DOCUMENT

M5 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 M5 Call. Its main objective is to provide technical information to help proposers to define their mission concept to the level required to enable the evaluation of the mission's technical feasibility.

Reference information from previous ESA missions that may be relevant to the preparation of the proposals can also be found at: http://sci.esa.int/home/51459-missions/.

1.2 Reference documents

[1] Soyuz User's Manual, issue 2.0, www.arianespace.com, 2012.

[2] Vega User's Manual, issue 4.0, www.arianespace.com, 2014.

[3] Ariane 5 User's Manual, issue 5.1, www.arianespace.com, 2011.

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

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

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

[7] ECSS-E-HB-11A DIR1, TRL guidelines, www.ecss.nl, 2016.

[8] Rayman et al, "Dawn: a mission in development for exploration of main belt asteroids Vesta and Ceres", Acta Astronautica 58 (2006) 605-616.

[9] ECSS-E-AS-11C, Adoption notice of ISO 16290, www.ecss.nl, 2014.

[10] ESTRACK facilities manual (EFM), issue 1.1, 2008.

[11] Marco Polo R CDF report, CDF-123(A), http://sci.esa.int/marcopolo-r/, 2011.

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
CREMA	Consolidated Report on Mission Analysis

DHS	Data Handling System
DV	Delta V (also ΔV)
EChO	Exoplanet Characterisation Observatory
EoL	End of Life
EPC	Etage Principal Cryogenic
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
MS	Member States
NEO	Near Earth Object
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
TBD	To Be Defined
TGO	Trace Gas Orbiter
TM	Telemetry
TRL	Technology Readiness Level
VESPA	Vega Secondary Payload Adapter
VEX	Venus Express
VOI	Venus Orbit Insertion

2 GENERAL GUIDELINES

The M5 Call is targeting a Cost at Completion (CaC) to ESA of 550 M€ and a mission adoption in 2021 (with launch around 2029). The purpose of this technical annex is to support proposers in defining mission profiles that are compatible with these programmatic boundaries.

General guidelines are summarised in the Table 1.

Element	Request	Comment
ESA CaC	≤ 550 M€	Includes all elements to be funded by ESA. Excludes MS contribution (e.g. for payload elements and for the Science Ground Segment), and international collaboration.
TRL	TRL=6 by mission adoption (November 2021)	ISO scale, see Appendix A. In practice, TRL 6 should be reached by the time of the mission selection for schedule critical elements, since the Technology Readiness will be one important element of the decision process.
International collaboration	No specific constraint beyond cost/schedule compatibility	The mission can be led by ESA, with or without international partnership, or led by the partner with ESA participation.
Launcher	If paid by ESA, must be one of the new European launcher family: - Vega-C - Ariane 62 - Ariane 64	See Chapter 3 for launcher performance. Considering the CaC target, the use of VEGA-C or Ariane 62 is recommended for M5 in case of ESA-only missions, although Ariane 64 is not excluded. Any launch from outside Europe requires careful consideration with regard to compliance with Export Control Regulations (e.g. for a launch from China).
S/C constraints	No specific constraints beyond programmatic compatibility	See Chapter 4 for system design aspects. Also depends on eventual international collaboration scenario and launcher selection.
Operations and lifetime	No specific constraints beyond programmatic compatibility	

Table 1: M5 Call general guidelines.

3 ANALYSIS OF SOME POTENTIAL MISSION PROFILES

3.1 Introduction

European launchers should be assumed, unless alternatives are made available by international partners. Currently, the situation of European launchers is evolving therefore some degree of uncertainty is present. The best current knowledge includes the following launchers:

- Current launchers from Kourou:
  - Vega: to be replaced by Vega-C when available with some overlap, uncertain beyond ~2020.
  - Soyuz: to be replaced by Ariane 62 when available with some overlap, uncertain beyond ~2024.
  - Ariane 5: to be replaced by Ariane 62/64 when available with some overlap, uncertain beyond ~2024.
- Future launcher nominal schedule:
  - Vega-C: Maiden flight planned in Q4 2018.
  - Ariane 62 and 64: Maiden flight planned in 2020, operational capability in 2023.

All current launchers will most likely not be available for M5. Their performance is given here as reference only (and also because the performance of the future launchers in most orbits is not firmly known today and is sometimes specified with regard to the performance of current launchers). At the time of writing this document, both Ariane 6 and Vega-C have not yet completed their Preliminary Design Review and their performance is thus subject to evolution. The general approach in this document is to consider reasonably conservative assumptions. Performance figures will be updated in due time as available.

All mass performance figures refer to the total launch mass and must therefore include the mass of the launcher adapter (see [1], [2] and [3], and Appendix C).

Shared launches are not excluded and can lower the launcher price and ESA CaC, but these would carry a number of additional constraints (e.g. the orbit compatibility and availability of a suitable launch opportunity at the desired time). Shared launch assumptions will be carefully evaluated by ESA and may not be retained, in particular if they are targeting orbits that are seldom used by the commercial spacecraft market (mainly driven by telecommunication spacecraft and to a lower extent Earth Observation spacecraft).

This Chapter provides examples of the launch capability for the European 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 in later phases for selected missions for optimising and fine-tuning the launcher performance to the specific mission needs.

Considering the CaC target, the use of Vega-C or Ariane 62 for M5 is recommended, although Ariane 64 is not formally excluded.

3.2 Current European launchers

3.2.1 Vega

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

Orbit	Performance
700 km circular, 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 to various LEO orbits.

Detailed performance curves can be found in the Vega User Manual [2], in particular:

- Performance to SSO orbits with two AVUM boosts in Figure 2.4.1e.

Performance for other (non-SSO) inclinations in LEO orbits can also be found in the old version of the User Manual (v3.1, 2006, no longer applicable, for reference only) in Figure 2.4. It is recommended to include a mass margin of at least 5% with respect to this and note that only SSO inclinations are guaranteed as indicated in [2] (in particular inclinations between 15° and 40° are uncertain due to launch range and stage fallout safety constraints).

3.2.2 Soyuz

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 (at a correspondingly higher cost), which can be enabling for some missions requiring access to orbits above LEO.

Detailed performance curves can be found in the Soyuz User Manual [1], in particular:

- Performance to SSO orbits in Figure 2.4.2b.
- Performance to sub- and super-GTO orbits in Figure 2.4.1.2a.
- Performance to HEO orbits in Figure 2.4.4a.
- Performance for Earth escape missions as a function of C3 (see Appendix B for the definition of C3) at a declination of ~0° in Figure 2.4.5a.

In addition, the performance for Earth escape missions at different declinations is shown in Figure 1.

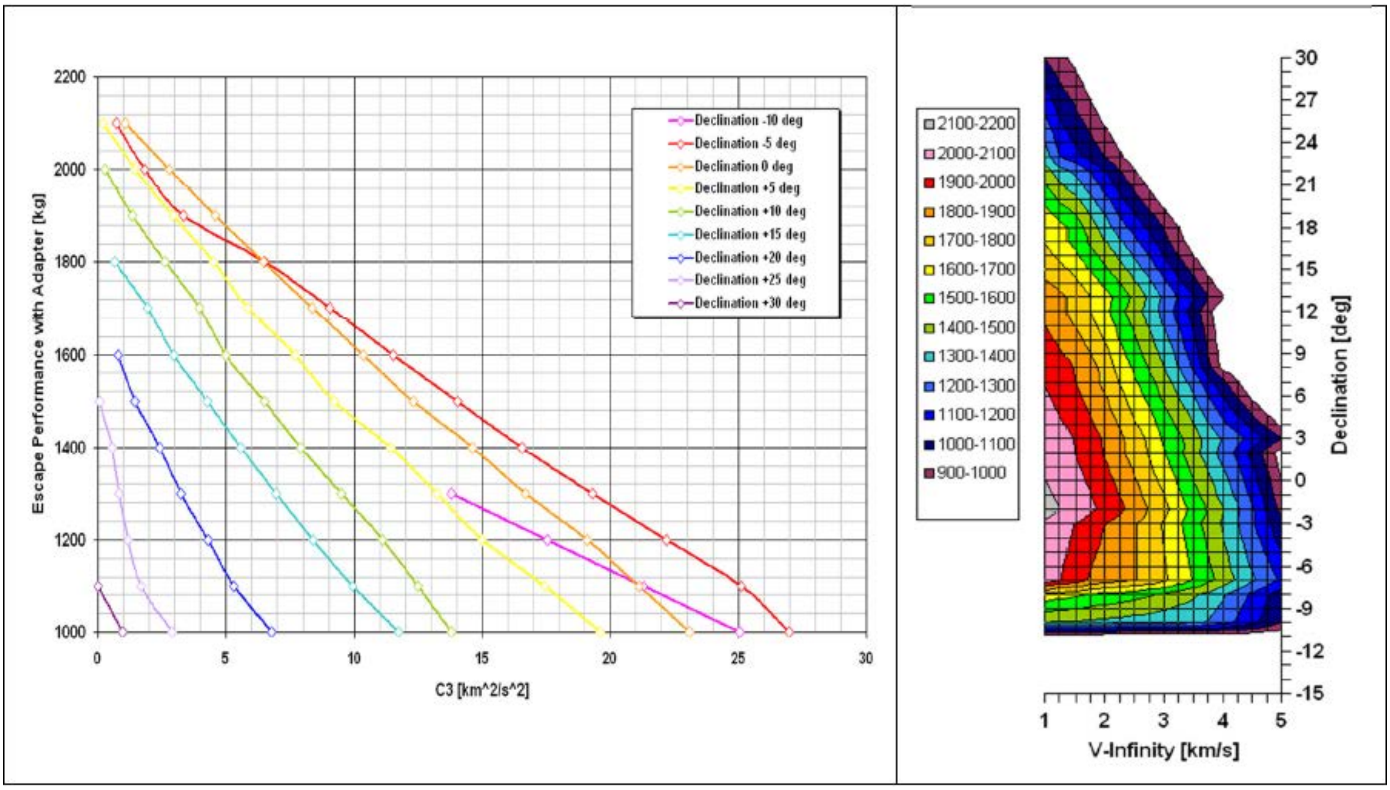

Figure 1: Soyuz-Fregat direct escape performance from Kourou for other declinations.

Table 3 provides a summary of the Soyuz performance for selected orbits.

Orbit	Performance
SSO (820 km)	**4400 kg**
GTO	**3250 kg**
HEO (apogee at Moon)	**~2310 kg**
Sun-Earth L1/L2	**2160 kg**
Earth escape to Sun-Earth L4/L5	**2150 – 2050 kg** (higher for longer transfer times)
Earth escape to Venus (before orbit insertion)	**~ 1750 kg** (for C3 ≈ 7.5 km²/s²)
Earth escape to Mars (before orbit insertion)	**~ 1650 kg** (for C3 ≈ 10.5 km²/s²)

Table 3: Indicative mass capability for a range of mission profiles with a Soyuz launch.

Based on the performance values from Table 3 for reaching Venus and Mars (details for standard Venus and Mars transfers are given in Appendix D), orbit insertion around these planets can be performed using the propulsion subsystem of the S/C. This would lead to the dry masses indicated in Table 4.

Orbit	Performance	ΔV	Wet/dry mass ratio
Venus (before orbit insertion)	~ 1750 kg (for C3 ≈ 7.5 km^2/s^2)	NA (from Table 3).	
Venus (after insertion into 2 day orbit)	~ 1250 kg	1.05 km/s	1.40
Venus (after circularisation at 300 km)	~ 525 kg	2.70 km/s	2.38
Mars (before orbit insertion)	~ 1650 kg (for C3 ≈ 10.5 km^2/s^2)	NA (from Table 3).	
Mars (after insertion into 4-sol orbit)	~ 1170 kg	1.07 km/s	1.41
Mars (after circularisation at 300 km)	~ 780 kg	1.26 km/s	1.50

Table 4: Orbit insertion around Venus and Mars after a Soyuz launch using the S/C propulsion subsystem. The wet/dry mass ratios indicated assume the S/C propulsion system has an Isp = 317 s.

The ~2 day HEO Venus orbit is defined as 300 x 123 863 km.

The ~4-sol HEO Mars orbit is defined as 300 x 96 000 km.

The reduction of the apogee of the orbit around Mars or Venus until orbit circularisation at 300 km can be achieved with aerobraking (demonstrated by Venus Express), saving the circularisation ΔVs indicated in Table 4. A small ΔV allocation of ~120 m/s is considered sufficient to perform such a manoeuvre. Note that for Venus, even at 300 km altitude, atmospheric drag is non-negligible and needs to be taken into account.

From Table 4, 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 [4] 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 [5] presents a study case for a Mars surface probe, where the instrumentation mass is ~16 kg for a probe entry mass of ~400 kg.

3.2.3 Ariane 5

Ariane 5 is the "high performance" launcher available to ESA. It can be used for any Earth orbit, including typical GTO transfers and HEO orbits, as well as interplanetary missions. Its performance is detailed in the Ariane 5 User Manual [3] and summarised in Table 5.

Orbit	Performance
SSO (800 km)	≥ 10 t
GTO	≥ 9.5 t
HEO (31600 km x 250 km, i=39.5°)	9.2 t
Moon transfer	7 t
L2 transfer	6.6 t
Interplanetary (C_3 = 12.1 km²/s²)	4.1 t (ECA version)

Table 5: Ariane 5 performance to typical orbits (from [3]).

For Earth escape orbits, the Ariane 5 performance is further detailed in Table 6 for various declinations and V_∞.

	Asymptote declination (degree)																											
Hyperbolic velocity (km/s)	-50	-45	-40	-35	-30	-25	-21	-20	-15	-10	-9.8	-5	-4	-3.5	-3.2	-3	-2.9	-2.8	-2.5	-2.4	-2.2	-2	-1	0	5	10	15	20
0.5	5302	5477																						6191	6078	5835	5474	5006
1	5021	5239	5441																					6059	5910	5615	5195	4667
1.5	4604	4882	5124																					5844	5648	5291	4805	
1.7																					6280							
1.9																				6134								
2	4044	4401	4697	4924															6080					5549	5295	4870	4316	
2.3																		5881										
2.4																	5802											
2.5		3795	4161	4443																				5180	4856	4359		
2.6																5630												
2.75															5504													
2.8															5476													
3			3522	3868	4167																			4744	4340	3774		
3.05													4741	5264		4773						4767	4750	4726				
3.15													4664			4679						4676	4658	4631				
3.3							3892																					
3.383													4445			4444						4404	4384	4355				
3.4							3818																					
3.5				3210	3563	3836	3680				2659													4251	3763	3136		
3.611											2666	4213																
4					2897	3220																		3714	3141			
4.5					2190	2559						3304												2155				
5								2244				2785																
5.5												2260																

Table 6: Ariane 5 ECA estimated performance for escape missions. Declinations around -15° are taken out as they would result in an EPC impact area on Africa. Black values are from the ExoMars 2008 CREMA, red values from an Arianespace study for Laplace, and blue values from a recent Arianespace study for JUICE. Note that all these examples assume mission specific constraints, but they provide a good order of magnitude for the launcher performance.

The specific cases of Venus and Mars can be derived using the same C3 and wet/dry mass ratios indicated in Table 4 (section 3.2.2). This is summarised in Table 7.

Orbit	Performance
Venus (before orbit insertion)	**~5.5 t** (C3 > 7.5 km²/s²)
Venus (after insertion into 2 day orbit)	**~3.9 t**
Venus (after circularisation at 300 km)	**~1.7 t**
Mars (before orbit insertion)	**~4.4 t** (C3 > 10.5 km²/s²)
Mars (after insertion into 4-sol orbit)	**~3.1 t**
Mars (after circularisation at 300 km)	**~2.1 t**

Table 7: Indicative Ariane 5 capability for Venus and Mars missions.

Reduction of the apogee of the orbit around Mars or Venus until circularisation at 300 km can also be achieved by aerobraking, as already indicated in section 3.2.2, saving a significant amount of ΔV and mass compared to the values indicated in Table 7.

3.3 Future European launchers

3.3.1 Vega-C

The reference performance for Vega-C is:

- > 1945 kg at 700 km circular polar orbit
- > 1505 kg at 800 km SSO

For most other orbits, it is assumed that Vega-C should perform as good as or better than Vega (see section 3.2.1), with a scaling similar to the SSO orbit reference case given above (TBC).

3.3.2 Ariane 6

The performance specifications of Ariane 6 are given in Table 8 and Table 9

The launch system net performances for GTO and SSO in single launch shall be :

Mission	Reference Orbit (Zp / Za / i / ω_p)	Net Performance
GTO	250 km / 35 786 km / 6° / 178° with direct de-orbitation* after P/L release	**≥ 5t**
SSO	800 km / 800 km / 98.6° with direct de-orbitation* after P/L release	**≥ 4,5t**

**with controlled re-entry*

Table 8: A62 performance objective.

The launch system performance* for commercial P/Ls shall be :

Mission	Reference Orbit (Zp / Za / i / ω_p)	Gross Performance
GTO	180 km / 35 786 km / 6° / 178° with direct de-orbitation** after P/L release	**≥ 10.5t**

** Including Payload Adaptor Fittings and dual launch adaptation structure*
*** with controlled re-entry*

Table 9: A64 performance objective.

In addition, the performance of A62 to the Sun Earth L1/L2 points is anticipated to be of the order of ~3.5 t.

For other orbits, it is recommended to use the Soyuz and A5 performance as a worst case (TBC) until further information is available from the Ariane 6 project. Different mission classes have been defined and are presented in the following sections.

Note that both A62 and A64 can be flown in a dual launch configuration. Additional information on potential A6 fairing options and LV – S/C adapter is available in Appendix C.

3.3.2.1 Class 1: standard missions

This class of missions will not require any special custom requirements.

A62 Configuration in single launch:

Mission	Orbit characteristics			
	Perigee altitude Zp [km]	Apogee altitude Za [km]	Inclination i [°]	Argument of perigee w_p [°]
SSO	[500, 900]	Za = Zp	Sun Synchronous linked to Za, Zp	N/A
GTO	[180, 400]	[35586, 35986]	[1, 8]	[175, 180]
L2 Transfer Orbit*	Free	[900000, 1550000]	Free	Free

Mission	Orbit characteristics		
	Semi axis a [km]	Inclination i [°]	Argument of perigee w_p [°]
Transfer Galileo 2G (A62) 2 S/C 1 or 2 S/C	 3500 3500 km < a < 23700 km	 56 56	 Free Free

Table 10: Class 1 A62 orbits (in single launch configuration).

A64 configuration (dual launch)

Mission	Orbit characteristics			
	Perigee altitude Zp [km]	Apogee altitude Za [km]	Inclination i [°]	Argument of perigee w_p [°]
GTO	[180, 400]	35786 ± X	[1, 8]	[175, 180]
GTO+	≥ 2200*	35486 ± X	[1, 8]	Free
GTO / GTO+	GTO: [180, 250] GTO+: ≥ 2200*	GTO: 35486 GTO+: 35486	[6, 7]	[175, 180]
MEO Equatorial	Zp ~ 7000	Za ~ 15000	[6, 7]	Free

* parameters defined by LSP for customer service

Table 11: Class 1 A64 orbits (in dual launch configuration).

3.3.2.2 Class 2: standard missions with extensions

This class of missions will require extra costs to be assessed on a case-by-case basis (can result in several M€ extra) for the addition of HW and SW kits and complementary processes and/or verifications.

Missions	Orbit characteristics				
	Perigee altitude [km]	Apogee altitude [km]	Inclination [°]	Argument of perigee [°]	P/L configuration
A62 - GTO	[180, 400]	[35586, 35986]	[1, 8]	[175, 180]	Dual launch
A62 – MEO (incl. GTO+)	[2200, 35486]	[2200, 35486]	[0, 8]	Free (if close to 180)	Single launch & dual launch
A62 - SubGTO	[180, 400]	[5000, 35586]	[1, 8]	[175, 180]	Single launch & dual launch
A62 & A64 - Escape	Orbits with eccentricity higher than 1 with Vinf. [km/s]: [1, 6] Declination [°]: [-5, 5]			Free	Single launch
A64 - L2	Free	[900000, 1550000]	Free	Free	Single launch

Table 12: Class 2 A64 L2 and Earth escape plus A62 orbits.

A64 in single launch

Mission	Orbit characteristics			
	Perigee altitude Zp [km]	Apogee altitude Za [km]	Inclination i [°]	Argument of perigee w_p [°]
GTO	[180, 400]	35786 ± X	[1, 8]	[175, 180]
GTO+	≥ 2200*	35486 ± X	[1, 8]	Free
MEO Equatorial	Zp ~ 7000	Za ~ 15000	[6, 7]	Free

A64 in single and dual launch

Mission	Orbit characteristics			
	Perigee altitude Zp [km]	Apogee altitude Za [km]	Inclination i [°]	Argument of perigee w_p [°]
MEO	[2200, 35486]	[2200, 35486]	0 or free	Free
LEO	Zp ~ 1000	Za ~ 1000	i ~20	Free

Table 13: Class 2 A64 Earth orbits.

3.3.2.3 Class 3: other missions

This class of missions may require significant extra cost to be assessed on a case-by-case basis (could be up to several tens of M€ extra) for delta qualification efforts.

A62 Configuration and A64 configuration in single and dual launch:

Mission	Orbit characteristics			
	Perigee altitude Zp [km]	Apogee altitude Za [km]	Inclination i [°]	Argument of perigee w_p [°]
LEO (A62 & A64) (including SSO, PEO)	[250, 1500]	[250, 1500]	[0, 102]	Free
Non Class 1 Transfer Galileo 2G (A62) 1 or 2 S/C 1 or 2 S/C	[250, 7000] [250, 7000]	[250, 7000] [250, 23686]	[54, 58]	Free
Direct Galileo 2G 1 or 2 S/C	22922 (circular)		[54, 58]	N/A
HEO (A62 & A64)	[180, 35486]	[36000, 1550000]	[1, 15]	Free
Escape (A62 & A64)	Orbits with eccentricity > 1 with Vinf. [km/s]: [1, 6] Declination [°]: [-35, 35]			Free

Table 14: Class 3 A62 and A64 orbits.

3.4 Summary of potential mission profiles using European launchers into direct transfer orbits

A summary of potential launchers and mission profiles based on the analyses developed above is given in Table 15. Table 16 provides the similar data for current launchers for reference. One can notice in particular the substantial performance improvement that is targeted with A62 in comparison to Soyuz.

Orbit	Vega-C	Ariane 62	Ariane 64
SSO (800 km)	~1505 kg	≥ 4.5 t	≥ 10 t
GTO	NA	≥ 5 t	≥ 10.5 t
Moon	NA	≥ Soyuz	≥ A5
L1/L2 transfer	NA	~ 3.5 t	≥ A5
Earth escape to L4/L5	NA	≥ Soyuz	≥ A5
Mars 300 km orbit (*)	NA	≥ Soyuz	≥ A5
Venus 300 km orbit (*)	NA	≥ Soyuz	≥ A5

Table 15: Summary of potential launchers and mission profiles. All performances indicated are TBC. The sign (*) indicates that insertion into Mars/Venus orbit and circularisation is to be performed by the S/C propulsion subsystem, not the launcher, as per Table 4 and Table 7.

Orbit	Vega	Soyuz	Ariane 5
SSO (800 km)	~ 1270 kg	~ 4410 kg	≥ 10 t
GTO	NA	3250 kg	≥ 9.5 t
Moon	NA	~ 2310 kg	7 t
L1/L2 transfer	NA	2160 kg	6.6 t
Earth escape to L4/L5	NA	2050-2150 kg	≥ 6.2 t
Mars 300 km orbit (*)	NA	~ 780 kg	~ 2.1 t
Venus 300 km orbit (*)	NA	~ 525 kg	~ 1.7 t

Table 16: Reference performance data for current launchers, for comparison. The sign (*) indicates that insertion into Mars/Venus orbit and circularisation is to be performed by the S/C propulsion subsystem, not the launcher, as per Table 4 and Table 7.

3.5 Additional capabilities with Solar Electric Propulsion

Beyond the reference orbits that can be directly reached with the European launchers (as illustrated in Table 15), additional mission profiles can be considered by making use of Solar Electric Propulsion systems. While this might not be required for Earth orbits, or even Mars or Venus missions (as these can already be achieved with launches into direct transfer orbits), this might be enabling for e.g. missions to NEO or main belt asteroids.

Heritage examples from past missions and missions under development are given in section 3.5.1, while example scenarios are given in section 3.5.2.

3.5.1 Heritage

	Mission profile	Electric propulsion system
SMART-1	A5 auxiliary payload with ASAP launched into GTO in 2003. 14 month transfer orbit from Earth to Moon, with several orbit raising manoeuvres, orbit altering via lunar resonances and weak stability boundary transfer via the Earth-Moon L1 point. Final Moon orbit was 300 km x 3000 km followed by Moon impact.	SNECMA PPS-1350 Hall Effect Thruster. Isp = 1640 s; 68 mN using 1190 W. More than 5000 hours of thrust with ~80 kg of Xenon consumed out of a 367 kg S/C. A ΔV_{el} of 3.9 km/s was delivered with electric propulsion.
DAWN [8] (NASA mission)	Delta II launch in 2007 with C3 = 3 km²/s². Thrust arcs to Mars followed by Mars gravity assist, and additional thrust arcs to Vesta (2.15 to 2.57 AU) and Ceres (2.55 AU to 2.99 AU). 11 km/s of ΔV was delivered post-launch.	Ion Propulsion System with 30 cm thrusters on 2-axis gimbals. Variable power throttling: - 92 mN with 2.6 kW - 19 mN with 0.5 kW - Isp between 1900 s and 3200 s ~2300 days of thrust with 450 kg of Xenon consumed out of a 1290 kg S/C.
BepiColombo	A5 launch in 2018 with C3 ≈ 12 km²/s². 15 month transfer orbit with 3 thrust arcs until 1st Venus flyby. Venus resonance orbit until 2nd Venus flyby / gravity assist to reduce the perihelion to near Mercury. 4 Mercury flybys with different resonance orbits and a 180 degree singular transfer to reduce the relative velocity to 1.5 km/s. 4 final thrust arcs to further reduce the relative velocity until weak capture by Mercury in 2024.	T6 ion thrusters (derived from GOCE T5 thrusters) on 2-axis gimbal. Isp = 4200 s. 290 mN maximum (2 thrusters at 145 mN in parallel) with ~10 kW. ~20,000 hours of thrust to use 580 kg of Xenon out of a 4100 kg S/C, producing a ΔV_{el} = 4.140 km/s.

Table 17: Some heritage missions with electric propulsion systems. Note that the Bepi Colombo mission profile described here is one option out of several potential scenarios with different transfer trajectories and launch dates.

Note that the PPS-1350 thrusters used on SMART-1 can now be replaced by the PPS-5000 (qualified, but lifetime test still on-going) with the specifications at various operating points detailed in Table 18.

Operating point	Input power	Thrust	Isp
High Isp	5 kW	230 mN	2350 s
High thrust	5 kW	271 mN	1730 s
High power	6 kW	325 mN	1760 s

Table 18: PPS-5000 operating points.

In comparison, the T6 ion thrusters used on BepiColombo offer a much higher Isp, which reduces the amount of propellant needed. On the other hand T6 offers a lower thrust ratio to input power, which results in a higher power generation capability needed on the S/C and longer thrust/cruise times.

As can be seen in Table 17, missions using SEP use highly complex / optimised trajectories and transfer strategies. Beyond the spirals or thrust arcs performed with SEP, optimisation is also done using gravity assist manoeuvres, orbit resonances, weak stability boundary transfers etc.

The use of SEP generally requires a dedicated optimisation of the mission profile and trajectory with regard to the target body, considering: the target ephemeris and launch date; the required C3 and the launcher performance; the possibility to use other bodies (e.g. Venus, Earth or Mars) for gravity assist manoeuvres or orbit perturbations via resonances which will significantly reduce the ΔV that needs to be imparted by the propulsion system. Therefore, the scenarios presented in section 3.5.2 are intended as indicative examples.

Mission profiles using SEP also differ from standard ballistic impulsive transfers in some aspects:

- The use of low thrust mandates a very low arrival velocity (close to 0 km/s) and a weak gravitational capture, if only SEP is available at arrival.
- The thrust level depends on the input power. For interplanetary transfers and assuming a solar powered spacecraft, the distance to the Sun must be taken into account.
- The transfer duration depends on the available thrust/mass ratio. Even assuming a constant thrust, the resulting S/C acceleration over time will not remain constant as the S/C mass reduces with the propellant being expelled.
- Comparison of mass budgets between chemical and electric propulsion is not straightforward and cannot be limited to comparing propellant and propulsion system mass figures. The mass required for powering the SEP (solar arrays; power control and distribution electronics) must be included for a sound comparison.

3.5.2 Example scenarios

3.5.2.1 Venus, Mars and the main asteroid belt

Table 19 and Table 20 show the ΔV requirements and the resulting thrust times and S/C masses for spiral transfers to Venus, Mars and the main asteroid belt. The underlying assumptions are:

- An A62 launch into a parabolic Earth escape orbit ($C3 \approx 0$ km²/s²). This results in an Earth trailing/heading orbit around the Sun, near circular at 1 AU with a velocity in the heliocentric reference frame ≈ 29.7 km/s.
- The A62 performance should be better than that of Soyuz (~2150 kg).
- The SEP spiral transfer is performed with the thrusters detailed in section 3.5.1 (PPS-5000 or 2xT6 as on Bepi Colombo).

	Spiral transfer ΔV_{elec} [km/s]	$\Delta V_{Hohmann}$ for comparison [km/s]
Venus	5.3	5.3
Mars	5.6	5.6
Main asteroid belt inner edge (~2.1 AU)	9.2	8.9
Main asteroid belt outer edge (~3.2 AU)	13.1	12.1

Table 19: Comparison of ΔV needed for spiral transfers with SEP vs Hohmann transfers. Inclination changes and gravity losses are not included.

	PPS-5000			2x T6 (as on Bepi Colombo)		
	Thrust time [days]	S/C dry mass [kg]	Xe needed [kg]	Thrust time [days]	S/C dry mass [kg]	Xe needed [kg]
Venus	450-500	~1550	~600	450-500	~1850	~300
Mars	500-600	~1500	~650	500-600	~1850	~300
Main asteroid belt inner edge (~2.1 AU)	800-900	~1200	~950	800-900	~1700	~450
Main asteroid belt outer edge (~3.2 AU)	1150-1250	~950	~1200	1150-1250	~1550	~600

Table 20: Resulting mission profiles for spiral transfers with SEP (2x T6 using 10 kW on the right vs PPS-5000 using 5 kW at high thrust operating point on the left: similar thrust levels result in similar transfer times, but a higher Isp results in less propellant mass for the 2x T6 option at the expense of more power).

For the Mars and Venus cases, an additional ΔV is also needed to spiral down to the final desired low altitude orbit. As an example, for the Mars case, this would take an additional ~6 months and increase the Xenon needed to ~900 kg to reach a 4-sol orbit. Alternatively, aerobraking can be used (as already discussed in section 3.2.2).

DAWN is a good illustrative example of a reference mission profile to the main asteroid belt (see Table 17).

3.5.2.2 NEO mission – MarcoPolo R example

Near Earth Objects are a very diverse class of solar system bodies in every respect, including the orbital elements. The sole criterion that defines a NEO is that its perihelion radius must be less than 1.3 AU. This includes objects with orbits that are very similar to that of the Earth (and therefore relatively easy to reach) as well as those with high aphelia and/or large inclination, that may be significantly more difficult to reach.

The MarcoPolo R CDF [11] mission analysis results are given here as a reference scenario for a mission to a NEO (including a sample capture on the asteroid and a return to Earth). The selected target was the Apollo-class binary asteroid 175706 / 1996 FG3. It has a 395 days period with a perihelion just below the Venus orbit (allowing Venus swing-bys to reduce the SEP ΔV) and a 2° inclination with respect to the ecliptic plane.

The T6 engine was selected with a power supply scaled to provide 3 kW at 1 AU with a thrust level of 82 mN. At 1.7 AU, this would decrease to 1 kW, the lower limit at which T6 is assumed to be operable. The minimum Isp during transfer was 3300 s. Three transfer options were calculated, these are shown in Table 21.

	sep01b	sep01c	sep04d
Max. wet mass [kg]	1584	1584	1696
SEP Delta-v [m/s]	4956	5899	6167
Xenon Consumption [kg]	188.5	219.3	243.3
RCS Delta-v [m/s]	160	145	130
Hydrazine consumption [kg]	96.5	87.1	77.1
Duration [y]	7.47	6.40	6.51
Final mass at Earth [kg]	1290.0	1268.8	1366.1
SEP On-Time [d]	1085	1446	1222
Min. Venus swing-by altitude [km]	300	300	3198
Min. Earth swing-by altitude [km]	9686	-	-

Table 21: MarcoPolo R mission summary for 3 SEP cases (from [11]).

Sep01b was the selected baseline (while sep04d was kept as a back-up option). Launch was to be achieved with Soyuz from Kourou in 2021 (2023 respectively) with a $C3 > 10\ km^2/s^2$. Two Venus swing-bys were included in the transfer to the asteroid, while one Earth swing-by was included on the return to Earth (respectively only one Venus swing-by on the return to Earth, none in the transfer to the asteroid). In addition to SEP, a mono-propellant chemical propulsion system was also included for navigation manoeuvres for each planetary encounter (gravity assists and Earth arrival) and nominal operations around the asteroid.

4 SYSTEM CONSIDERATIONS

4.1 Transfer durations, data rates and power considerations

The following table provides useful information including typical transfer durations, data downlink capabilities (more details in section 4.2) and power generation capabilities. These apply regardless of the launcher used.

<table>
<tr><th>Orbit</th><th>Typical transfer duration</th><th>Typical science TM data rates</th><th>Power</th></tr>
<tr><td>LEO</td><td rowspan="2">< 1 day</td><td rowspan="2">X band: 10 Mbps
S band: ~1 Mbps
(5 Mbps max)</td><td rowspan="5">@ 1 AU
Solar radiation: ~1300 W/m2
Cosine loss for 36° off-pointing: 80%
Cell efficiency: 28%
System losses: 15%
Cell packaging ratio: 70%
Ageing: 86% (@ 3.75%/year for 4 years)
~150 W/m2 at EoL</td></tr>
<tr><td>HEO</td></tr>
<tr><td>Moon</td><td>< 1 week (direct transfer)
~70 – 130 days (low energy transfer)</td><td>X band: ~150 Mbps</td></tr>
<tr><td>Sun Earth L1/L2</td><td>~1 month</td><td>X band: 5-10 Mbps
K band: 75 Mbps</td></tr>
<tr><td>Heading/trailing heliocentric orbit and Sun-Earth L4/L5</td><td>14 – 50 months
(in increments of 1 year)</td><td>Ka band: 150 kbps</td></tr>
<tr><td>Venus</td><td>100 – 180 days
(conjunction transfer)
350 – 450 days
(1.5 revolution transfer)</td><td>X band: 63 – 228 kbps
(superior vs. inferior conjunction)</td><td>Approximately 1.9 times the value at Earth
Higher temperatures may further reduce the solar cell efficiency.</td></tr>
<tr><td>Mars</td><td>9-11 months
(conjunction transfer)
21 – 26 months
(1.5 revolution transfer)</td><td>X band: 38 – 230 kbps
(superior vs. inferior conjunction)</td><td>Approximately 0.36 times the value at Earth</td></tr>
</table>

Table 22: Typical transfer durations, TM data rates and power generation for potential orbits.

For the specific case of the Sun-Earth L4/L5 Lagrange points and Earth trailing orbits, these orbits are achieved with an initial Earth escape manoeuvre into a hyperbolic trajectory, followed by a final insertion manoeuvre for L4/L5 points (breaking may also be needed for trailing orbits, depending on the requirements).

The L5 point is found to be less demanding in terms of ΔV 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 23. 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]	ΔV for escape from 300 km LEO [km/s]	Departure C3 [km²/s²]	Arrival manoeuvre [km/s]	Wet/dry mass ratio (mass before/after insertion manoeuvre)
14	3.292	2.016	1.419	1.58
26	3.227	0.582	0.763	1.28
38	3.213	0.272	0.521	1.18
50	3.207	0.157	0.396	1.14

Table 23: Approximate Sun-Earth L5 transfers. The performance can be found for each launcher with the C3 given in the 3rd column. The final mass injected in L5 can be found by dividing by the wet/dry mass ratio (assuming an Isp = 317 s) in the last column.

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 $C3 \geq 0\ km^2/s^2$.

4.2 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

Note that the above formula does not take into account limitations that may result from international regulation rules.

The typical data rates given in Table 22 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 RF output power, 3 to 15 m ground antenna, possible data rates ~ 20 to 200 Mbps but in practice limited by regulations on maximum bandwidth for Science Missions to ~10 Mbps.
- Euclid: L2 orbit, Ka band, 70 cm HGA, 50 W RF output power, 35 m ESTRACK ground antenna, data rate 75 Mbps.
- Mars/Venus-Express (similar numbers for ExoMars TGO): X-band, 1.6 m HGA (1.3 m for VEX), 65 W RF output power (70 W for VEX), 35 m ground antenna, data rates 38 – 230 kbps (63 – 228 kbps for VEX) at superior vs inferior

conjunction. Note: at inferior conjunction, the potential data rate achievable by Mars Express is much higher (see Figure 2) 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.

- Solar Orbiter: Ka band, 1.1 m HGA, 73 W output power, 35 m ground antenna (derived from BepiColombo), data rate 150 kbps at 1 AU.

Note that the use of S band is not recommended due to likely future regulation restrictions.

Figure 2 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.

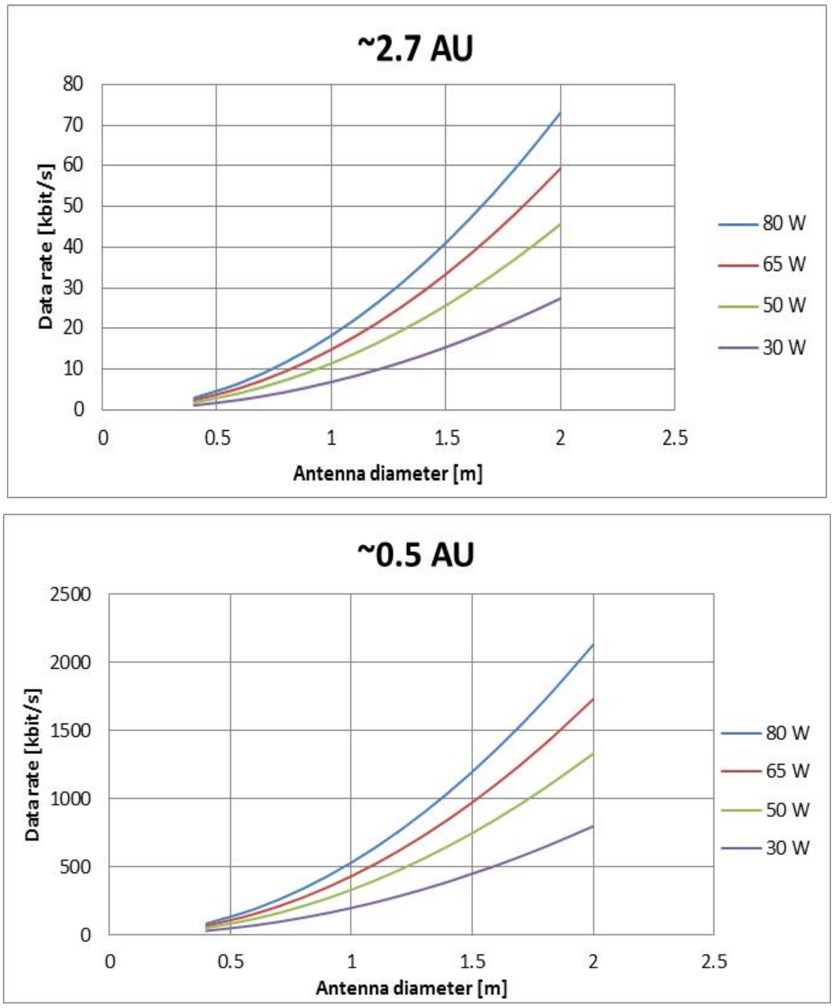

Figure 2: Data rate for a Mars mission, based on scaling the Mars Express data rate, as a function of antenna diameter and RF output power, for superior (top) and inferior (bottom) conjunctions.

4.3 Ground station characteristics

The reference for ground stations is the ESA ESTRACK network (details in [10]). This network is currently in evolution, with e.g. some 15 m stations being retired from service or handed over to third parties. Considering the M5 timescale, the following stations in the Core Network can be assumed:

Name	Antenna diameter [m]	Frequencies (Tx /Rx)	Note
Cebreros-1	35	X/X Ka	Includes capability in the 25.5-27 GHz band
Malargüe-1	35	X/X Ka	
New Norcia-1	35	S X/S X	Complemented by 4.5 m Acquisition Aid Antenna in X-band for LEOP
Kourou-1	15	S X/S X	Availability uncertain in 2029
Kiruna-1	15	S /S X	8025-8500 MHz RX X-band
Kiruna-2	13	S /S X	7600-8500 MHz RX X-band
Maspalomas-1	15	S X/S X	Availability uncertain in 2029

Table 24: ESTRACK Core Network ground stations available in the M5 timeframe.

Additionally, stations from the Augmented Network consisting of commercially-owned antennas can also be considered:

Name	Antenna diameter [m]	Frequencies (Tx / Rx)	Note
South Point (Hawaii)	13	S X/S X	
Santiago (Chile)	9	S/S	
Dongara (Australia)	13	S /S X	8000-8500 MHz RX X-band
Svalbard (Norway)	13	S /S X	7500-8500 MHz RX X-band
Troll (Antartica)	7.3	S X/S X	

Table 25: ESTRACK Augmented Network ground stations available in the M5 timeframe.

Finally, stations from the Cooperative Network consisting of antennas owned by Cooperating Space Agencies could also be considered (preferably as back-ups only or during critical operations such as LEOP). Their availability should be explicitly confirmed by the owning entity.

4.4 Space debris regulations

All ESA missions (see reference [6]) have to ensure that no additional orbital debris will contaminate the protected regions (in yellow in Figure 3). The practical consequence is the need to implement a propulsion subsystem for a S/C operating in the LEO or GEO protected regions, for either moving the S/C into graveyard orbits at its end of life, or ensuring its re-entry in the atmosphere within a specified maximum duration of 25 years.

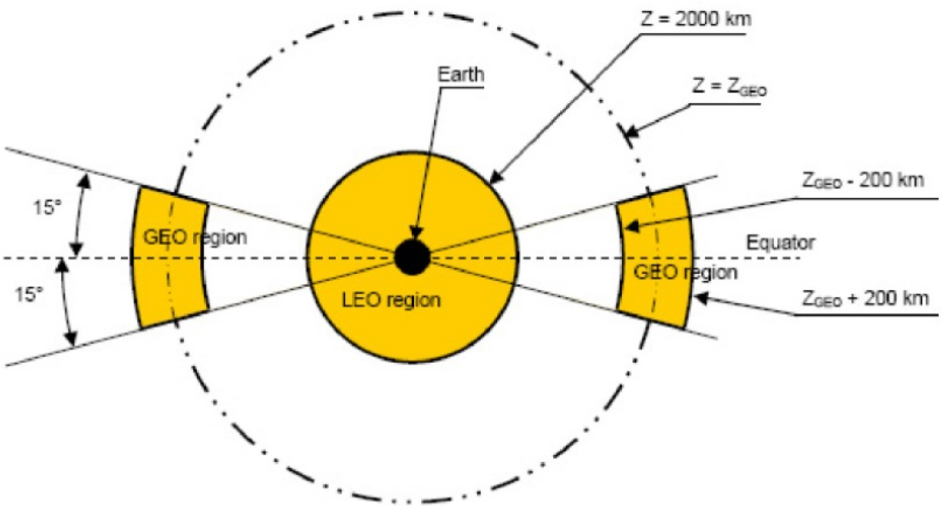

Figure 3: LEO and GEO protected regions [6].

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. 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 must 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.

5 SOME MISSION EXAMPLES AND HERITAGE

Table 26 and Table 27 provide some examples of European science missions that could be of relevance to the M5 Call.

Mission	Launch	Operational orbit	Launch mass [PL mass]	Power	Propulsion	Downlink	Pointing
Cluster (4 S/Cs)	2000 2 Soyuz in dual launch	HEO polar orbit 13000 x 119000 km^2	4800 kg [284 kg]	224 W Per S/C	650 kg bi-propellant	2-262 kbps S band	Spin rate: 15 rpm
MEX (same platform as VEX)	2003 Soyuz	Mars 330 x 10.530 km^2	1223 kg [116 kg]	650 W	457 kg bi-propellant	38-230 kbps X band	APE = 0.15°
Corot	2006 Soyuz	LEO 896 km i=90°	668 kg [300 kg]	530 W	90 m/s mono-propellant	722 kbps S band	APE = 0.5'' (telescope used as a FGS)
Gaia	2013 Soyuz	Sun-Earth L2	2030 kg [710 kg]	1910 W	237 kg mono-propellant + 59 kg cold gas	10 Mbps X band	Spin rate: 1 °/min Fine pointing with payload in the loop
Lisa Pathfinder	2015 Vega with propulsion module	Sun-Earth L1	1910 kg, iIncl. 214 kg prop. module dry mass + 1250 kg propellant. [178 kg]	650 W	Propulsion module + bi-propellant + cold gas	52 kbps X band	APE = 0.05°

Table 26: Examples of European science missions, past or in operation.

Mission	Launch	Operational orbit	Launch mass [PL mass]	Power	Propulsion	Downlink	Pointing
CHEOPS	2017 shared launch passenger compatible with Soyuz, Vega, etc.	LEO SSO, dusk-dawn (650-800 km)	280 kg [60 kg]	200 W	17 kg mono-propellant	1.2 Gbit/day S band	APE = 4'' (telescope used as a FGS)
Solar Orbiter	2018 Atlas V	165 days Sun orbit (0.28 AU perihelion)	1800 kg [190 kg]	1012 W (at 1.47 AU)	228 kg bi-propellant	150 bkps (at 1 AU) X band	APE = 3.5'
Euclid	2020 Soyuz	Sun-Earth L2	2160 kg [295 kg]	2500 W	125 kg mono-propellant	75 Mbps X/K band	APE = 2.5''

Table 27: Examples of European science missions under development.

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.

Technology Readiness Level	Milestone achieved for the element	Work achievement (documented)
		Critical function test plan. 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 28: Summary definition of the ISO TRL levels, taken from [7] (contains guidelines for the interpretation and implementation of the TRL requirements defined in [9] based on ISO 16290).

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. 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, also referred to as the hyperbolic departure or escape velocity ($V_\infty = 0$ 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 4 and Figure 5. 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.

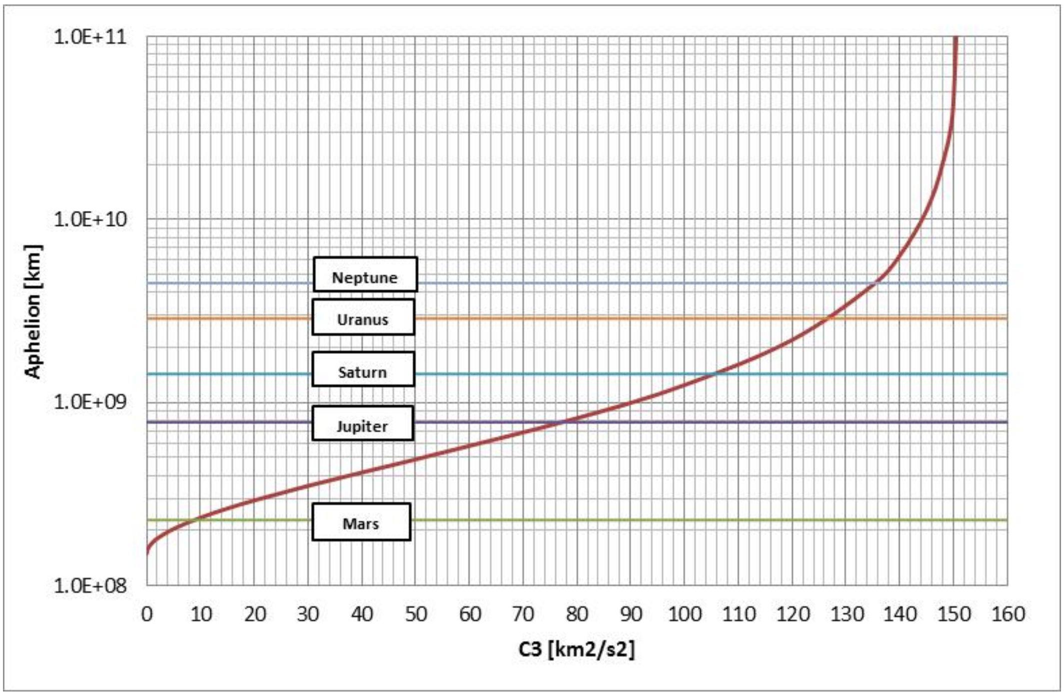

Figure 4: C3 values required to reach the external planets, assuming a direct Hohmann transfer. The escape velocity from Earth is applied at an altitude of 300 km, the red curve indicates the aphelion of the Hohmann transfer in the heliospheric reference frame. The semi-major axes of the orbit of the external planets are indicated.

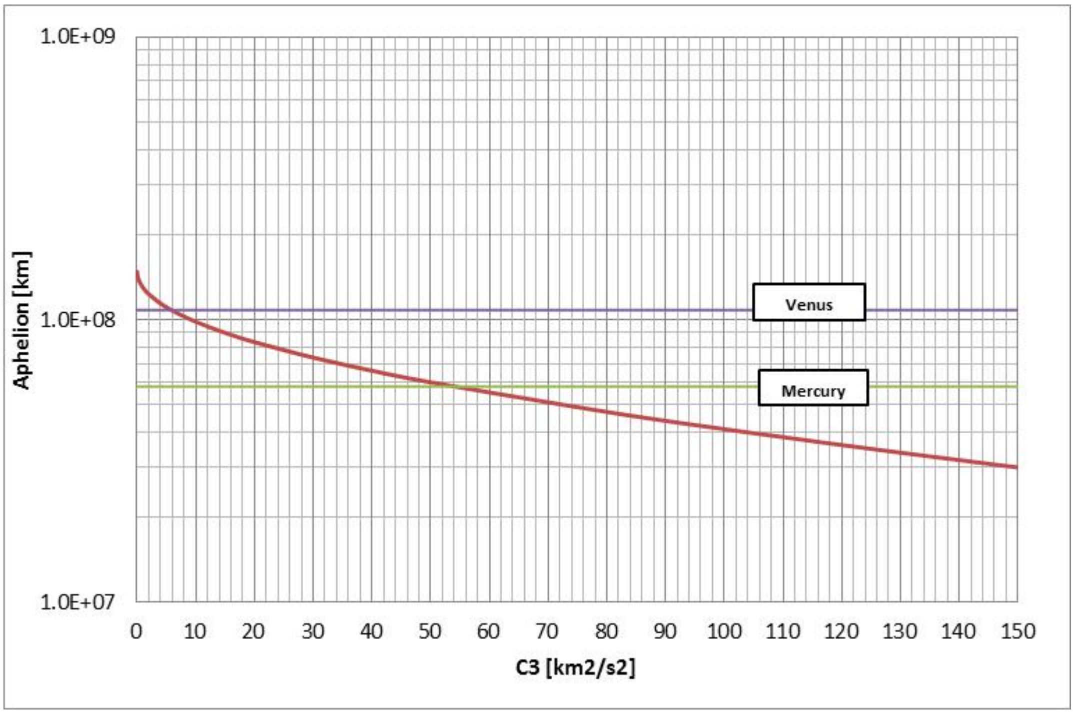

Figure 5: Same as Figure 4 for the inner planets.

Specific C3 values for 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.

APPENDIX C – A6 FAIRING AND ADAPTER

Possible fairing options for A6 are illustrated in Figure 6 and Figure 7. Note that the dimensions included refer to the geometric envelope rather than the dynamic envelope which is still TBD.

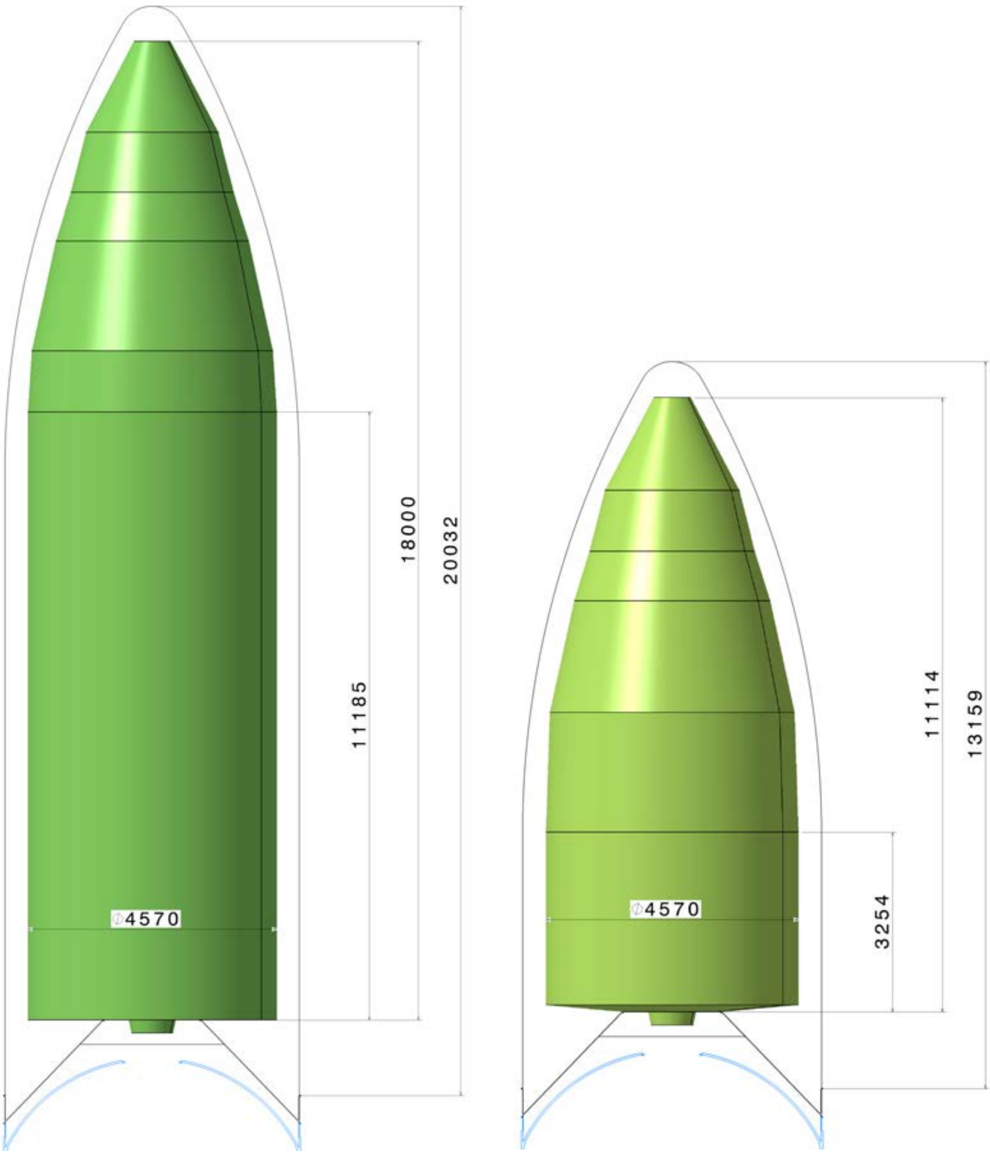

Figure 6: Possible A6 fairings, with long version on the left and short version on the right. A single baseline will be selected within 2016.

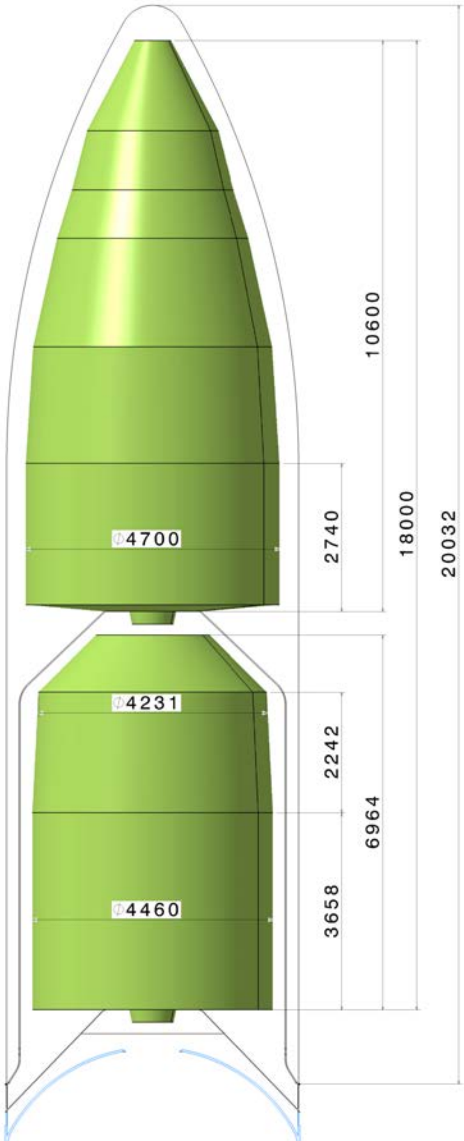

Figure 7: A6 dual launch configuration with SYLDA under long fairing option.

Concerning the A6 – S/C launch adapter, a single LV adapter with an upper diameter of 1780 mm is currently baselined. Bespoke S/C adapters can be added on top to increase or decrease this interface diameter to the S/C, but these will not be considered as a standard service and will induce additional costs.

APPENDIX D – STANDARD BALLISTIC TRANSFERS TO VENUS AND MARS

Ballistic transfers from the Earth to Venus repeat over an 8 year cycle: the opportunities in 2029 will be very similar to those in 2021, those in 2031 to those in 2023 etc. Therefore, the 8 year cycle of direct ballistic transfers from 2021 to 2028 is sufficient to assess the transfers also for the years that follow.

This is illustrated in Figure 8 and summarised in Table 29. The upper plot shows the departure velocity (or V_∞ as explained in Appendix B). The C3 can be derived to assess the launcher performance for all launch dates and transfer durations (where $C_3 = {V_\infty}^2$). The middle plot shows the arrival velocity, i.e. the S/C velocity relative to Venus upon arrival, and therefore gives an indication of the ΔV required for orbit insertion (the exact ΔV required will depend on the targeted orbit). The lower plot is simply the sum of both departure and arrival plots.

For the Earth to Mars transfer case, there are significant differences between the individual launch opportunities due to the eccentricity and inclination of the Mars orbit (see Figure 9)

Note that the C3 values used in Table 3 and Table 7 (7.5 km^2/s^2 for Venus and 10.5 km^2/s^2 for Mars) correspond to average values within the Figures below, therefore real values will vary depending on the exact launch dates.

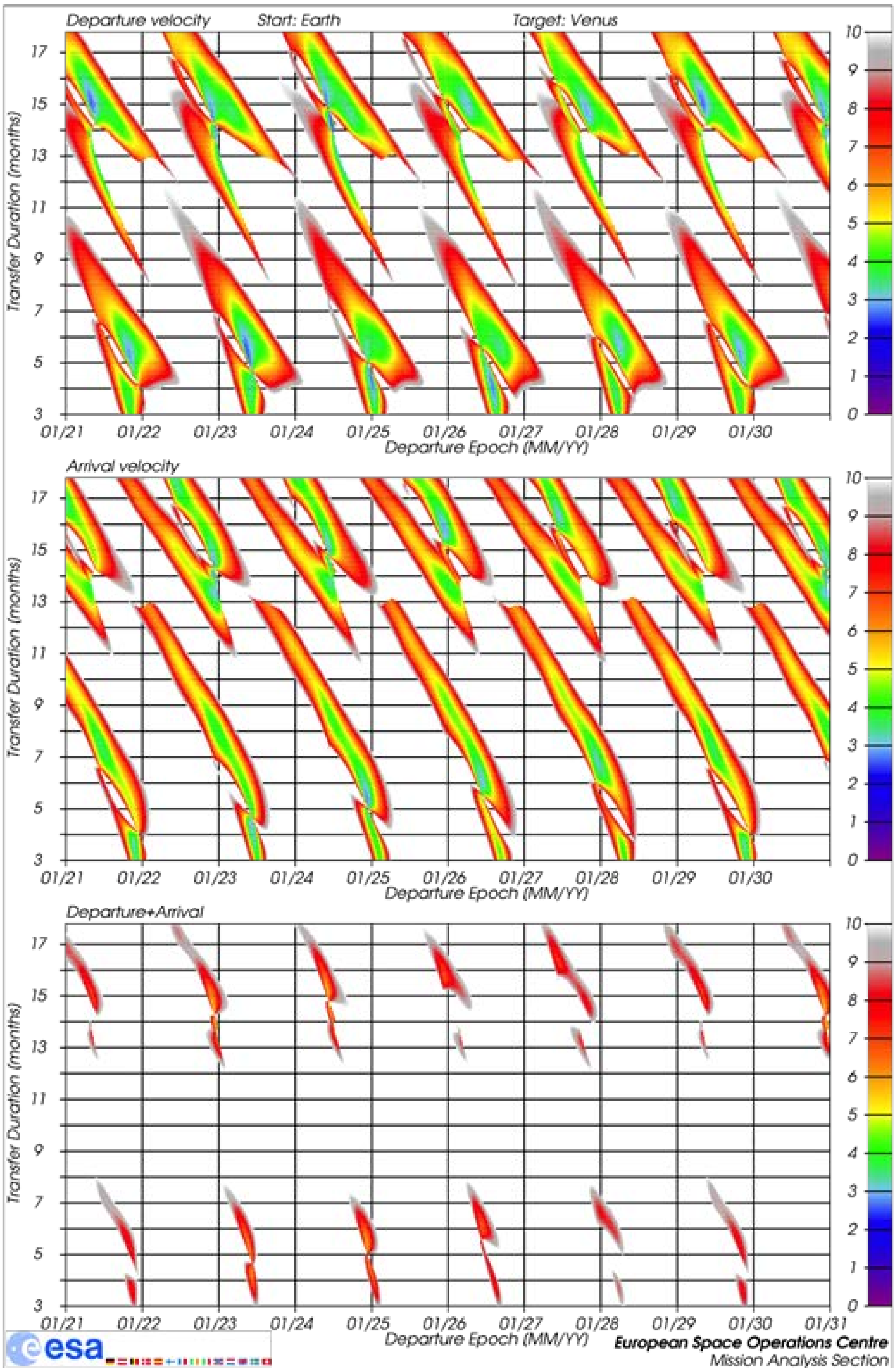

Figure 8: Earth-Venus transfers, 2021-2030. Note that the upper half of each figure refers to long transfers (1.5 heliocentric orbital revolution), as opposed to direct conjunction transfers in the lower halves. The colour scale on the right is the velocity in [km/s].

These Venus transfers are summarised in Table 29.

Launch Opportunity	Hyp. Escape Velocity [km/s]	Hyp. Arrival Velocity [km/s]	Transfer Duration [d]
2021 T1	3.613	5.416	106
2021 T2	2.803	4.764	160
2023 T1	3.408	3.662	120
2023 T2	2.482	3.904	159
2024 T1	2.653	3.789	135
2024 T2	4.088	3.784	186
2026 T1	2.685	4.931	124
2026 T2	3.552	7.591	166
2028 T1	3.343	6.016	113
2028 T2	3.062	6.130	171

Table 29: Complete cycle of ballistic, direct Earth-Venus transfers.

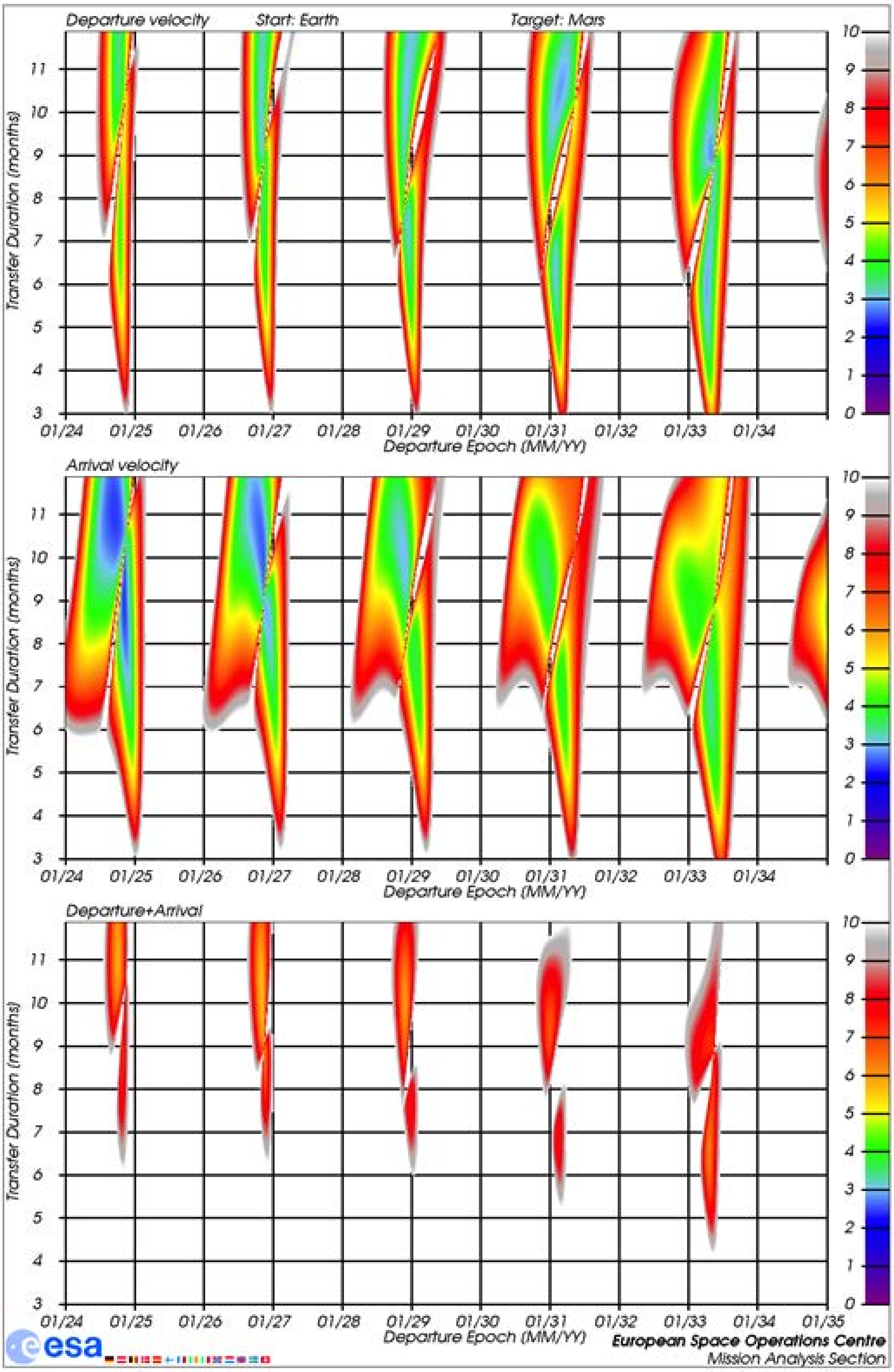

Figure 9: Earth-Mars transfers, 2024-2034.