here assumes a Vega insertion in a generic 300 km circular, low-inclination LEO orbit, from which escape can be achieved in any direction through multiple burns of the propulsion module for apogee raising, followed by a final burn for insertion in the Earth escape hyperbola trajectory. 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 4. 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 (see Appendix B for the definition of 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 [km^2/s^2]	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 4: Approximate Earth-Sun 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 $C3 \geq 0\ km^2/s^2$.

3.2.2.3 Escape orbits

Based on the LISA-Pathfinder launch approach of using a propulsion module, one can estimate the mass capability for escape orbits with Vega. Figure 4: shows the approximate performance of a bi-propellant propulsion module into escape. As for the L4/L5 case, 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 insertion in the 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, see Table 3). Since the longer tank takes full benefit of 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 LPF case for which the mission profile was extensively optimised.

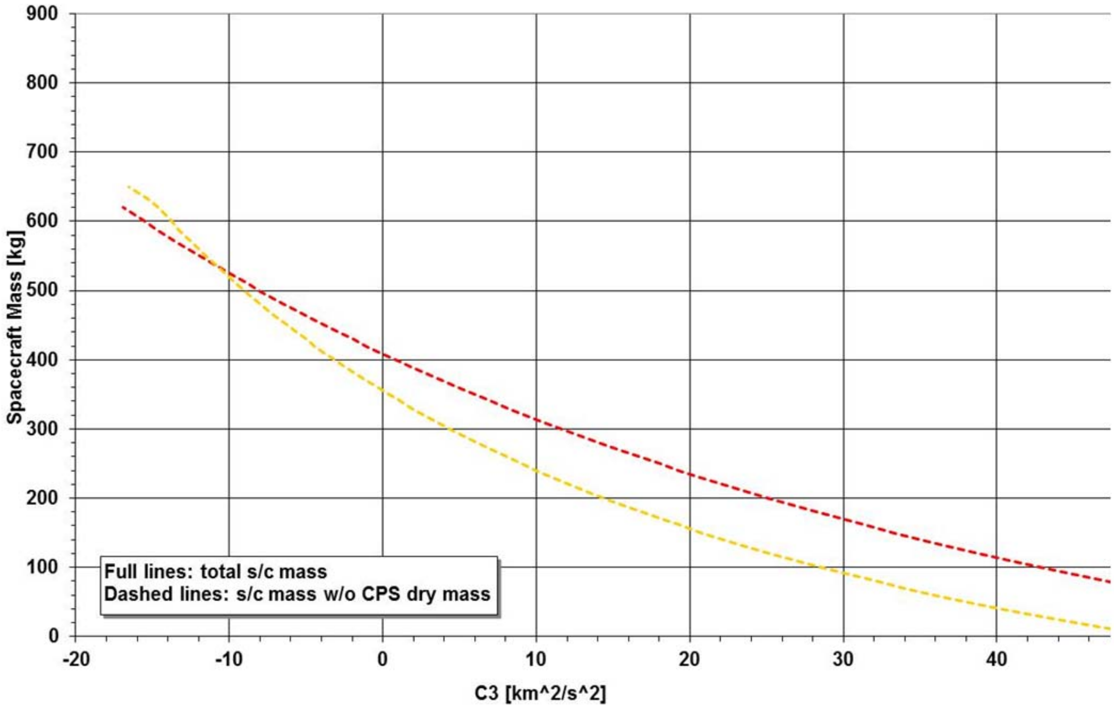

Figure 4: Approximate escape performance for a Vega launch with a bi-propellant stage, assuming launcher insertion in a 300 km LEO low-inclination orbit. The yellow line assumes a 2000 short tank (corresponding to the LISA-PF propulsion module). The red line assumes a 2000 long tank and takes full benefit 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; C3~ -2 km^2/s^2 for missions to the Moon; and C3 ~0 km^2/s^2 for L1/L2.

3.2.2.4 Venus and Mars cases

Figure 4 can be used for evaluating the mass capability of a Vega-launched mission to Mars and Venus, for which the C3 required for the interplanetary transfer is ~9 km^2/s^2. If an orbit around the target planet is envisaged, the mass performance need to be further reduced to take into account the orbit insertion ΔV. Examples are shown in Table 5 and Table 6.

Launch date	02/10/2024	04/11/2026
Esc. velocity [km/s]	3.48	3.172
Esc. declination [degree]	5	5
S/C wet mass [kg]	**301**	**322**
Mars arrival	12/08/2025	05/08/2027
Mars Orbit Insertion (including gravity losses) [m/s]	758	1105
S/C mass in 4-sol HEO [kg]	**231**	**219**

Table 5: Mars orbit insertion mission examples with Vega and bi-liquid propulsion module.

Launch date	07/06/2023	19/12/2024
Esc. Velocity [km/s]	3.127	2.683
Esc. Declination [degree]	-3.3	5
S/C wet mass [kg]	**324**	**342**
Venus arrival	26/10/2023	08/05/2025
Venus Orbit Insertion (including gravity losses) [m/s]	863	979
S/C mass in 2 day HEO [kg]	**240**	**243**

Table 6: Venus orbit insertion mission examples with Vega and a bi-liquid propulsion module.

The ~4-sol HEO Mars orbit is defined as 300 x 96000 km. The ~2 day HEO Venus orbit is defined as 300 x 123863 km.

From these HEO orbits, reducing the apogee until circularization into a 300 km altitude orbit around Mars or Venus will require an additional ΔV increment. This is shown in Figure 5 and Figure 6: , along with the resulting S/C wet to dry mass ratio.

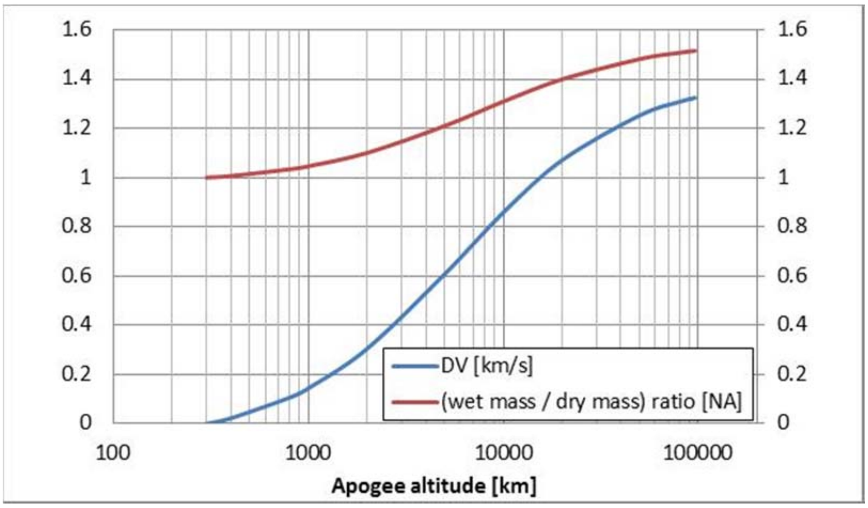

Figure 5: ΔV required to reduce the apogee of a HEO orbit around Mars down to a 300 km altitude circular orbit (blue curve), and resulting wet to dry mass ratio of the S/C to perform this ΔV assuming Isp = 325 s (red curve).

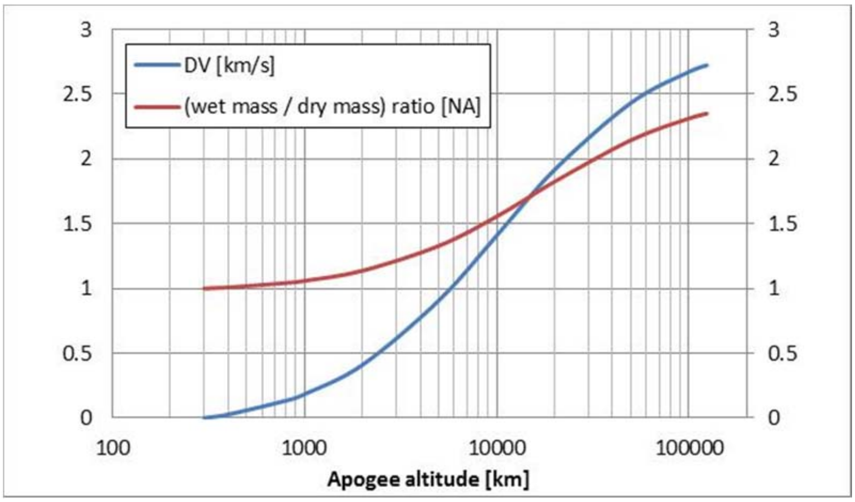

Figure 6: ΔV required to reduce the apogee of a HEO orbit around Venus down to a 300 km altitude circular orbit (blue curve), and resulting wet to dry mass ratio of the S/C to perform this ΔV assuming Isp = 325 s (red curve).

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

3.2.2.5 Alternative configurations for the propulsion module

The above analysis is based on a bi-propellant chemical propulsion engine. Alternative configurations can be envisaged for the propulsion stage, by making use of electric propulsion or solid –fuel propulsion stages.

- Electric propulsion stage:

The propulsion stage can make use of Solar-powered Electric Propulsion (SEP, used e.g. for SMART-1 and Bepi-Colombo) potentially without the need for a separate propulsion module. Electric propulsion engines feature low thrust (~ 0.1 N) and high specific impulse (Isp ~ 1500 s -2500 s. A dedicated analysis would be needed for specific mission cases to evaluate potential advantages. Examples of space qualified European SEP engines include:

- The PPS1350-G SNECMA engine, which was used for Smart-1 and Alphasat (Xenon fuel, 0.09 N thrust, 1500 W, ISP = 1660 s, total impulse 3.4E6 N.s)
- The T6 Qinetiq engine, used for Bepi-Colombo (Xenon fuel, 0.075 N thrust, 2430 W, Isp = 3710 s, total impulse 5.2E6 N.s)

- Solid propulsion stage:

Solid-fuel rocket motors produce high thrust levels (around 4 kN or more) with a slightly lower specific impulse than bi-propellant stages (around 290 s instead of 320 s). They are also not re-ignitable, and generally require spin stabilization to maintain attitude stability during manoeuvres (and therefore a de-spinning device following the manoeuvre). Their main interest is related to the high thrust level that can achieve a large ΔV in one go with very low gravity losses in comparison to bi-propellant engines. Hence the Van Allen belts have to be crossed only once, keeping the accumulated radiation dose low. The spin rate and acceleration level often lead to significant structural loads that preclude array deployment or payload commissioning prior to separation from the escape stage. Small solid-fuel motors are no longer manufactured in Europe, and would have to be purchased outside of Europe, potentially with export restrictions applying (e.g. from ATK in the USA). Solid-fuel motors may not necessarily be available in specific sizes so that the propellant mass, though variable, cannot always be tailored to the exact value needed, which can lead to mass penalties. Assuming the same Vega launch as in previous sections (~2300 kg into a 300 km low inclination LEO), Figure 7 shows a possible option, using the ATK STAR48B motor, which results in a somewhat better performance than with the bi-liquid propulsion stage. Since the STAR48B motor requires a minimum propellant load of 1608 kg for ignition, this option leads to C3 values always ≥ 12 km^2/s^2.

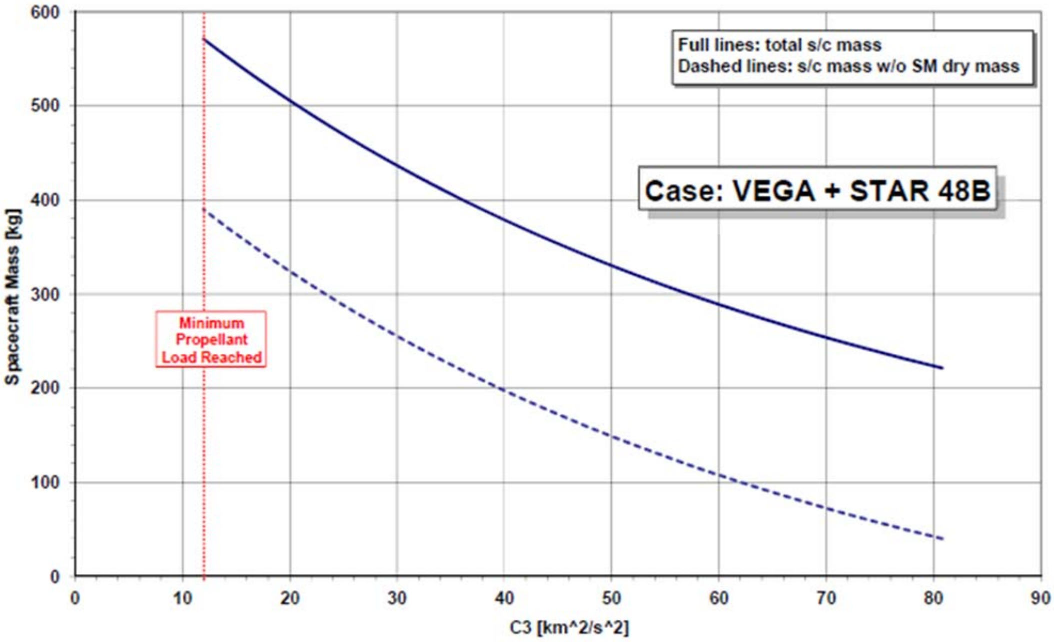

Figure 7: Performance as a function of C3 for a Vega launch using the STAR48B based solid-fuel propulsion stage (the dashed line provides the estimated useful delivered spacecraft mass without the solid propulsion module dry mass).

3.2.3 Summary of Vega based mission profiles

A summary of potential mission profiles with Vega is given in Table 7, based on the analyses developed in section 3.2.2. The Table includes useful information for the data downlink capability (see also section 3.4) and for the power subsystem. These apply regardless of the launcher used, and thus should also be used when designing Soyuz-based mission profiles.

Orbit	Performance	Typical transfer duration	Typical science TM data rates	Power
LEO	**2.300 kg** @ 300 km (i=5°) **1.480 kg** @ 400 km SSO **1.140 kg** @ 1000 km SSO	< 1 day	**X band:** **20 – 200 Mbps** S band also possible (~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	**1.963 kg** @ 200 x 1500 km **~ 650 kg** @ 300 x 36000 km (both at equatorial i=5.4°)			
Moon before insertion [after insertion in 300 km orbit]	**~ 420 kg** **[~320 kg after insertion]** (C3 = -2 km^2/s^2)	**< 1 week** (direct transfer) **~70 – 130 days** (low energy transfer)	**X band: ~150 Mbps**	
Sun Earth L1/L2	**~ 420 kg** (C3 = 0 km^2/s^2)	**~1 month**	**X band: 5-10 Mbps** **Ka band: 75 Mbps**	
Heading/trailing heliocentric orbit and [Sun-Earth L4/L5]	**≤ 400 kg** (C3 > 0 km^2/s^2) [~ 230-350 kg for L5]	**14 – 50 months** (in increments of 1 year)	**Ka band: 150 kbps**	
Venus before insertion [after insertion in 2-day HEO]	**~ 340 kg** **[~240 kg after insertion]** (for C3 ≈ 7.5 km^2/s^2)	**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.
Mars Before insertion [after insertion in 4-sol HEO]	**~ 320 kg** **[~220 kg after insertion]** (for C3 ≈ 10.5 km^2/s^2)	**9-11 months** (conjunction transfer) **21 – 26 months** (1.5 revolution transfer)	**X band: 38 – 230 kbps** (superior vs. inferior conjunction)	Approximately 0.36 times the value at Earth

Table 7: Potential mission profiles with Vega, using a bi-liquid Chemical Propulsion System for orbits above HEO. The low inclination (equatorial) LEO performance is to be confirmed with Arianespace.

3.3 Soyuz launch

Soyuz is the "middle performance" launcher available to ESA, and is being used for a variety of missions in the Science Programme. A Soyuz launch provides a better launch performance than Vega (to be traded off against the higher cost), which can be enabling for some missions requiring access to orbits above LEO. The Soyuz performance for escape orbits is illustrated in both Fig. 8 and Fig. 9, with more details available from the User Manual. Note that while in principle Soyuz can carry large spacecraft, the limitations of Table 1 should be considered also in this case.

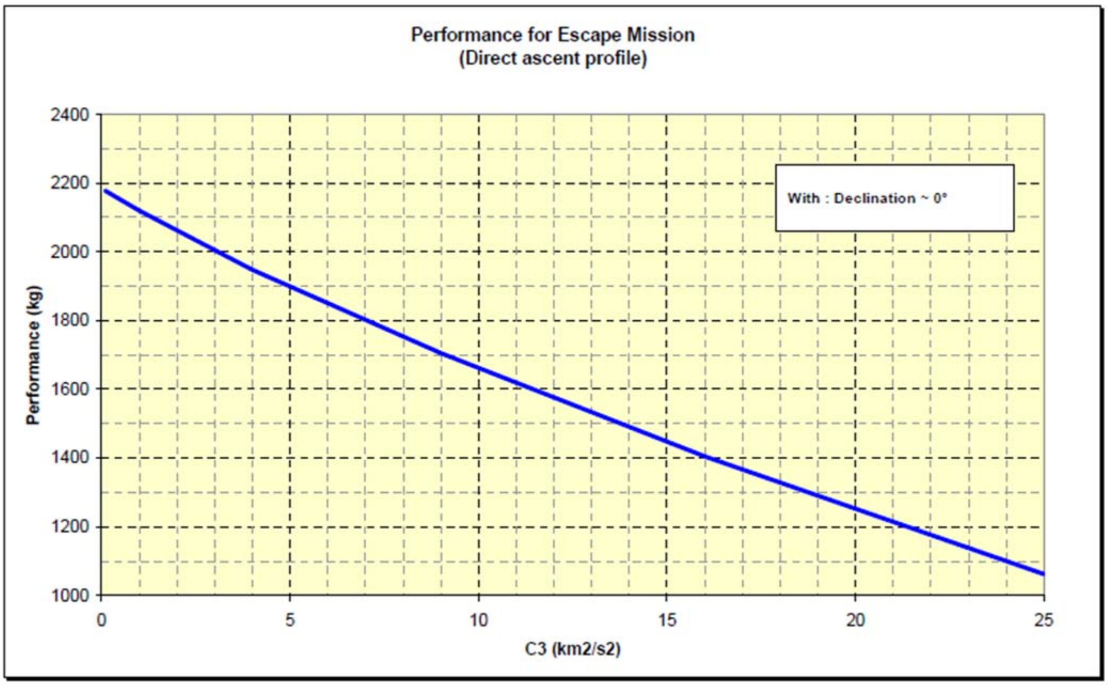

Figure 8: Soyuz performance for Earth escape mission as function of C3 at a declination of ~0° (from [1]).

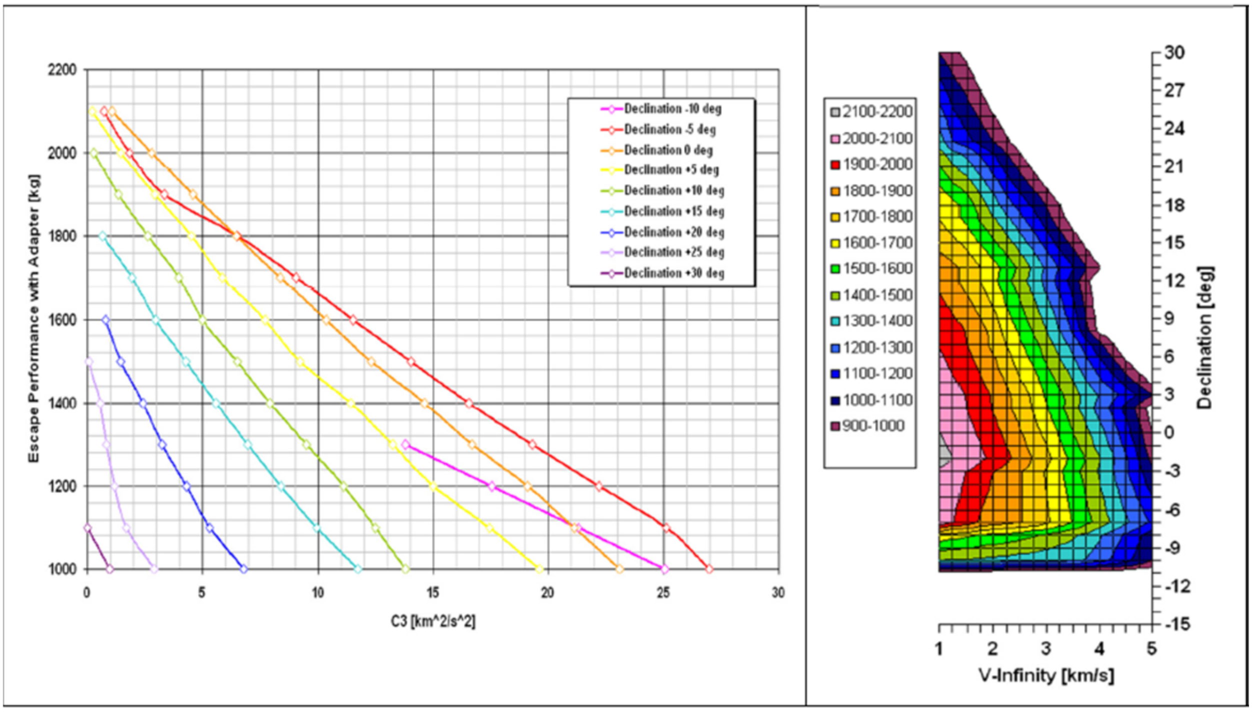

Figure 9: Soyuz-Fregat direct escape performance for launches from Kourou.

Table 8 provides a summary of the Soyuz performance for selected orbits.

Orbit	Performance
Sun-Earth L1/L2	**2160 kg**
Sun-Earth L4/L5	**2150 – 2050 kg** (depending on transfer time, see Table 4)
Mars **(before orbit insertion)**	**~ 1650 kg** (for C3 ≈ 10.5 km²/s²)
Mars **(after insertion into 4-sol orbit)**	**~ 1170 kg**
Mars **(after circularisation at 300 km)**	**~ 780 kg**
Venus **(before orbit insertion)**	**~ 1750 kg** (for C3 ≈ 7.5 km²/s²)
Venus **(after insertion into 2 day orbit)**	**~ 1250 kg**
Venus **(after circularisation at 300 km)**	**~ 525 kg**

Table 8: Mass capability for a range of mission profiles with a Soyuz launch. The same ΔV and wet/dry mass ratios as in section 3.2.2.4 were used for Mars and Venus insertion and circularisation. In designing a viable M4 mission candidates the constraints from Table 1 have also to be considered.

From Table 8, one can see that the Soyuz's performance allows envisaging a moderate-size mission to Mars or Venus, including the possibility of a small atmospheric/surface probe to either planet. In this case the small/medium size orbiter would act as a data relay from the surface probe to Earth. The probe must be designed to withstand the planet atmospheric entry and landing, and operate the landed science instruments over the targeted lifetime (long term survivability on Venus is out of scope due to the harsh surface environmental conditions). In practice, this requires the science instrumentation mass to be a small fraction of the probe mass. Reference [3] presents a study case for a Venus atmospheric probe, where the instrumentation mass is ~12 kg for a probe entry mass of ~270 kg. Reference [4] presents a study case for a Mars surface probe, where the instrumentation mass is ~16 kg for a probe entry mass of ~400 kg. Of course in designing a viable M4 mission, the constraints from Table 1 (including the constraint on the total spacecraft mass) have to also to be applied.

3.4 Data transmission and link budget considerations

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

$$\text{Data Rate} \propto P.(D_t/\lambda)^2.\ (D_r/\lambda)^2.\ (\lambda/d)^2$$

where:

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

The typical data rates given in Table 7 refer to examples of communication subsystem parameters from previous missions:

- Small platforms in LEO/HEO (indicative, from small European platforms): X band, < 10 cm patch, horn, helix or isoflux LGA, ≤ 10 W output power, 3 to 15 m ground antenna, data rates ~ 20 to 200 Mbit/s.
- Euclid: L2 orbit, Ka band, 50 cm HGA, 35 W output power, 35 m ESTRACK ground antenna, data rate 75 Mbps.
- Mars-Express: X-band, 1.6 m HGA (1.3 m for VEX), 65 W output power, 35 m ground antenna, data rates 38 – 230 kbps (superior – inferior conjunction). Note: at inferior conjunction, the potential data rate achievable by Mars Express is much higher (see Figure 10) 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.
- SolO: Ka band, 1.1 m HGA, 35 W output power, 35 m ground antenna (re-use from BepiColombo), data rate 150 kbps.

Figure 10 shows an analysis (based on the Mars Express data rate) of the achievable X band data rate from Mars as a function of the output power and the spacecraft antenna size.

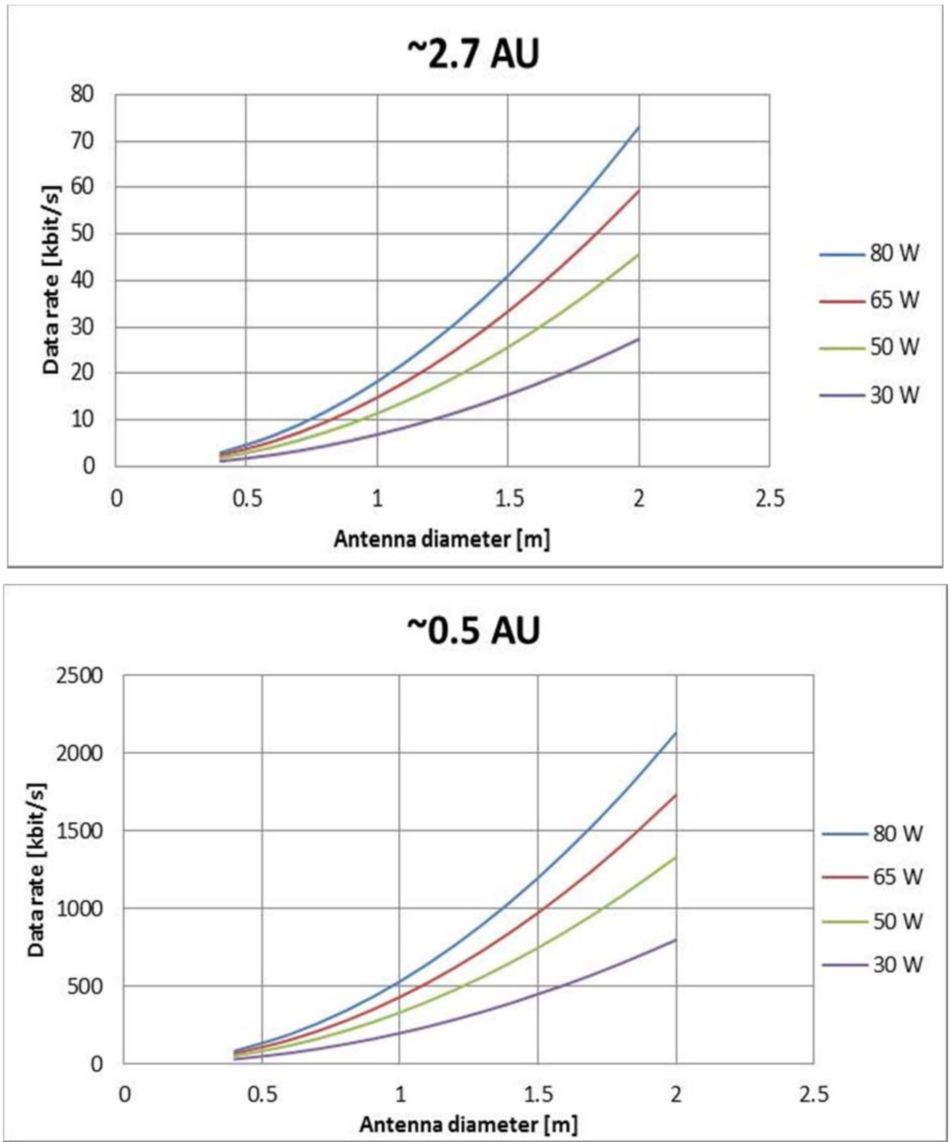

Figure 10: Data rate for a Mars mission, based on scaling the Mars Express data rate, as a function of antenna diameter and output power, for superior (top) and inferior (bottom) conjunctions.

3.5 Space debris regulations

All ESA missions (see reference [5]) have to ensure that no additional orbital debris will contaminate the protected regions (Figure 11). 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.

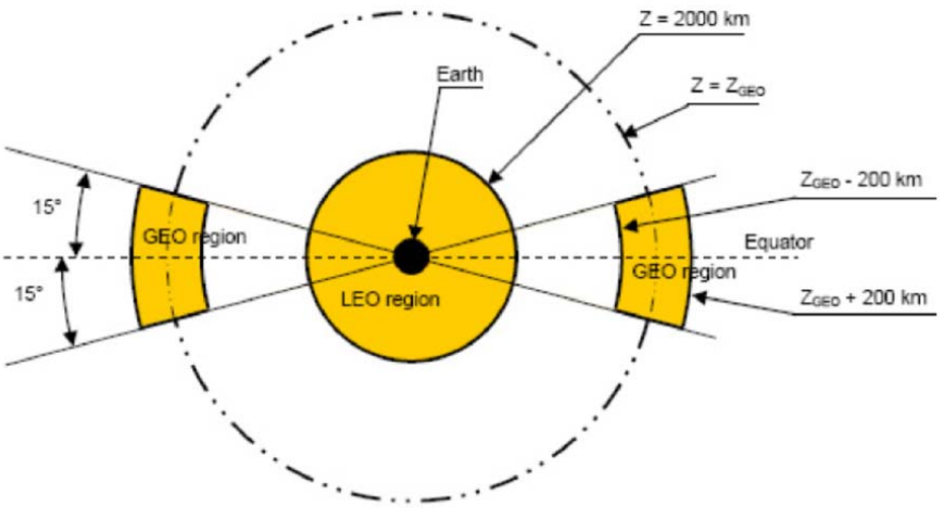

Figure 11: LEO and GEO protected regions [4].

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 potentially the case here for LEO missions launched with Vega (S/C mass potentially between 1000 and 2000 kg in LEO/HEO). For small missions (typically < 1000 kg), an un-controlled re-entry is acceptable, as long as it happens within 25 years.

This requirement applies to the S/C, as well as to any other debris generated by the mission, such as LV upper stages, multi-S/C adapters, ejectable covers etc.

The ΔV required for this manoeuvre will need to be included in the sizing of the propulsion subsystem. As a worst-case estimate, this ΔV can be calculated as follows:

- For an un-controlled re-entry manoeuvre, the perigee should be lowered to an altitude ≤ 60 km. Depending on the initial orbit, more efficient solutions might include placing the S/C into a higher graveyard orbit, or into a very low circular orbit with a Hohmann transfer and let atmospheric drag lower the altitude naturally until re-entry is achieved within 25 years (this depends on the Solar activity, but typically requires lowering the spacecraft altitude to ≤ 550 km by using the on board propulsion system at the end of life).
- For a controlled re-entry manoeuvre, the perigee should be lowered to an altitude of 0 km.

4 SOME MISSION EXAMPLES AND HERITAGE

Table 9 provides examples of European science missions of relevance to M4 Call.

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'
MEX (same platform as VEX)	2003 Soyuz 2-1b with Fregat upper stage	Mars (330 km x 10.530 km i=86.9°)	1223 kg [116 kg]	650 W	457 kg Bi-propellant	38-230 kbit/s X band	APE = 0.15°
Lisa Pathfinder	2015 Vega with propulsion module	Sun-Earth L1	1910 kg Includes 214 kg prop. module dry mass + 1250 kg propellant. [178 kg]	650 W	Propulsion module + Bi-propellant + cold gas	52 kbit/s X band	APE = 0.05°
Corot	2006 Soyuz 2-1b	LEO 896 km 90°	668 kg [300 kg]	530 W	90 m/s Mono-propellant	1.5 Gbit/day S band (722 kbit/s)	APE = 0.5'' (telescope used as a FGS)
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 FGS)

Table 9: Examples of European science missions, completed or under development. Data rates for orbits beyond L1/L2 are typically achieved with a ground station contact of 6 to 8 hours per day.

APPENDIX A - TRL DEFINITION (ISO SCALE)

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.

Technology Readiness Level	Milestone achieved for the element	Work achievement (documented)
		Model definition for the critical function verifications. Model test reports.
TRL 7: Model demonstrating the element performance for the operational environment	Performance is demonstrated for the 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 performance requirements, including 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 10: Summary definition of the ISO TRL levels (Courtesy from ISO. For further details, the reader is invited to refer to the ISO document 16290 [6]).

APPENDIX B – 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
- μ/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 12 and Figure 13. 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 escape velocity required.

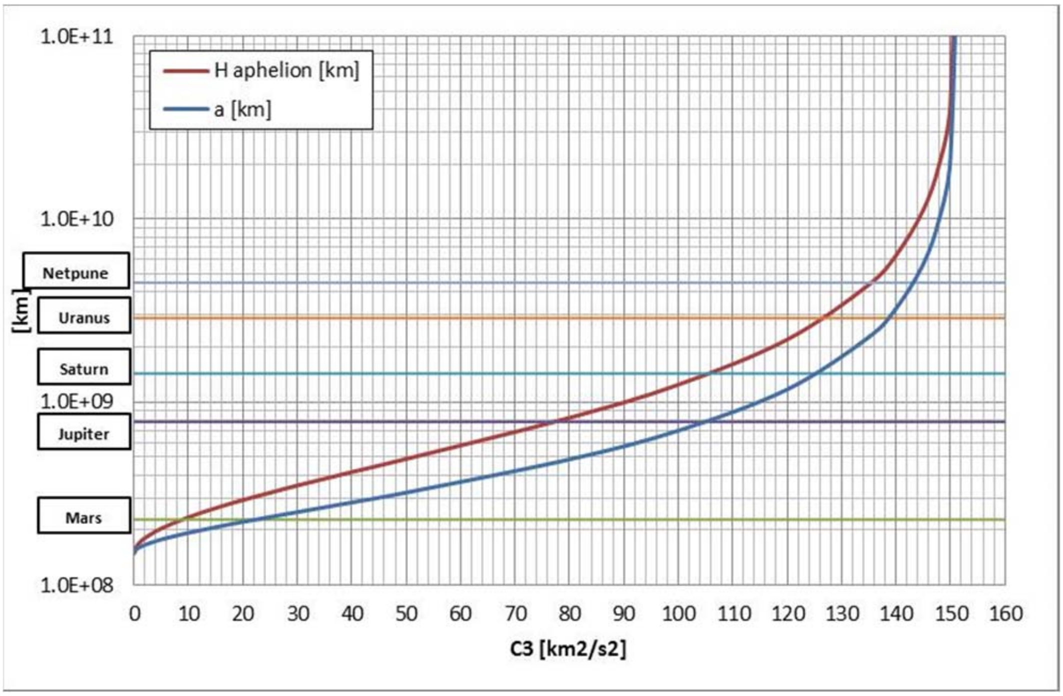

Figure 12: C3 values required to reach the external planets, assuming direct Hohmann transfer. The semi-major axes of the orbit of the external planets are indicated. H and a are respectively the aphelion and semi-major axis of the transfer ellipse.

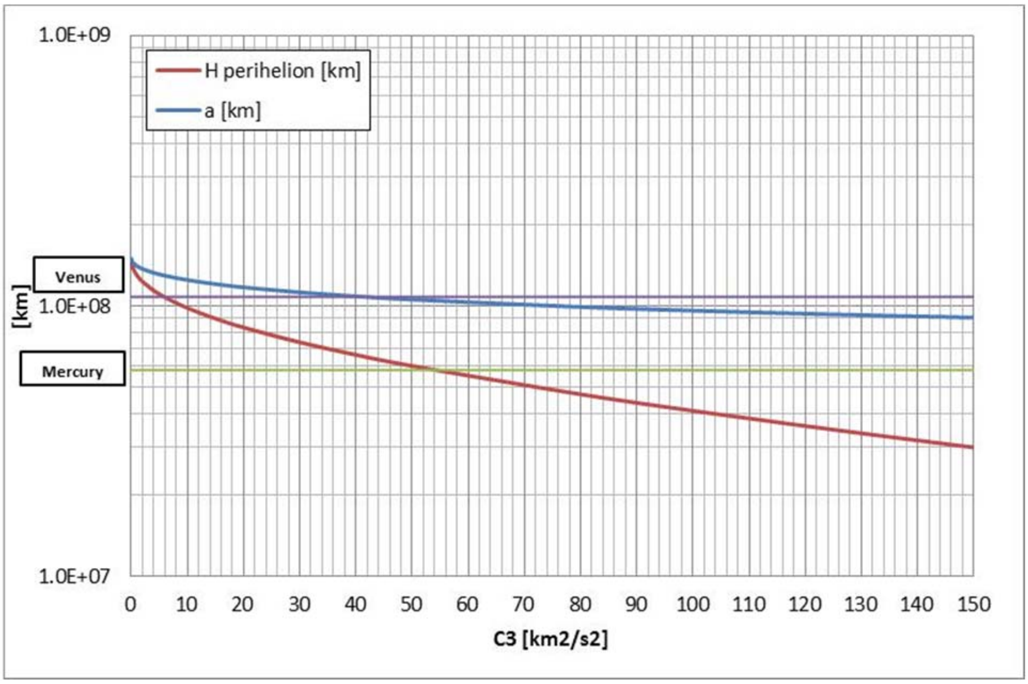

Figure 13: Same as Figure 12 for the inner planets, with H being now the perihelion of the transfer orbit.