CALL FOR A MEDIUM-SIZE AND A FAST MISSION OPPORTUNITY IN ESA'S SCIENCE PROGRAMME - TECHNICAL ANNEX

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1. INTRODUCTION

This document provides technical and programmatic information useful for preparing the mission proposals in answer to the Call for medium (M-class) and fast (F-class) mission opportunity in the ESA Science Programme (also called M and F in this document). It also includes constraints and assumptions to be considered for the mission proposal.

Furthermore, the proposers can access to information related to previous ESA missions at http://sci.esa.int/home/51459-missions/.

2. DEFINITIONS AND ACRONYMS

The acronyms and abbreviated terms are defined in Annex A.

3. REFERENCE AND NORMATIVE DOCUMENTS

3.1. REFERENCE DOCUMENTS

RD[1] Ariane 6 User's Manual, Issue 2.0, Feb. 2021, www.arianespace.com link

RD[2] Ariane 6 User's Manual for Multi-Launch Service (MLS), issue 0.0, July 2021, www.arianespace.com link

RD[3] Vega C User's Manual, Issue 0.0, May 2018, www.arianespace.com link

RD[4] SSMS Vega-C User's Manual, issue 1.0, September 2020, www.arianespace.com link

RD[5] ECSS-E-HB-11A, Technology readiness level (TRL) guidelines, Mar. 2017, www.ecss.nl

RD[6] ECSS-E-HB-60-10A, Control performance guidelines, Dec. 2010, www.ecss.nl

RD[7] ESSB-HB-E-003, ESA pointing error engineering handbook, Jul. 2011, www.ecss.nl link

RD[8] ESSB-HB-U-002, ESA Space Debris Mitigation Compliance Verification Guidelines, issue 2.0m 14 Feb 2023 www.ecss.nl link

RD[9] M8 F3 Call briefing and Q&A link

3.2. NORMATIVE DOCUMENTS

ND[1] ECSS-E-AS-11C, Definition of the Technology Readiness Levels (TRLs) and their criteria of assessment, Oct. 2014, www.ecss.nl link

ND[2] ECSS-E-ST-50-05C Rev. 2, Radio Frequency and Modulation, www.ecss.nl link

ND[3] ECSS-U-ST-20C, Planetary protection, www.ecss.nl link

4. SUMMARY OF THE CALL BOUNDARY CONDITIONS

4.1. BOUNDARY CONDITIONS FOR THE M MISSION

Element	Request	Comments or Guidelines
ESA CaC	≤ 670 M€ (2025 e.c.)	Includes all elements to be funded by ESA, including the launch services. Excludes Member State and international partner contributions.
Science objectives and science payload	The science objectives of this mission are open.	The science payload shall be defined in relation with the targeted science objectives. The core science objectives and the proposed concept shall be sufficiently robust for enabling technical convergence by following a design-to-cost approach during the phases 0/A.
Launch	around 2041	Mission-dependent
Launcher	The M mission will nominally be launched either by Ariane 62/64 or by the Vega-C/E launcher.	Other schemes may be considered subject to providing evidence of their feasibility. Non-European launchers to be procured by ESA are excluded.
Spacecraft dry mass	≤1500 kg	Recommended upper limit not to exceed in view of the cost target
Platform and Science Payload TRL	TRL 5-6 by mission adoption	ISO TRL scale, see Appendix B. The spacecraft (platform and payload) can feature new developments but shall rely on substantial heritage. In practice, TRL ≥ 5 shall be targeted for the mission selection (end of Phase A). Therefore, some delta-developments are possible provided they can be safely completed within ~3 years. It is recommended that the mission does not contain any element with TRL < 4 at the time of the proposal. In case some elements are at TRL 4 and are critically needed for achieving the mission science goals, the proposer shall present a credible path for reaching TRL ≥ 5 by the mission selection. In such case, the proposer is also invited to identify (if possible) a back-up scenario at TRL ≥ 5 with reduced mission performance. The payload definition level must reach SRR level (detailed design including interface requirements) by the mission adoption, within ~ 5 years. ESA is ready to support the instrument detailed design and pre-developments during the phases 0/A for securing the payload development schedule. Following the mission selection at the end of

		the phase A, the Member States will fund the payload pre-development activities in the phase B1 then, following the mission adoption, the flight models production. Proposers are invited to submit in the proposal their views for the payload development plan, including pre-development needs, by distinguishing (as far as possible at this stage) the activities needed in phases 0/A and those to be achieved in phase B1. The role, responsibilities, and heritage of the major payload providers must be defined in the proposal.
International collaboration	Can be envisaged, provided a clear support and commitment from the international partner are available.	The M mission must be ESA-led. The firm commitment is expected for the study phase, but not at the call (step 1). European alternatives shall be proposed as risk mitigation.
Spacecraft and science operations	The spacecraft operations are nominally under ESA responsibility with contributions from the Member States or partners to the science ground segment. Nominal duration of science operations typically < 3-4 years	Other collaboration schemes may be considered subject to providing evidence of their feasibility. The contribution to the Science Ground Segment shall be detailed. For a complex Science Ground Segment (e.g. astrophysics mission such as Euclid) involving several institutes or industry, and requiring funding from several Member States, the proposed organisation scheme shall be developed by including the expected contribution from ESA. The nominal duration of science operations does not include the cruise phase, nor the disposal (as applicable).

Table 1: Boundary Conditions for the M-class mission

4.2. BOUNDARY CONDITIONS FOR THE F MISSION

Element	Request	Comments or Guidelines
ESA CaC	≤ 205 M€ (2025 e.c.)	Includes all elements to be funded by ESA, including the launch services. Excludes Member State and international partner contributions.
Science objectives and science payload	The science objectives of this mission are open.	The science payload shall be defined in relation with the targeted science objectives. The core science objectives and the proposed concept shall be sufficiently robust for enabling rapid technical convergence by following a design-to-cost approach in the preparation phase.
Launch	around 2034	Mission-dependent
Launcher	The F mission will nominally be launched with the Vega-C/E launcher or small European launcher (as available).	Other schemes may be considered subject to providing evidence of their feasibility. Non-European launchers to be procured by ESA are excluded.
Spacecraft dry mass	Typically ~500 kg	Including payload and all margins
Spacecraft wet mass	Shall be indicated by the proposer	The actual launch mass constraint will depend on the target orbit and the associated Vega-C/E performance. However, a mass constraint is also introduced to limit the spacecraft cost in line with the CaC constraints. The spacecraft wet mass encompasses the platform(s) with its propulsion subsystem(s), the propellant needed for the mission (including disposal, when applicable), and the scientific instrumentation. The launcher adapter is excluded, but any spacecraft dispenser, if needed, shall be included.
Overall science payload mass	Typically ≤ 70 kg	The payload upper mass limit is a recommended guideline with due regard to the overall cost and schedule constraints. The actual allowable payload mass can be lower depending on the mission profile (see sections 5 and 6). The proposers shall keep in mind the need to ensure a fast and reliable payload development and qualification schedule, typically 3 years starting from the mission adoption.

Platform TRL	TRL $\geq$ 6 at mission proposal	ISO scale, see Appendix B. Proposers are invited to build their mission proposal by relying on existing platform capabilities, with minimum modifications. The platform is nominally procured by ESA As a rule, the platform equipment shall be at TRL $\geq$ 7 (space qualified for the mission needs and available) before the mission adoption. TRL 6 is nominally required at the time of the mission proposal (TRL 5 acceptable) since the Technology Readiness may drive the schedule and will be one important element of the decision process.
Science Payload TRL	TRL $\geq$ 6 by the mission adoption	The credibility of the payload development and qualification schedule will be an important selection criterion. The proposed payload can be a new development but must rely on significant heritage and fully available technologies. Limited delta-verifications and pre-developments can be envisaged during the definition phase. The payload definition level must reach PDR status before the mission adoption, within ~3 years, and ESA is ready to support the instrument detailed design and pre-developments during this phase for securing the payload development schedule. Proposers are invited to submit in the proposal their views for the payload development plan, including pre-development needs. The role, responsibilities, and heritage of the payload providers must be defined in the proposal.
International collaboration	Can be envisaged, provided a clear support and commitment from the international partner are available.	The F mission must be ESA-led.
Spacecraft operations	The spacecraft operations are nominally under ESA responsibility with contributions from the Member States or partners to the science ground segment. Nominal duration of science operations typically < 2 years	Other collaboration schemes may be considered subject to providing evidence of their feasibility. The contribution to the Science Ground Segment shall be detailed by including the expected contribution from ESA. The nominal duration of science operations does not include the cruise phase, nor the disposal (as applicable).

Table 2: Boundary Conditions for the M-class mission

5. MISSION CONCEPT DEFINITION

5.1. LAUNCH VEHICLES

The proposed launch vehicle shall be one of the European launcher family, such as Ariane 6 and Vega-C/E. The following sections provide an overview of the launch mass capability for various destinations using Ariane 62/64 or Vega-C. The mass figures are indicative, and 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.

The proposed mission can feature a single or multiple spacecraft, subject to compatibility with the programmatic conditions. For multiple spacecraft, the proposers are invited to consider the additional structure mass that will be needed to hold and release the assembly (dual launch structure or dispenser).

The selected launch vehicle may have a delivery capability that is exceeding the proposed space segment mass. In principle, sharing the launcher with some co-passenger could then be considered for either the F or M mission for lowering the launcher costs. However, the feasibility and cost of a shared launch scenario need to be carefully assessed on a case-by-case basis, by considering the co-passenger mass and orbit, the dual launch structure mass and the exact launch release scenario, for example the use of the launcher upper stage to bring the spacecraft from the passenger orbit to the operational orbit. For the purpose of the proposal, unless the proposer has a clear view of a specific shared launch scheme and can provide evidence of its feasibility, it is recommended to baseline the full launcher price.

5.1.1. Ariane 6

There are two versions of Ariane 6: Ariane 62 and Ariane 64, depending on the number of boosters employed. They are described in RD[1]. Given the ESA CaC constraint for this Call, the use of Ariane 64 is unlikely, except in some specific cases (e.g. for high Delta-v missions) for which the use of Ariane 64 could be more cost efficient: for example, when the spacecraft can be reduced in complexity and cost by exploiting the higher performance of A64 and therefore compensating the cost difference between Ariane 62 and 64.

5.1.1.1. Ariane 6 performance

The following table provides the indicative Ariane 62/64 performance data for selected orbits that can be of interest for science missions.

Launch orbit	Orbital parameters	Performance (kg)	
		A62	A64
SSO	600 x 600 km, i = 97.4 deg	6800	-
LEO Polar	900 x 900 km i = 90.0 deg	6600	-
Sub-GTO	250 x 22,500 km, i = 6.0 deg	6600	-
GTO	250 x 35,768 km, i = 6.0 deg, ωp =178.0 deg	5100	11700
HEO	250 x 100,000 km, i = 6.5 deg, ωp =170.0 deg	3900	-
Lunar transfer	200 X 400,000 km, i=6.0 deg	4200	9400
SEL2 transfer	165 X 1,500,000 km, i=6.0 deg	3500	8000
Earth Escape	V_∞ = 2.5 km/s, δ = 0.0 deg	2600	6900

Table 3: Ariane 6 performance – indicative values; specific performance shall be provided by Arianespace.

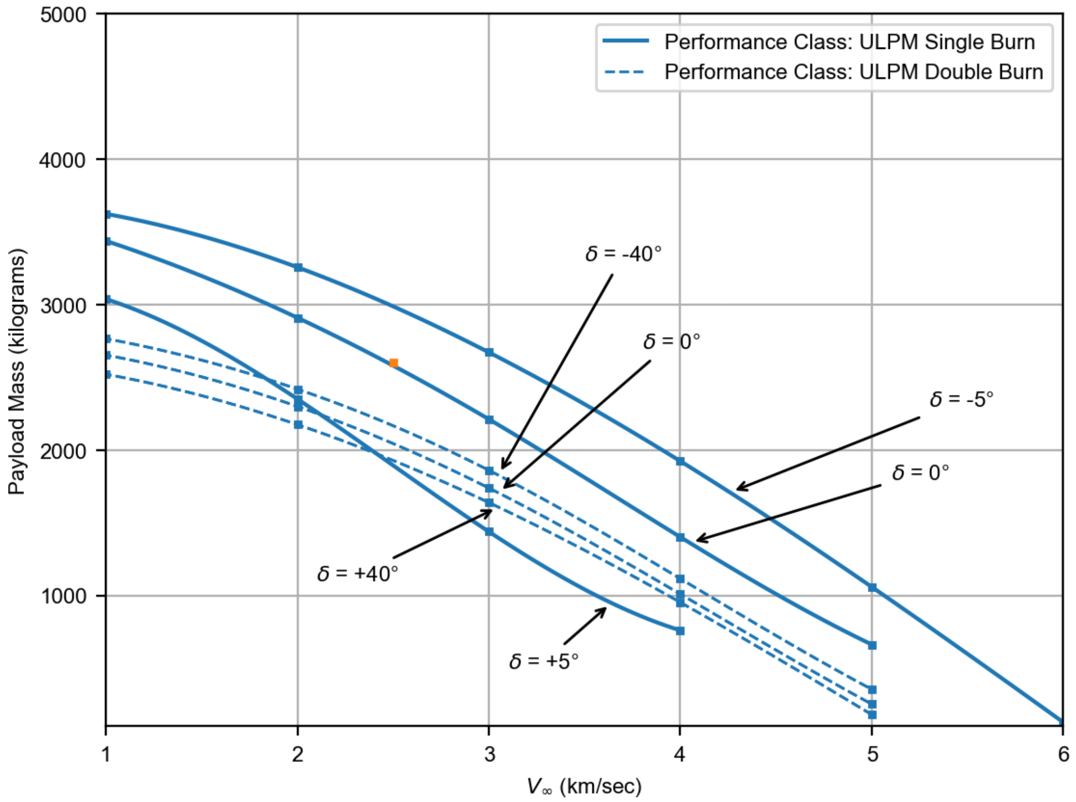
Note:
The performance figures given in this section are expressed in term of payload mass including:

- The spacecraft separated mass;
- The dual launch carrying structure if any (system for auxiliary payload or dual launch system);
- The adapter (PAF) or dispenser.

Off-the-shelf adapters, with separation interface diameter of 937 mm, 1194 mm and 1666 mm are available as described in RD[1]. The associated masses should be considered by the proposers:

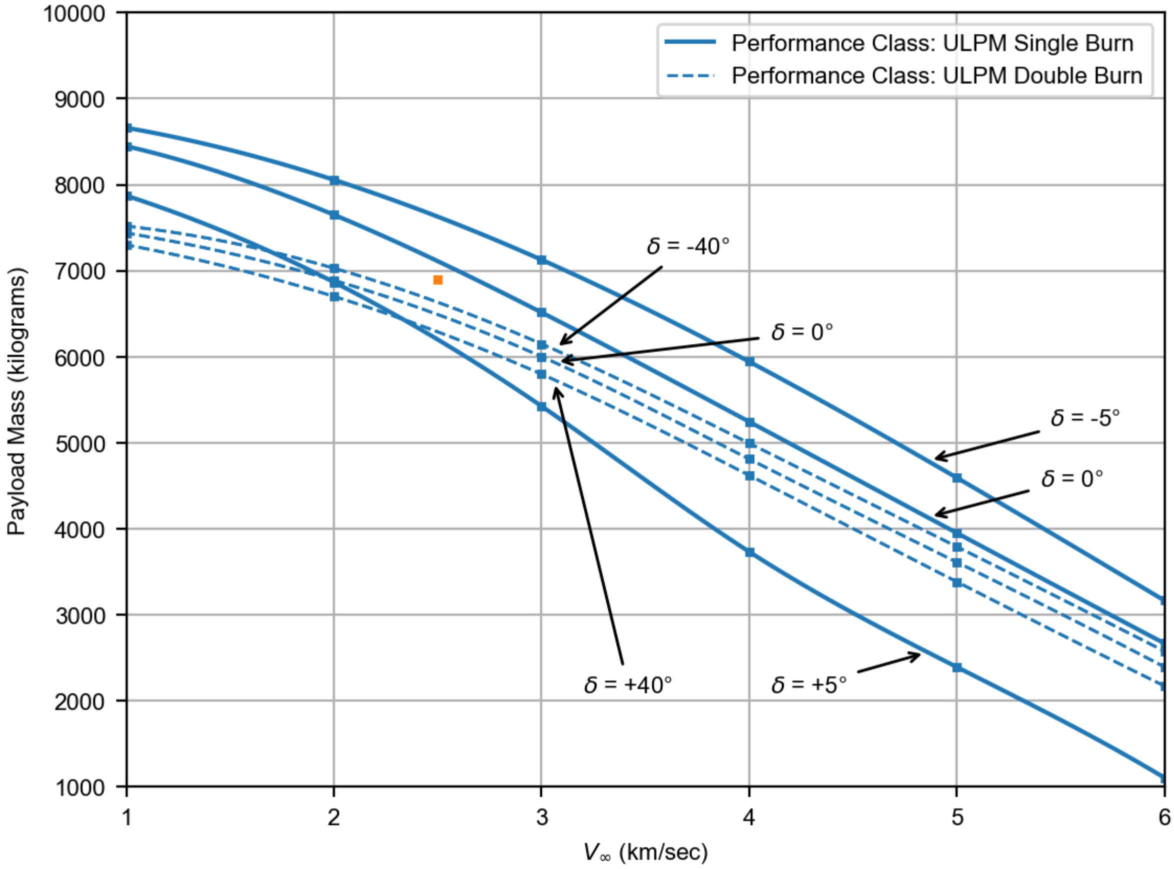
- Interface diameter 937 mm: 120.0 kg
- Interface diameter 1194 mm: 130.0 kg
- Interface diameter 1666 mm: 160.0 kg

For Earth escape missions, the following plots describe the performance data for a range of hyperbolic excess velocity, V_∞. and for a set of the declination of the launch azimuth, DLA, which corresponds to the V_∞ direction relative to the inertial Earth mean equator coordinate system. ULPM designates the launcher upper stage. The performance data is based upon preliminary numerical simulations, only be considered for the purpose of planning and preparation of proposals.



V_∞ (km/s)	Payload Mass (kilograms)							
	ULPM Single Burn, DLA (°)			ULPM Double Burn, DLA (°)				
	-5.0	0.0	+5.0	-40.0	-20.0	0.0	+20.0	+40.0
1.0	3624	3438	3038	2766	2715	2653	2571	2521
2.0	3257	2910	2351	2419	2345	2300	2228	2175
3.0	2672	2212	1439	1861	1793	1739	1678	1639
4.0	1924	1402	760	1116	1065	1010	982	953
5.0	1059	663	NA	355	303	253	225	181
6.0	129	NA	NA	NA	NA	NA	NA	NA

Figure 1: Representative Performance for Earth Escape Missions, Ariane 62



V∞ (km/s)	Payload Mass (kilograms)							
	ULPM Single Burn, DLA (°)			ULPM Double Burn, DLA (°)				
	-5.0	0.0	+5.0	-40.0	-20.0	0.0	+20.0	+40.0
1.0	8661	8447	7872	7518	7485	7438	7367	7300
2.0	8056	7650	6867	7029	6963	6888	6800	6701
3.0	7127	6515	5426	6149	6079	6002	5915	5798
4.0	5943	5245	3734	4995	4911	4812	4711	4622
5.0	4594	3952	2394	3793	3710	3613	3520	3385
6.0	3167	2669	1108	2578	2487	2395	2305	2174

Figure 2: Representative Performance for Earth Escape Missions, Ariane 64

5.1.1.2. Ariane 6 launch configuration and mechanical interfaces

There are several launch configurations possible with Ariane 6: single launch, dual launch or launch in Multi-Launch Service (MLS) configuration.

Available payload volumes and standard mechanical interfaces for all configurations are described in [RD1] and [RD2]. However, it shall be noted that the MLS Users Manual RD[3] is currently being updated with the introduction of a larger “HUB” structure in place of the one described in the current MLS UM, allowing to accommodate up to 4 payloads of 1 ton class on its side. A preliminary scheme is displayed below:

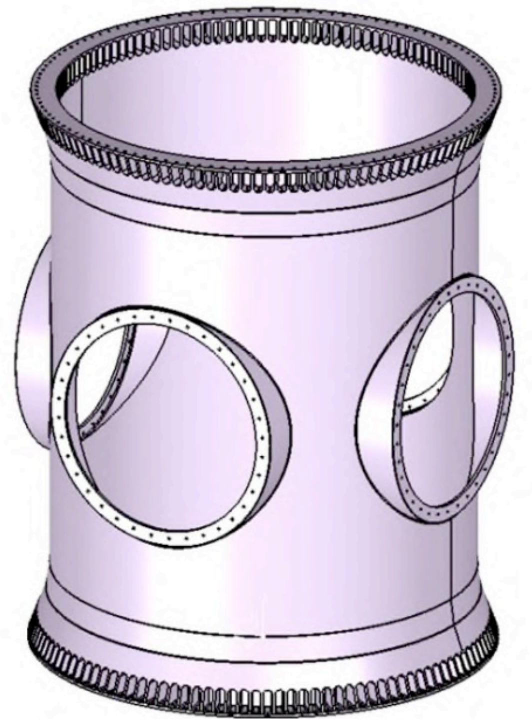


Figure 3: HUB structure under Development

The dual launch configuration is standard with A64. It is not yet standard with A62, but it can be considered as in the case of Ariel and Comet Interceptor. The Dual Launch Structure typical mass is around 880 kg. The standard and short versions as described in [RD1] can be envisaged. An allocation for the cost of this additional structure must be included in the mission CaC.

The Multi Launch Service (MLS) configuration is relevant to small/mini satellites with a maximum mass of 1200 kg. Available options are described in detail in RD[1].

5.1.2. Vega-C / Vega-E

Vega-C has been conceived for circular, or near-circular Low-Earth Orbits but it can be also used in a variety of other orbits. The Vega-E evolution should be compatible with the M development schedule and is expected to provide a slightly better performance than Vega-C.

5.1.2.1. Vega-C performance

Table 4 provides the Vega-C performance for other selected orbits that can be of interest for science missions

Launch orbit	Orbital parameters	Performance (kg)	
		Vega-C	Vega-E
LEO Polar	500 x 500 km, i = 88.0 deg	2250	TBD
LEO Intermediate inclination	700 x 700 km, i = 70.0 deg	2350	
LEO equatorial	600 x 600 km, i = 5.4 deg	2980	
LEO high equatorial	7400 x 7,400 km i = 15.0 deg	600	
HEO equatorial	250 x 5,700 km, i = 6.0 deg	1600	
	150 x 20,000 km i = 6.5 deg	600	

Table 4: Vega-C estimated performance for selected destinations

The following figure (taken from RD[3]) provides the performance for a LEO circular Sun Synchronous Orbit (SSO).

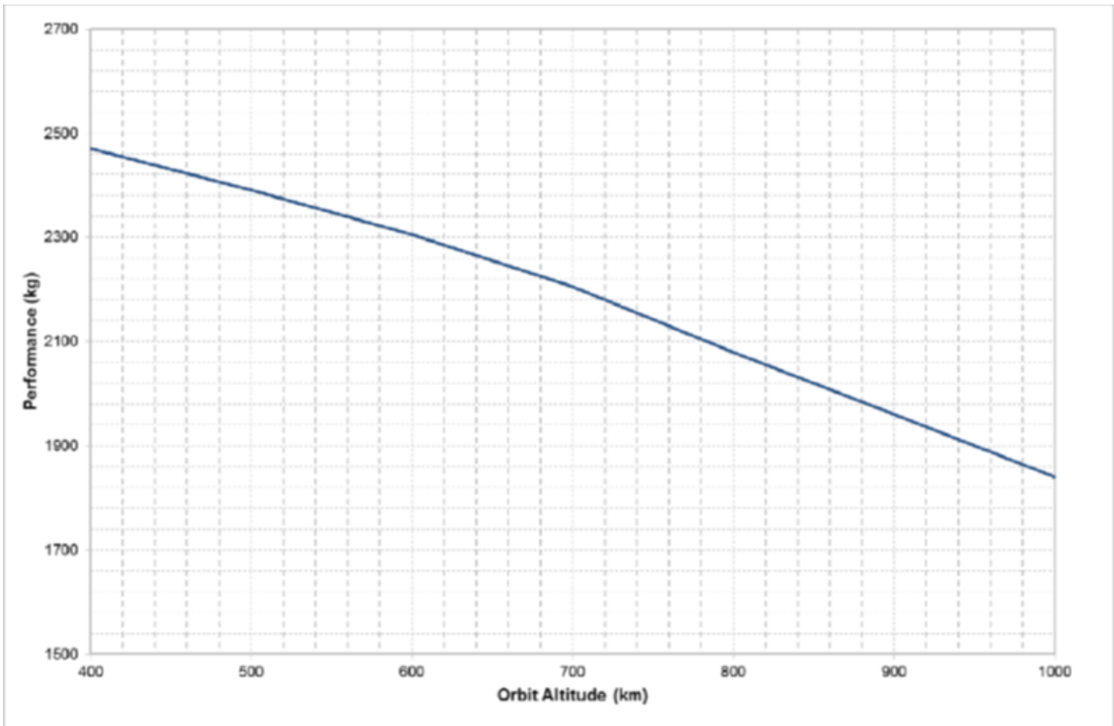


Figure 4: Vega-C performance in SSO

Higher energy orbits can be achieved by using an additional propulsion module/kick stage or through own spacecraft propulsion (either electrical or chemical).

The following figure shows an example of quasi-equatorial apogee altitude vs payload mass (=delivered SC mass after using the PM and excluding the PM dry mass) achievable by using a bi-propellant propulsion module and starting from a HEO quasi equatorial launch orbit (200 x 1550 km, 6 deg inclination).

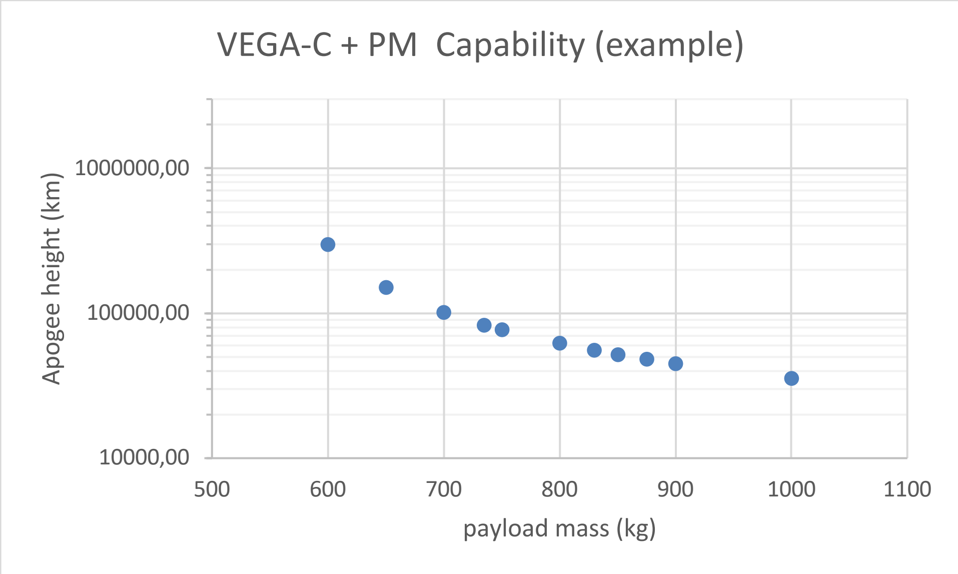


Figure 5: Vega-C capability with Bi-Propellant Propulsion Module (example)

For a specific mission case, depending on the desired spacecraft mass into the final orbit, an optimisation needs to be run to define the Vega-C insertion orbit and the propulsion module design.

The availability of the propulsion stage shall be verified, and its cost included in the mission CaC.

5.1.2.2. Launch configuration and mechanical interfaces

There are several launch configurations possible with Vega-C: single launch, dual launch and launch in Small Spacecraft Mission Service (SSMS) configuration. Available payload volumes and mechanical interfaces are detailed in RD[3] and RD[4]. The SSMS configuration is suitable for the launch of nano, micro and mini satellites.

5.1.3. European small launcher

Currently several small launchers are under development in Europe. Their use for F-missions might be an option but will need careful consideration concerning their timely availability, performance and reliability.

Due to the ongoing rapid development and changes in this sector, only a non-exhaustive and non-exclusive list of providers is disclosed here. Consolidated performance mass figures cannot be listed here, but the links given below provide indicative information from the launcher provider.

Launcher - Company	Country	Foreseen maiden flight	Website
Maïa Space	France	2025	Maïa Space
Latitude - Zephir	France	2025	Zephire
RFA ONE	Germany	2025	RFA ONE
Isar Aerospace – Spectrum	Germany	2025	Spectrum
HyImpulse – SL1	Germany	2026	SL1
PLD Space – Miura 5	Spain	2026	Miura 5
Orbex – Prime	UK	2025	Prime
Skyrora – Skyrora XL	UK	2025	Skyrora XL

Table 5: Non-exhaustive list of European mini launchers.

5.1.4. Other launch vehicles

Launch services from an international partner may be considered if there is a documented intent from the partner to provide it at own cost.
Launch from China shall not be considered, as compliance to Export Control Regulations (see section 8.4) cannot be guaranteed.

5.2. MISSION AND SPACECRAFT

The following sections provide some information, data and considerations that can be useful for a preliminary sizing of the mission.

5.2.1. Transfer to the final orbit

Whenever the mission operational orbit is different from the launch orbit, a transfer scenario needs to be defined. This may include propulsive manoeuvres (either by chemical or electric propulsion), orbit resonances and weak stability boundary transfers.

Mission profiles using Solar Electric Propulsion (SEP) differ from standard ballistic impulsive transfers for a few important aspects:

- The thrust level depends on the input power and is generally much lower than for chemical propulsion.
- The transfer duration depends on the available thrust/mass ratio. Even assuming a constant thrust, the resulting SC acceleration over time will not remain constant as the SC mass reduces while the propellant is 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) is often significant and it must be included for a sound comparison.

5.2.1.1. Examples of orbit transfer for the F-mission assuming a Vega-C launch

Table 6 provides key parameters for a set of potential orbits that can be reasonably considered. The list of targets is not meant to be exhaustive, the intention being to provide an order of magnitude of the achievable performance. Other orbits can be considered but the proposal needs to show evidence that they meet the call boundary conditions.

All figures should be viewed as preliminary and need further consolidation through detailed studies, which could lead to reductions through the design-to-cost approach.
Similarly, it shall be stressed that Table 6 considers only potential feasibility of the different mission concepts from a mass perspective without addressing the programmatic constraints applicable to the F mission. Therefore, the cost and schedule compatibility will have to be assessed on a case-by-case basis.

Potential Target Orbit for F Mission	Delivered F Mass at Target Orbit	Indicative F Science Payload Mass	SC Delta-V to Reach Target Orbit	Expected Transfer time	Indicative Nominal SCI Operations	Remarks
SEL1/SEL2	up to ~ 600kg	up to ~ 70kg	~ (3-4) km/s	~ (2.5-3.5) years	~ < 2 years	The figures of this example correspond to SEP propulsion transfer. Usage of Chemical Propulsion for the transfer will require downgrade of the spacecraft class (and payload mass).
SEL4/SEL5	up to ~ 590kg	up to ~ 70kg	~ (3-4) km/s	~ (3-4) years	~ < 2 years	The figures of this example correspond to SEP propulsion transfer. Usage of Chemical Propulsion for the transfer will require downgrade of the spacecraft class (and payload mass).
Equatorial/Inclined Earth Orbit (circular or HEO)	up to ~ 700kg	up to ~ 70kg	variable (depending on mission design)	variable (depending on mission design and propulsion thrust)	~ < 2 years	Considering the VEGA-C User Manual performance and mission cost constraints, a maximum spacecraft wet mass of ~ 700 kg is being assumed. Indicative SC Delta-V would be in the range of (270-470) m/s for Chemical Propulsion, and (3-4) km/s for Electric Propulsion, without the propellant needed for disposal (when applicable).
Equatorial Earth Orbit up to 13000 km Apogee	up to ~ 700kg	up to ~ 70kg	provided by launcher	direct injection	~ < 2 years	According to VEGA-C User Manual performance, assuming a working point of ~700 kg wet mass spacecraft (due to cost envelope constraints).
Near Earth Orbit	up to ~ 600kg	up to ~ 70kg	~ (3-4) km/s	~ (3-4) years	~ < 2 years	The mission design would follow the transfer to SEL2 (or SEL1) plus an additional push to reach the selected NEO, for a total Delta-V up to around 4.2 km/s.
Trans-GEO (from circular)	up to ~ 600 kg	up to ~ 70kg	2.35 km/s	~7-8 months	until failure	Satellite injection in a circular 7400 km altitude orbit (to cross radiation belts fast during the transfer) and SEP raising using a Hall thruster. Target orbit is in graveyard region ~300 km beyond the geostationary orbit. Inclination reduction to 0 and station keeping will not be required once target orbit is reached. Later deorbiting also not required
Moon Orbit (from circular)	up to ~ 600kg	up to ~ 70kg	~4-4.5	~3-4 years	~ < 2 years	Assuming SEP orbit raising, and satellite injection in a circular 7000 km altitude orbit (to avoid radiation belts during the transfer).

Table 6: Potential destinations for the F mission using SEP

The following guidelines and explanations are provided for the proposers:

- Given the current Vega-C performances, in many of the potential indicative mission profiles, a circular injection orbit of 3000 x 3000 km altitude is being used, as baseline, to avoid increased radiation dose, due to transition through the Van Allen belts, which could jeopardize the health of the science payload during the long electric propulsion orbit transfer.
- Knowing the constraints, the maximum spacecraft Delta-V which could be identified in the preliminary feasibility study is around 4 km/s, by using Solar Electric Propulsion technology with a specific impulse in the order of 1660s.
- Such Delta-V limitation rules out F-missions with final destinations beyond the Sun-Earth Lagrange Points (i.e. much farther than 1 AU distance from the Sun).
- In any case, depending on the selected orbit for science operations, the mission analysis design will have to be optimised, considering the most efficient transfer approach for the mission (i.e. gravity assists, weak-boundaries transfers).
- In order to comply with the reduced operational cost envelope, the following mandatory drivers are defined:
    - The need for ground commanding of manoeuvres during the science operations phase should be minimised as much as possible. The science operations concept should be defined accordingly.
    - The trade-off between ground-commanding vs on-board autonomy should favour on-board autonomy whenever mission safety could be assured.

5.2.1.2. Examples of orbit transfer for the M-mission assuming an Ariane 6 launch

Some examples of electric propulsion transfers are provided here below for different interplanetary targets and with two different electric propulsion engine technologies (Hall Effect Thruster, Ion Engine).

	Hall effect thrusters ~300 mN			**Ion engines ~2x 150 mN**		
	Thrust time [days]	S/C dry mass [kg]	Xe needed [kg]	Thrust time [days]	SC 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 7: Examples of Electric Propulsion transfer for the M mission assuming Ariane 62 launch

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 less demanding in terms of Delta-V 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). It 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 propellant demands for reaching L4/L5 can be lowered by increasing the transfer time, as illustrated in the table below. 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 ($C3 = V_\infty^2$) and the Delta-V 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 8: 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 SEL5 can be found by using the wet/dry mass ratio and the arrival delta-V values (assuming an Isp = 317 s).

For the case of drifting, Earth leading/trailing orbits, no arrival manoeuvre is required. The only Delta-V to consider is the one required to reach Earth escape velocity, with a C3 ≥ 0 km^2/s^2.

5.2.2. Mass and Power resources: F-mission case

The preliminary cost/schedule analysis has indicated that the spacecraft design should make use of an existing platform as much as possible, in order to limit the development and operational costs. No or minimal changes to the existing platforms will considerably help in meeting the strict cost and schedule constraints for the F mission. The proposals shall identify the required changes (if any) to be made to an existing platform and show evidence of their potential compatibility with the Call boundary conditions.

Typical mass ranges to be considered for the F-mission (see also section 4.2) are as follows:

- Science payload mass < 70 kg
- Total spacecraft dry mass below ~ 500 kg (including the science payload)

Large Delta-Vs will require the use of Solar Electric Propulsion (SEP) as explained in section 5.2.1.1. Some existing platforms are designed for providing a peak power up to 2 KW at 1 AU and could be used for the F mission by accommodating the SEP.

5.2.3. Mass and Power resources: M-mission case

Considering the nature and cost envelope of this mission, it is recommended to design the mission with a spacecraft dry mass below 1500 kg. Proposals with SC dry mass higher than this value will still be evaluated, provided there is evidence they could fit with the Call boundary conditions (in particular, the CaC).
Should the SC mass be significantly below the allowable launch mass, ESA may consider a dual launch with some future F-mission (not the one targeted by this Call, for schedule reasons) or a shared launch.

No specific limitation on power is defined (provided the CaC limit is maintained). Medium class missions typically have an overall power demand below 2000 W.

5.2.4. Communications

ESA science missions shall comply with ITU frequency allocation requirements (see ND[2]). ITU assigns frequency bands for the different space telecommunication services. Science missions fall into the Space Research (SR) service category, which is split in two sub-categories depending on the SC distance to Earth in the operational orbit:

1. Near Earth or Category A for SC altitude above Earth surface < 2 Mkm (this includes Sun-Earth L1 and L2 missions, for instance),
2. Deep Space SR (DS) or Category B for SC altitude above Earth surface ≥ 2 Mkm.

The frequency allocations are reported in the following table extracted from ND[2]. The table also reports the max bandwidth that can be allocated to a single mission for specific bands. Actual allocation will be, in practice, a fraction of that value and the actual useable data rate performance will depend not only on theoretically link budget and allowed bandwidth by regulations, but also strongly by limits given by the selected communication hardware (e.g. transponder, TWTA power, etc.). Limitation coupled with constraints on the ground station visibility and the onboard memory, puts a limit on the maximum science data volume that can be transmitted to ground in a given time (e.g. a week).

Types of mission	Link	Band	Frequencies (MHz)	Max bandwidth allowed	Examples and achievable data rates
LEO, HEO, SEL1/SEL2, Lunar	Uplink	S	2 025 – 2 110	Not applicable	CHEOPS RX
		X	7 190 – 7 235	Not applicable	Gaia RX
		Ka	40 000 – 40 500	Not applicable	Not used yet. Equipment and Ground infrastructure not yet available
LEO, HEO, SEL1/SEL2, Lunar	Downlink	S	2 200 – 2 290	6 MHz	CHEOPS Tx (0.6 Mbps)
		X	8 450 – 8 500	10 MHz	Gaia TX (up to 10 Mbps)
		K	25 500 – 27 000	No limitation	Euclid TX (70 Mbps), PLATO TX (40 Mbps)
Earth trailing, SEL4/SEL5, Planetary, Solar	Uplink	S	2 110 – 2 120	New assignments in this band are formally discouraged	
		X	7 145 – 7 190	Not applicable	Solar Orbiter RX
		Ka	34 200 – 34 700	Not applicable	EnVision (for radio science)
Earth trailing, SEL4/SEL5, Planetary, Solar	Downlink	S	2 290 – 2 300	New assignments in this band are formally discouraged	
		X	8 400 – 8 450	Function of symbol rate (see ND[2])	Mars Express TX (up to 230 kbps), Solar Orbiter TX (up to 600 kbps)
		Ka	31 800 – 32 300	No limitation	BepiColombo TX, JUICE TX (up to 50 kbps) EnVision TX
		Ka	37 000 – 38 000	No limitation	Not used yet. Equipment and Ground infrastructure not yet available

Table 9: Allowed frequency bands and associated bandwidths

As an example, typical achievable X-band data rates as a function of distance to Earth, SC High Gain Antenna diameter and RF power output (assuming ESTRACK 35m ground antennas) are indicated in the following figure.

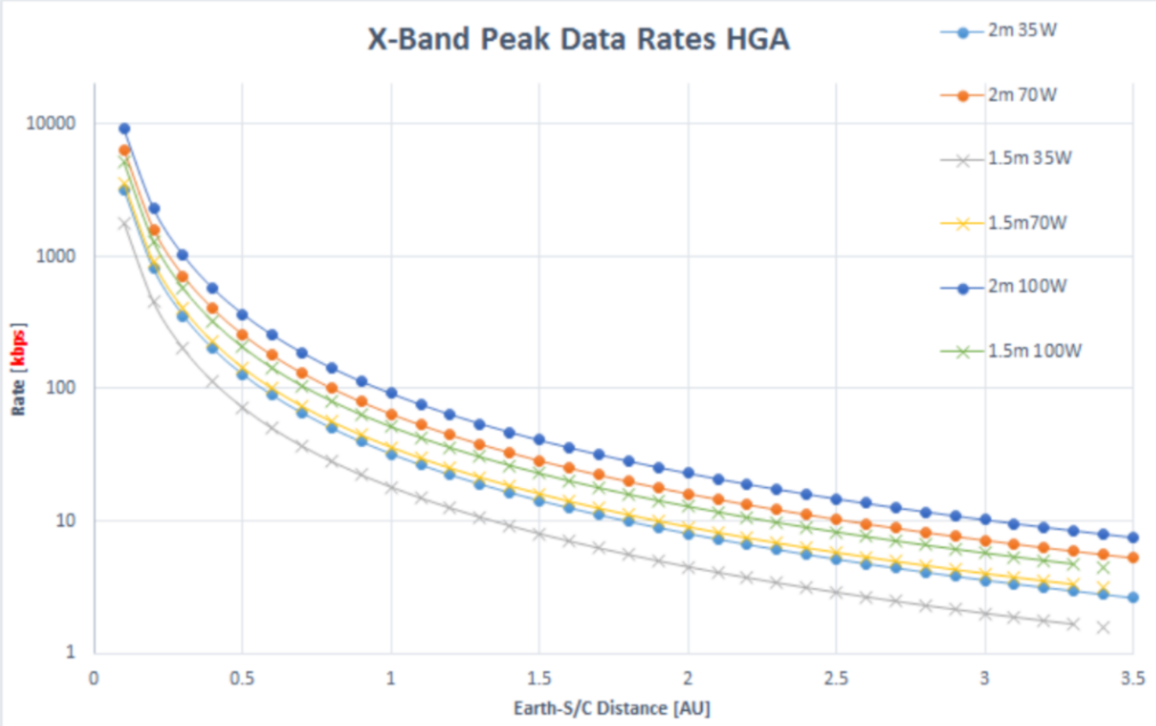


Figure 6: Typical achievable downlink data rate in X-band vs Earth-SC distance, antenna size and communication power. The actual data rate performance depends not only on the theoretical link budget calculation but also on allowed bandwidth by regulations and limits due to hardware (e.g. transponder, TWTA power), in particular for high data rates and bandwidth limitations (proximity to Earth = see table above).

5.2.5. Spacecraft Budgets and Margins

This section summarizes the minimum margins to be considered by the proposers at system level.

Parameter	Margin	Comments
SC dry mass	25%	The nominal total spacecraft dry mass (excluding the 25% system margin) must be evaluated by including the maturity margins (If no detailed design exists 20% design maturity margin is recommended; for a recurring element it can 5-10%). at equipment or subsystem level. The total spacecraft dry mass shall include the total platform dry mass plus the allocated payload mass. **The payload level margin included in the allocated payload mass shall be clearly identified.** Propellant mass shall be calculated with the total dry mass at launch including system margin.
Delta Velocity	5%	The total delta-velocity capability of the spacecraft shall include this system level margin.
Power	30%	The total power demand of the spacecraft shall include this system level power margin. The payload level power margin shall be clearly identified.
Pointing	100%	The pointing accuracy, knowledge and stability error predictions shall include this system level margin.
Data Rate	50%	The calculation of the total payload data rate shall include this system level margin
Data Volume	50%	The calculation of the total payload data volume shall include this system level margin.
Communication Link	3 dB	The communication link budget for all mission phases shall be calculated with a minimum nominal margin of 3 dB.

Heat Rejection for cryogenic systems	20-100%	The calculated heat rejection capacity of the cryogenic systems which are operating at temperature below 100K shall include the following system level margin: - 20% for systems operating between 50K and 100K - 50% for systems operating below 50K - 100% for systems operating below 2K

Table 10: Recommended System Contingencies and Margins

5.2.6. Pointing Requirements

Science measurement requirements in most cases imply requirements on spacecraft pointing accuracy and knowledge. Those may have significant impact on the spacecraft design and cost (e.g. need of a micro-propulsion system or high performance AOCS sensors)

Pointing requirements are specified through pointing error indices introduced in the ESA pointing error engineering handbook [RD[7]].

A simplified description of the most common indexes is (see also Figure 7):

Absolute Pointing Error (APE): difference between a wished direction and the actual one at any given time. This is a measure of the spacecraft capability of pointing accurately. In many cases, the APE represents the difference between the line of sight of an instrument and the required direction of the target. As such, it may be derived e.g. from the need to keep the light coming from the target within the focal plane surface or inside a slit, in some cases of spectroscopy.

Relative Pointing Error (RPE): difference between the instantaneous direction and the average one in a given time interval. This is a measure of the pointing stability of the spacecraft over a relevant observation time. For imaging systems, such error causes image blurring, i.e. its maximum allowed value may be derived from the required spatial resolution of an instrument.

Absolute Knowledge Error (AKE): difference between the actual direction and the measured one at a given time. This defines the required performance for AOCS sensors onboard. Attitude knowledge is always part of the pointing error. However, it may be the driving requirement in case attitude is reconstructed on ground by e.g. image postprocessing.

Relative Knowledge Error (RKE): the equivalent index of the AKE but applied to the RPE.

All pointing errors are random functions of time, requiring a statistical specification often using Gaussian distribution. The limit value is often expressed as a 2-σ value, i.e. corresponding to 95% of the cases.

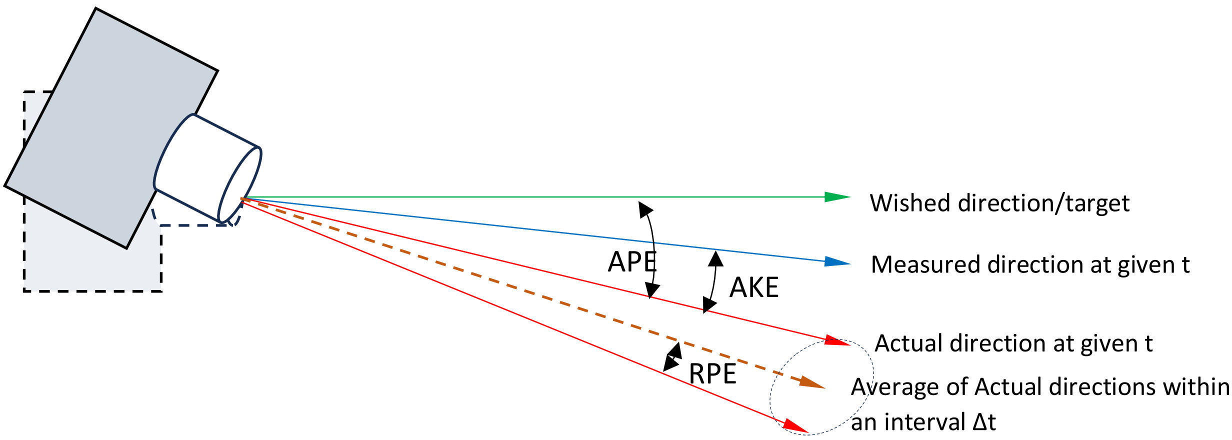


Figure 7: A simplified description of the most common indexes

The proposer is expected to express and justify the critical pointing requirements for the proposed mission, i.e. those driving the science measurement performance and possibly the spacecraft cost. In some cases, the use of on-board AOCS sensor may not be sufficient to comply with pointing requirements (in particular, RKE) if they are very tight. Then, the use of the instrument measurements in the AOCS loop may improve considerably the pointing performance. This has been implemented in many astronomy missions either by directly using the science instrument data or by adding a dedicated Fine Guidance Sensor (FGS) in a science instrument focal plane.

The table below provides (for information) some pointing requirement formulations for an instrument line of sight (LoS) as a directional half cone angle.

ARIEL			
Parameter	**LoS (arcsec)**	**Δt**	**Probability (%)**
Absolute Pointing Error (APE) in coarse pointing mode	10.0		99.7
Absolute Pointing Error (APE) in fine pointing mode	1.0	-	99.7
Relative Pointing Error (RPE) in fine pointing mode	0.23	10 h	99.7
XMM			
Absolute Pointing Error (APE)	30.0	-	95.0
Relative Pointing Error (RPE)	6.0	2 min	95.0
Absolute Knowledge Error (AKE)	10.0	-	99.7
PLATO			
Absolute Pointing Error (APE)	270	-	99.7
Relative Pointing Error (RPE)	0.8	2.5 s	95.0

Table 11: Examples of Pointing Requirements Formulation

5.3. GROUND STATIONS

The reference for ground stations is the ESA ESTRACK network. This network is in constant evolution, with some 15 metre stations being retired from service or handed over to third parties. Considering the mission timescale, the following stations can be assumed:

Ground Stations	LEOP	Transfer Cruise	Critical Phases	Science Phase
Cebreros (X/XKa) 35 m		X	X	X
Malargüe (XKa/XKa) 35 m		X	X	X
New Norcia-1 (X/XKa) 35 m		X	X	X
New Norcia-3 (X/XKa and X/XK) 35 m		X	X	X
New Norcia-2 (X/SX) 4.5 m	X			
Kourou (SX/SX) 15 m	X	X	X	X
Kiruna-1 (S/SX) 15 m	X	X	X	X
Kiruna-2 (S/SX) 13 m	X	X	X	X

Table 12: Available ESTRACK Core Network Ground Stations

	CEBREROS-1 (X / X Ka)	MALARGÜE-1 (X / X Ka)	NEW NORCIA-1 (S X / S X)	NEW NORCIA-3 (X / X Ka)	KIRUNA-1 (S / S X)	KIRUNA-2 (S / S X)	KOUROU-1 (S X / S X)
TERMINAL	**CEB1**	**MLG1**	**NNO1**	**NNO1**	**KIR1**	**KIR2**	**KRU1**
Longitude	04 deg 22' 03.18" W	69 deg 23' 53.51" W	116 deg 11' 29.40" E	116 deg 11' 29.40" E	20 deg 57' 51.57" E	20 deg 58' 00.77" E	52 deg 48' 16.79" W
Latitude	40 deg 27' 09.68" N	35 deg 46' 33.63" S	31 deg 02' 53.61" S	31 deg 02' 53.61" S	67 deg 51' 25.66" N	67 deg 51' 30.34" N	5 deg 15' 05.18" N
Altitude [m]	794.095	1550.00	252.2558	252.2558	402.1724	400.6815	-14.6709
Antenna Diameter [m]	35	35	35	35	15	13	15
FUNCTIONALITIES					Common backend for KIR1/2		
Doppler	YES	YES	YES	YES	YES	YES	YES
Delta-DOR	YES	YES	YES	YES	NO	NO	YES
UPLINK							
S-band TX band [MHz]	N/A	N/A	2025-2120	N/A	2025-2120	2025-2120	2025-2120
S-band EIRP [dBm]	N/A	N/A	127.8 (SHPA) 112.1 (SSPA)	N/A	101	99	111.2 (SHPA) 104.7 (SLPA)
X-band TX band [MHz]	7145 - 7235	7145 - 7235	7145 - 7235	7145 - 7235	N/A	N/A	7145-7235
X-band EIRP [dBm]	138 (XHPA) 128 (XLPA) 122 (XSPA)	138 (XHPA) 128 (XLPA) 122 (XSPA)	138 (XHPA) 128 (XLPA)	138 (XHPA) 128 (XLPA) 122 (XSPA)	N/A	N/A	112.8
Ka-band TX band [MHz]	N/A	N/A	N/A	N/A	N/A	N/A	N/A
Ka-band EIRP [dBm]	N/A	N/A	N/A	N/A	N/A	N/A	N/A
DOWNLINK							
S-band RX band [MHz]	N/A	N/A	2200-2300	2200-2300	2200-2300	2200-2300	2200-2300
S-band G/T [dB/K]	N/A	N/A	37.5	37.5	27.7 (at 5 deg El.)	21.4 (at 5 deg El.)	29.1
X-band RX band [MHz]	8400 - 8500	8400 - 8500	8400 - 8500	8400 – 8500	8025-8500	7600-8500	8025-8500
X-band G/T [dB/K]	50.8 (at 10 deg El.)	50.8 (at 10 deg El.)	50.1	50.8 (at 10 deg El.)	36.9 (at 5 deg El.)	35.6. (at 5 deg El.)	41
Ka-band RX band [MHz]	31800 - 32300	31800 - 32300	N/A	31800 - 32300	N/A	N/A	N/A
Ka-band G/T [dB/K]	55.8 (at 10 deg El.)	55.7 (at 10 deg El.)	N/A	55.7 (at 10 deg El.)	N/A	N/A	N/A

Table 13: Ground Station characteristics

In order to establish the coverage visibility and preliminary space link performances, the ground station locations and parameters in Table 13 can be used.

Additionally, stations from the Augmented Network consisting of commercial 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 (Antarctica)	7.3	S X/S X	

Table 14: ESTRACK Augmented Network ground stations.

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.

When considering stations beyond the core ESTRACK network, their capability to comply with the frequency allocations specified in 5.2.4 shall be checked.

5.4. SCIENCE OPERATIONS ELEMENTS

ESA's Science Operations Centres comprise a standard set of tools, infrastructure, and services providing:

- a standardized SOC - MOC interface for exchange of all mission products including their technical validation and first-look analysis,
- tools for generation of conflict free and optimized instrument operations plans, timelines and payload command sequences,
- orchestration and data management tools for execution / provision / ingestion of mission software and products to / from Consortia,
- a platform for hosting calls for proposals from Guest Observers,
- a centralised archive of mission data to the scientific community worldwide, providing free and open data discovery and access services, complemented by tools to facilitate archival research.

Proposals shall indicate what is expected from ESA, and what elements are intended to be provided by the proposing team.

It is expected that contributing science operations elements developed and operated by the proposing team, including their sustaining engineering, are briefly described in the proposal. Typical examples include: algorithms or tools for optimizing the scientific planning of observations and respective payload configuration, scientific data processing tools, payload health and performance monitoring tools, tools for deriving high-level scientific products, etc.

Any mission driver related to science operations with an expected impact on the mission science return and cost should also be detailed in the proposal (e.g., large data volumes, high-processing computing needs, complex data processing algorithms with low TRL, on-board automated science-driven operations or science data processing, etc).

6. PAYLOAD ELEMENTS

6.1. CRYOGENIC PAYLOADS

ESA has a long history in cryogenic missions for space science using passive and active systems to reach various cold temperature levels for the instruments. For instruments requiring detector cooling down to 50-100mK, ESA was in charge of providing the pre-cooling down to 2K, whereas the final cooling stages (down to sub-K range) are part of the Focal Plane Assemblies provided as a Member State/partner contribution.

System	Heritage	Comment
Stored cryogen cryostat	sfHe cryostats for ISO, Herschel	- Cost not compatible with M-class
Single stage active cooling down to 50K (incl. single multi-stage cooler)	Earth Observation, EnVision …	- Use of Integrated Cooler Detector Assemblies to be considered - Provision of active cooler systems by ESA could be envisaged within M-class budget envelope
Passive cooling down to 50K	Planck Telescope, Ariel Telescope and FGS, NIRSpec, Euclid …	- Passive cooling down to 50K using multiple cooling stages demonstrated in Europe - Passive cooling down to 30K demonstrated by JWST, but verification by test is very demanding and might not be compatible with an M-class mission
Active cooling from ambient down to 2K	Athena study with X-IFU cryostat	- Very complex system requiring large, newly developed coolers - Not compatible with an M-class mission
Passive/active cooling down to 4K	NewAthena X-IFU cooling (4K->50mK), JWST MIRI (6K), Ariel AIRS (42K), Planck HFI cooling (0.1K)	- Passive cooling down to 50-60K to limit the parasitic heat loads on the lower temperature system - Active cooling system(s) for lower temperatures - Non-redundant systems

Table 15: Cryogenic architectures

6.1.1. 4K cooling system with passive pre-cooling for large payloads

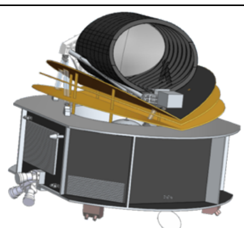
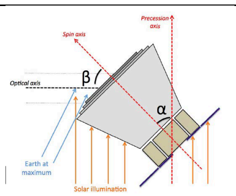
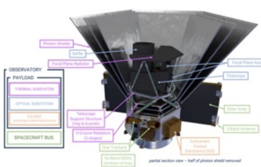
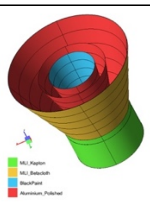
Following the reformulation study of Athena, a passive/active cooling chain is considered the simplest system and might be affordable in an M mission, if either an off the shelf 4 K cooler system is considered or the active coolers are provided as a Member State/partner contribution (as done in the past).
For enabling future M missions requiring cryogenic sub-K detectors, ESA is initiating in parallel to this call the development of a suitable 4 K cooler system, which should enable a cryogenic system within the M-class budget. The last cooling stages below 4 K are assumed to be undertaken by the instrument consortium. The key parameters of the 4 K cooler system are summarised in the following Table.

	Parameter	Comment
Heritage	Use of existing, qualified compressors from Earth observation and/or Ariel	Heritage compressors with existing drive electronics enable a non-redundant system and minimises development risks
Cooling Power at 4.5K	~30 mW @ 4.5 K for the instrument (cooler specification will include allocation for margins and system parasitic in addition)	Limited to 4.5 K since cooling at 2 K is considered too demanding

Cooling Power at intermediate temperatures	100 mW @ 20 K and 500 mW @ 120 K (cooler spec will include allocation for margins and system parasitic in addition)	to intercept parasitic heat from structure, harness etc
Mass	< 55 kg	Non-redundant system derived from existing building blocks
Power	500 W	During nominal operations
Exported vibrations	< 1N rms all axes	Cooler assembly in the SVM
Separation length cold to warm	> 2m	Distance between the Cooler assembly in SVM and the cold tip

Table 16: Key performance parameter of a 4K cooler system to be developed by ESA

For the passive cooling systems down to 50 K, various systems have been used or studied. A few examples are shown in the Table below.

Mission /Study	**Configuration**	**Comment**
Planck		➢ Slow spinner, full sky survey in L2 ➢ VGroove-3 passive cooling in orbit below 50K ➢ Tested in CSL with Helium shrouds ➢ Cold Payload mass (Instruments incl. coolers and reflector): ~180kg ➢ Shared launch and development with Herschel ➢ 20 K sorption cooler from JPL (US) ➢ 4K JT cooler from RAL (UK) ➢ 0.1K dilution from ALAT (FR)
Ariel		➢ 3-axis stabilised SC in L2 ➢ PLM to be provided by consortium ➢ PLM mass: ~500 kg ➢ To be tested in RAL Space using dedicated facility under development ➢ 32 K Ne-JT cooler from RAL (UK) ➢ Optics and FGS passively cooled to 60-80 K
CMB Polarisation Mission CDF study		➢ CDF study: https://sci.esa.int/s/AGdG7lw ➢ Spinner, Full sky survey
SphereX		➢ Small US mission in LEO (sun-synchronous) ➢ Full sky survey, always pointing away from the earth ➢ Temperature achieved 50-80 K ➢ https://spherex.caltech.edu
Modified SphereX concept		➢ Derived concept from SphereX in LEO (SSO 895km, dusk dawn) within the limits of Vega ➢ Limited pointing (±10deg away from anti-Earth) for ~ 15min ➢ 60-80K passive cooling for ~100kg cold payload

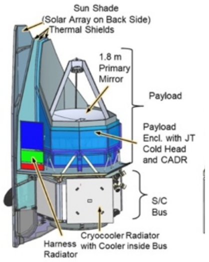
PRIMA (Astrophysics probe candidate for NASA)		➢ NASA/APEX mission candidate under study ➢ All Aluminum 1.8m Telescope at 4.5K ➢ Passive and active cooling ➢ https://prima.ipac.caltech.edu

Table 17: V-groove systems

The overall costs of the 4K cooling system is driven by:

- Cryogenic support structure: as a guideline, the cold payload should not exceed 300kg
- V-Groove panels number and size: deployable systems should be excluded
- Active cooling system: The active 4K cooler is a significant contributor. Cooling down to 20-30K can be achieved with cheaper single stage systems (next chapter)
- Cryogenic testing: specific test facilities (e.g. need for Helium-cooled shrouds in vacuum chambers) and test duration are significant cost contributors. To limit these costs, it is assumed that only functional tests are performed at instrument system level (i.e. full performance and calibration is performed by the instrument provider prior to delivery) and that cooldown accelerators are implemented at instrument level
- System complexity: Additional requirements on the cryogenic system (e.g. EMC, microvibration, pointing) need to be critically assessed

Assuming a successful development of the 4K cooler system, the provision by ESA of a 4K cryogenic system for the payload will consume a significant part of the ESA CaC (e.g. in the range of ~8%) and hence excludes the provision of other payload elements by ESA (e.g. telescope, detectors).

6.1.2. Single Stage coolers

Several single stage coolers have been developed in Europe for space applications. Table 18 provides a list of current suppliers retained in ESA projects.

Name	Type	link
RAL-STFC	Stirling and JT cooler	RAL Cryogenics-and-Magnetics
AIM Infrarot	Single stage Stirling and PulseTube cooler	AIM-IR space cooler
Thales Cryogenics	Single and 2-stage Pulse Tube cooler	TCBv cooler
Air Liquide	Pulse Tube cooler	Air liquid space cooler

Table 18: European cooler supplier

In addition to the above coolers, the 4K cooling system described in Table 16 can also be used as a self-standing cooling system to cool simple Detectors/Focal plane as done for the SMILES payload (JP) on the ISS.

6.2. DETECTORS

6.2.1. Detectors under development

ESA has several ongoing detector developments targeting Science applications. The CIS(304) CMOS detector is being developed by Teledyne-E2V (UK) and should reach TRL 5 during the M/F study Phase. Its characteristics are provided for information.

Parameters	CIS304-33
Pixel pitch	10 µm
Array size	4500 x 4340
Pixel type	Pinned Photodiode, Switchable dual gain
Operating Modes include	Rolling Shutter Global Shutter (Simple global shutter and with Digital Double Sampling (DDS)) High Dynamic Range (HDR) Staircase (multiple nondestructive reads)
ADC bits	Selectable 12 or 14 bits
Frame rate	16 fps @12 bits (Rolling shutter) 16 fps max @ 12 bits (Simple Global shutter) 8 fps max @ (Global Shutter with DDS)
Region of Interest	Capable of selecting region of interest in row and column directions • freedom to select any or all combinations of adjacent odd and even rows • can select which of the 6 outputs to use to select different groups of columns
Multiple Gain settings	Any combination of pixel and pre-amplifier gain: Pixel gain: x1, x10 Pre-Amplifier gain: x1, x3, x7, x15, x31
Full Well Capacity	> 140 ke$^-$ (lowest pixel gain setting) 15 ke- (highest pixel gain setting)
Noise	Expectations: 2 e$^-$ (high gain - pixel and pre-amp - rolling shutter) 5 e$^-$ (high gain global shutter with DDS) 30 e$^-$ (low gain rolling shutter)
Dynamic Range with HDR operation	95dB typical.
QE @550nm	95%. This is dependent on anti-reflection coating (for BSI only)
Dark Current	0.01e-/s @-50°C Dark current halves for every reduction of 5-6°C
Interface	6 CML outputs 50 MHz Master Clock SPI
Package format	9K PCB
Power Dissipation	2.6 W (at full frame rate) (TBC)

Table 19: CIS-304 predicted performance

7. MISSION IMPLEMENTATION CONSTRAINTS

7.1. ESA MISSION CLASSIFICATION

In 2024 ESA has introduced a mission classification scheme. The purpose of such classification is to provide a framework to define the management, engineering and product assurance approach to be applied to an ESA mission.

There are four categories of missions defined in the table below

Mission Class	Mission Characteristics	Mission Description	Typical Mission Examples
ALPHA (high criticality)	✓ Top class missions ✓ Extremely critical and strategic for ESA. ✓ Budget > 400 M€ ✓ Lifetime > 7 Years.	✓ Critical strategy/safety (e.g. human spaceflight) ✓ Requirements are high, acceptable risk is very low. ✓ Performances to be met whatever it takes	✓ Aeolus-2 ✓ ARGONAUT ✓ EarthCARE ✓ MetOP-SG ✓ MTG ✓ VIGIL ✓ ...
BETA (medium criticality)	✓ High class missions, ✓ Highly critical and strategic for ESA ✓ Budget 200 to 400M€, ✓ Lifetime 5 to 7 Years,	✓ Requirements are relatively high, and the acceptable risk is low. ✓ Finding the best compromise between risk and cost to deliver the mission	✓ Copernicus ✓ Comet-I ✓ EnVision ✓ FLEX ✓ HARMONY ✓ Sentinel Missions ✓ ...
GAMMA (low to medium criticality)	✓ Medium class missions, (e.g. hosting New Space type of mission) ✓ Medium critical and strategic for ESA Budget 25 to 200M€ ✓ Lifetime 2 to 5 Years,	✓ Requirements are moderate with a non-negligible risk. ✓ Mission is designed according to a hard cost limit (affordability approach)	✓ Aurora ✓ Camila ✓ MicroGeo ✓ RAMSES ✓ SCOUTs ✓ WISDOMS ✓ ...
DELTA (low criticality)	✓ Low class mission, ✓ Low critical and strategic for ESA ✓ Budget < 25M€, ✓ Lifetime <2 years	✓ Requirements are very limited with a significant risk. ✓ Almost full delegation to industry (Minimum requirements but increased risk)	✓ YPSAT

Table 20: ESA Mission Classification

The classification of a mission is dependent on a set of conditions that include allocated budget, development time, operational needs, etc. The classification is performed in phase 0/A by the Agency.

For missions in the class alpha and beta, it is expected that the ECSS standards will largely apply, including the Product Assurance ones. Tailoring in specific areas may be acceptable but will have to be discussed and justified.

For missions in the class gamma, relaxations of the ECSS Standards are allowed, in particular in the areas of electronic components and materials and processes, where industrial practices and standards may be considered acceptable.
Failure tolerance is normally iterated during the study phase and later decided by the Programme/Project. Failure tolerance affecting safety is not tailorable and it is independent from

mission class. The relevant requirements are specified directly in ECSS Q-40 and in the safety regulations of the proposed launcher.

The classification of a mission is performed in phase 0/A by the Agency. However, as a guideline, M mission candidates are expected to fall into alpha or beta class while, F missions will likely be in the gamma class (e.g. F2 mission ARRAKIHS) and, in a few cases may be beta.

7.2. SPACE DEBRIS MITIGATION

In October 2023, ESA has issued a new Space Debris Mitigation Requirements Document RD[8], that is applicable to all ESA projects.
The new requirements are more stringent than the policy in the previously applicable ISO standard.
They apply mostly to spacecraft in Earth orbit, i.e. all Earth-bound orbits including orbits around Sun-Earth Lagrangian points. However, a subset of requirements is applicable also to Lunar orbit.

Hereafter, is a summary of the main practical mission constraints stemming from the new requirements:

1. Spacecraft shall not release any object in space (e.g. telescope cover) in nominal operations
2. Spacecraft shall be passivated at the end of their mission, i.e. energy from batteries and propulsion system shall be depleted.
3. Spacecraft shall have sufficient capability to perform collision avoidance manoeuvre, if warned of incoming debris on its trajectory.
4. Spacecraft shall be disposed of at the end of the mission. The spacecraft reliability at end of mission shall allow a disposal manoeuvre with 90% probability of success
5. The disposal shall be achieved by one of the following means, in order of preference:
    - Immediate Earth atmospheric re-entry after end of mission
    - Disposal in an orbit with a natural orbital decay leading to Earth re-entry in less than 5 years and cumulative spacecraft collision probability (from its end of life until re-entry) with space objects larger than 1 cm below 10^{-3}
    - If not operating in, nor crossing, the LEO protected region, disposal in a graveyard orbit that satisfies both following conditions: (a) Long-term perturbation forces do not cause it to cross the protected regions within 100 years and (b) cumulative collision probability with space objects larger than 1 cm is below 10^{-3} for up to 100 years after the end of life.
    - It does not cross the GEO protected region for at least 100 years with a probability >90%.
6. Uncontrolled re-entry is not allowed if casualty risk is > 10^{-4}

LEO and GEO protected regions are shown in Figure 8 below.

Except for very small satellites launched at low altitudes, one practical consequence for the spacecraft design is the need to implement a propulsion disposal manoeuvre at end of life.

As an example, a mission in the Sun-Earth Lagrange points L1 or L2, will comply with the requirement by performing a ~10 m/s delta-V manoeuvre at the end-of-life. This transfers the SC into a heliocentric orbit that does not cross the protected regions for at least 100 years with a probability >90%.

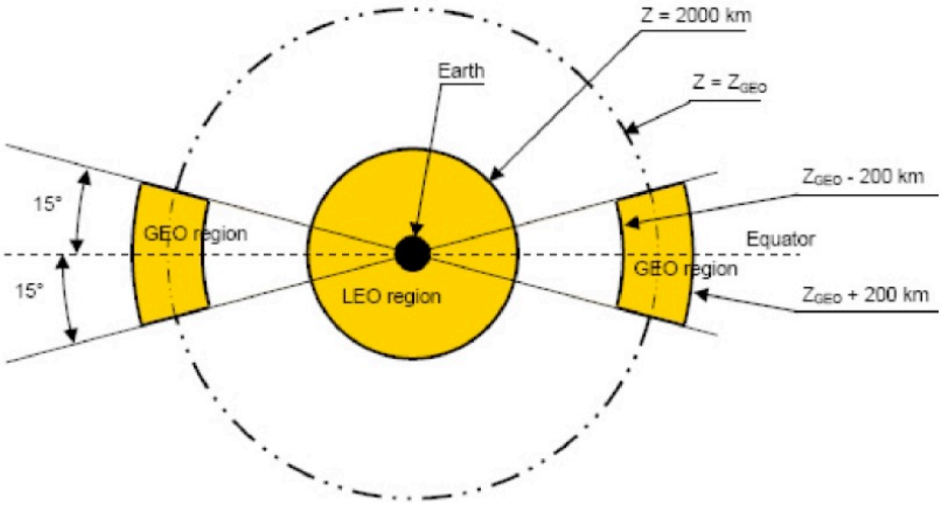


Figure 8: LEO and GEO protected regions [6].

When fragments of the SC may survive an uncontrolled re-entry, a controlled re-entry manoeuvre has to be performed to mitigate the risk of ground casualty. The Delta-V required for this manoeuvre must be included in the sizing of the propulsion subsystem. This requirement applies to the SC as well as any other large debris generated by the mission, such as launch vehicle upper stages, multi-SC adapters, ejected covers etc.

7.3. PLANETARY PROTECTION

ESA Planetary missions shall comply with the categories and associated requirements reported in ND[3].

7.4. TECHNOLOGY READINESS

7.4.1. Technology Readiness requirements for the M-mission

The overall spacecraft development must be compatible with an implementation schedule consisting of ~ 5-6 years for the preparation phase (phases 0/A/B1) and ~ 6-7 years for the development phase (C/D).

TRL 5/6 (as defined in Appendix B) is formally required for all mission elements (platform and payload) only by the mission adoption (end of Phase B1). However, for schedule critical elements, it is recommended that TRL ≥ 5 is safely reached at the end of the phase A, since the Technology Readiness will be a major element for the mission selection.

Therefore, the spacecraft can be a new development but shall rely on substantial heritage. It is recommended that the mission does not contain any element with TRL < 4 at the time of the proposal. In case some elements are at TRL 4 and are critically needed for achieving the mission science goals, the proposer shall present a credible path for reaching TRL ≥ 5 by the mission selection (therefore within ~ 3 years). In such case, the proposer is also invited to identify a back-up scenario at TRL ≥ 5 with reduced mission performance.

As for the F mission, when assessing the technology readiness, the following guidelines shall be considered:

- Reference to heritage shall consider potential obsolescence of components, subsystems and human expertise.
- If a technology has already flown but for a different application and in a less demanding environment, its TRL is ≤ 4.

7.4.2. Technology Readiness requirements for the F-mission

The overall spacecraft development must be compatible with a fast implementation schedule, consisting of 2-3 years for the preparation phase (phases 0/A/B) and ~4 years for the development phase (C/D).

The platform must rely on existing technologies and should be derived from flight proven platforms, aiming at maximising reuse. Therefore, TRL 5/6 is required at the time of the mission proposal for all platform elements. Technology Readiness Level (TRL) 7 (as defined in Appendix B) is also nominally required for all platform elements at adoption (end of Phase B). If the platform is derived from an existing flight proven design, with a few low-risk modifications limited to specific elements, TRL 6 may be tolerated for these elements at the time of the mission adoption subject to compatibility with the fast implementation schedule.

The payload can be a new development but must also rely on available technologies and shall be compatible with a fast implementation schedule, typically ~ 2 years for the preparation phases (phases 0/A/B) and ~ 3 years for the development phase (phases C/D). Therefore, it is recommended to have all payload elements at TRL ≥ 5 already at the time of the mission proposal. In case the payload features some critical element at TRL 4 with a credible path to reach TRL 5 within ~ 2 years, the proposer is invited to also consider a back-up scenario with a TRL ≥ 5 and lower performance.

When assessing the technology readiness, the following guidelines shall be considered:

- Reference to heritage shall consider potential obsolescence of components, subsystems and human expertise.
- If a technology has already flown but for a different application and in a less demanding environment, its TRL is ≤ 4.

8. PROGRAMMATIC ASSUMPTIONS

8.1. RESPONSIBILITIES

The share of responsibilities between ESA and the Member States on the payload elements shall be clearly identified in the proposal.

For an ESA-led mission, the nominal scheme is to have the spacecraft launch and operations (MOC) carried out by ESA. The science operations are led by ESA (SOC) with contributions from the Member States to be defined in the proposal. In case other schemes are proposed, their feasibility will be assessed based on the proposal content.

8.1.1. Payload Provision

As a rule, any Member State payload provision shall be commensurable with the lead Member State funding capability. Therefore, large payload elements involving a large consortium of Member States shall be proposed under ESA responsibility and **included in ESA cost** (the CaC must still remain within the given limits**)**, considering as a minimum:

- overall payload management,
    - Interface management between major sub-systems
    - schedule
- system engineering
    - end-to-end system performance
    - system level analysis (mechanical, thermal, EMC …)
- system AIV (i.e. system level integration on the SC and functional/environmental tests)

Typical examples of payload elements that can be considered under the Member States responsibility are planetary instruments, or focal plane instruments with associated electronics, of typical mass below 100 kg.

8.2. MISSION REFERENCE SCHEDULE

Table 21 and Table 22 provide the reference schedule to be assumed respectively for the M-and F-missions.

Event M	Date or duration	Note
Start of Study Phase 0	Q4 2026	Up to five candidates
Downselection of Phase A candidates	Q4 2027	Typically, three out of five candidates could be considered until end of Phase A
Mission Selection	Q2 2030	At the end of the Phase A
Mission Adoption	Q4 2032	At the end of the Phase B1
M Launch	~2041	Approximate date, mission dependent
Nominal in-orbit operations	Typically 3-4 years	Must be compatible with ESA CaC

Table 21: Reference schedule for the M-mission

Event F	Date or duration	Note
Start of Study Phase 0	Q4 2026	Typically one candidate and a one backup (for a short time), with the intention to rapidly focus the effort on the selected mission
Mission Adoption	Q2 2030	At the end of Phase B
F Launch	~2034	Approximate date, mission dependent
Nominal in-orbit operations	Typically 2 years	Must be compatible with ESA CaC

Table 22: Reference schedule for the F-mission

8.3. MISSION COST ELEMENTS

ESA Cost at Completion (CaC) target is 670 M€ for the M mission, and 205 M€ for the F mission. The CaC covers all ESA activities following the mission adoption, in particular:

- The spacecraft development phase (B2/C/D/E1 for the M mission and C/D/E1 for the F mission)
- The MOC and SOC developments
- The launch services
- The nominal in-orbit operations, including disposal at the nominal end of life.

Table 23 and Table 24 provide an **indicative cost breakdown** for an ESA mission, respectively for the F mission with a Vega-C launch and for the M mission with an Ariane 6 launch (full and shared). In case no sharing partner is evident a full launcher cost shall be assumed.
It is important to note that these tables are indicative and subject to several influence factors like operations duration and complexity, launcher details, technology readiness and risks.

F- Mission	**Indicative portion of total CaC**	
	Full Vega	Shared Vega
Spacecraft and Payload contribution under ESA responsibility	42 %	53 %
Launch Vehicle	26 %	13 %
ESA Project	10%	12%
Operations (MOC and SOC)	10%	
Margin	12%	

Table 23: Indicative cost breakdown for the F-mission using Vega-C.
Shared assumes 50%-50% launch cost sharing

M-Mission	**Indicative portion of total CaC**				
	Full Vega	Full A62	Shared A62	Full A64	Shared A64
Spacecraft and Payload contribution under ESA responsibility	52%	46%	52%	43%	52%
Launch Vehicle	8%	16%	8%	20%	10%
ESA Project	14%	12%	14%	12%	14%
Operations (MOC and SOC)	14%				
Margin	12%				

Table 24: Indicative cost breakdown for the M-mission using Ariane 6 or Vega-C.
Shared assumes 50%-50% launch cost sharing.

8.4. EXPORT CONTROL

European and US regulations for export control are increasingly affecting the provision of instruments for ESA. The instrument teams shall therefore ensure that relevant key persons can get clearance for US/European export control and that deliverables to ESA are compliant with national export rules.

APPENDIX A - ABBREVIATIONS AND ACRONYMS

Abbreviation	Definition
ACS	Attitude Control System
AIT	Assembly, Integration and Testing
AIV	Assembly, Integration and Verification
AME	Absolute Measurement Error
AOCS	Attitude and Orbit Control System
APE	Absolute Pointing Error
AU	Astronomical Unit
Bps	Bits per second
CaC	Cost at Completion
CDF	Concurrent Design Facility
CDR	Critical Design Review
CoG	Centre of Gravity
CSL	Centre Spatial de Liège
DHS	Data Handling System
DLS	Dual Launch Structure
DSN	Deep-Space Network
e.c.	Economic Conditions
ECSS	European Cooperation for Space Standardisation
EM	Engineering Model
EMC	Electromagnetic Compatibility
EoL	End of Life
ESA	European Space Agency
ESAC	European Space Astronomy Centre
ESOC	European Space Operations Centre
ESTEC	European Space Research & Technology Centre
FFT	Fast Fourier Transform
FM	Flight Model
FoR	Field of Regard
FoV	Field of View
GEO	Geostationary Earth Orbit
GL	Gravity Loss
GTO	GEO Transfer Orbit
HEO	High Elliptical Orbit
HGA	High Gain Antenna
ISO	International Organisation for Standardisation
ITU	International Telecommunication Union
Kbps	Kilobits per second
LEO	Low Earth Orbit
LEOP	Launch and Early Operations Phase
LGA	Low Gain Antenna
LoS	Line of Sight
LV	Launch Vehicle
MAR	Mission Adoption Review
Mbps	Megabits per second
MLI	Multi-Layer Insulation
MLS	Multi Launch Service
MOC	Mission Operations Centre
MoI	Moment of Inertia
MRD	Mission Requirements Document
MSR	Mission Selection Review
N/A	Not Applicable

PA	Product Assurance
PAS	Payload Adapter System
PDD	Payload Definition Document
PDR	Preliminary Design Review
PFM	Proto Flight Model
PI	Principal Investigator
PLM	Payload Module
PM	Propulsion Module
PSD	Power Spectral Density
PSF	Point Spread Function
QM	Qualification Model
RD	Reference Document
RF	Radio Frequency
RMS	Root Mean Square
RPE	Relative Pointing Error
RSS	Root Sum Square
SAA	Solar Aspect Angle
SC	Spacecraft
SciRD	Science Requirements Document
SDC	Science Data Centre
SEL1	Sun-Earth Lagrangian point 1
SEL2	Sun-Earth Lagrangian point 2
SEL4	Sun-Earth Lagrangian point 4
SEL5	Sun-Earth Lagrangian point 5
SEP	Solar Electric Propulsion
SNR	Signal to Noise Ratio
SOC	Science Operations Centre
SSCE	Sun Spacecraft Earth angle
SSMS	Small Spacecraft Mission Service
SSO	Sun Synchronous Orbit
SVM	Service Module
TBC	To Be Confirmed
TBD	To Be Defined
TM	Telemetry
TWTA	Travelling Wave Tube Amplifier
UTC	Coordinated Universal Time

APPENDIX B - DEFINITION OF TECHNOLOGY READINESS LEVEL (TRL)

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.

Technology Readiness Level	Milestone achieved for the element	Work achievement (documented)
TRL 6: Model demonstrating the critical functions of the element in a relevant environment	Critical functions of the element are verified, performance is demonstrated in the relevant environment and representative model(s) in form, fit and function.	Definition of performance requirements and of the relevant environment. Identification and analysis of the element critical functions. Design of the element, supported by appropriate models for the critical functions verification. Critical function test plan. Model definition for the critical function verifications. Model test reports.
TRL 7: Model 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 build 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 realisation. 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.