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

TECHNICAL ANNEX

Prepared by	Future Missions Department (SCI-F)
	Directorate of Science
Document Type	MAN - Manual / User Guide / Handbook
Reference	ESA-SCI-F-MGT-MAN-001
Issue/Revision	1
Date of Issue	13/12/2021
Status	Issued

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1. INTRODUCTION

This document provides the boundary conditions, together with a set of technical and programmatic information, for the mission proposals in response to the call for medium (M-class) and fast (F-class) mission opportunities in the ESA science programme (also called M7 and F2 in this document).

Proposers can access 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 Appendix 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

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

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

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

RD[5] ESA Tracking Stations (ESTRACK) Facilities Manual EFM, DOPS-ESTR-OPS-MAN-1001-OPS-ONN, Issue 2.1, March 2019

RD[6] ECSS-E-HB-11A, Technology readiness level (TRL) guidelines, Mar. 2017, www.ecss.nl

RD[7] ECSS-E-HB-60-10A, Control performance guidelines, Dec. 2010, www.ecss.nl

RD[8] ESSB-HB-E-003, ESA pointing error engineering handbook, Jul. 2011, www.ecss.nl

RD[9] ESSB-HB-U-002, ESA Space Debris Mitigation Compliance Verification Guidelines, www.ecss.nl

3.2. NORMATIVE DOCUMENTS

[ND1] ECSS-E-AS-11C, Definition of the Technology Readiness Levels (TRLs) and their criteria of assessment, Oct. 2014, www.ecss.nl

[ND2] ECSS-E-ST-50-05C Rev. 2, Radio Frequency and Modulation, www.ecss.nl

[ND3] ECSS-U-ST-20C, Planetary protection, www.ecss.nl

4. SUMMARY OF THE BOUNDARY CONDITIONS

For the present Call, the management and system activities of large payload elements are foreseen to be under ESA responsibility (see Sect. 7.1). The relevant costs must be included in the ESA CaC.

4.1. BOUNDARY CONDITIONS FOR THE F MISSION

Element	Request	Comments or Guidelines
ESA Cost at Completion (CaC)	≤ 175 M€	Includes all elements to be funded by ESA, including the launch services and the contributions to the payload (if applicable). Excludes Member State and international partner contributions.
Science objectives and instruments	Any science objective can be proposed.	The science instruments must be defined in relation with the science objectives. The core science objectives and the proposed concept must allow a rapid technical convergence through a design-to-cost approach in the preparation phase.
Launcher	The baseline assumption foresees use of the Vega-C launcher.	Other schemes may be considered subject to feasibility. Procurement of non-European launchers by ESA is excluded.
Spacecraft dry mass	Of order 450-500 kg	Including payload and all margins.
Spacecraft wet mass	≤ 750 kg	The actual launch mass constraint will depend on the target orbit and the associated Vega-C performance. The mass constraint stated here provides a limit to the spacecraft cost in line with the CaC constraints. The spacecraft wet mass includes the platform(s) with the 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, must be included.
Overall science payload mass	Of order ≤ 70 kg	The actual payload mass may be lower depending on the mission profile (see sections 3 and 4). Proposers should 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 (S/C) TRL	TRL ≥ 6 (on the ISO scale, see Appendix B) at mission proposal	The platform equipment must be at TRL ≥ 7 (space qualified for the mission needs and available) at the time of the mission adoption. TRL 6 is ideally required at the time of the mission proposal (TRL 5 acceptable if properly justified). To minimize mission cost ESA will aim to reuse existing platforms (possibly with some modifications). While proposers are free to suggest possible options for the platform, the actual decision on the platform feasibility, reuse, etc. remain with ESA.
Science Payload TRL	TRL ≥ 6 at mission adoption	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 verifications and pre-developments, lasting up to two years, can be envisaged during the definition phase. The maturity of the proposed payload must be such that a status compatible with a Preliminary Desing Rewiew (PDR, i.e. TRL 6, including the completion of the instrument detailed definition and the confirmation of all interface requirements) can be reached within ~3 years from the proposal selection, before the mission adoption. ESA can support the detailed design of the instrument and (if needed) pre-developments prior to adoption in the interest of securing the payload development schedule. Proposers are invited to spell out the role, responsibilities, and heritage of the payload providers together with the expected funding scheme. The preliminary payload development plan should be presented by identifying (as possible) the pre-developments needed during the phases 0/A/B.
International collaboration	Can be envisaged, contingent on support from the proposed international partner.	The F mission must be ESA-led.
Spacecraft operations	The baseline approach foresees spacecraft operations under ESA responsibility, with possible contributions from the Member States or partners	Other schemes may be considered subject to feasibility.

	to the science ground segment. Nominal duration of science operations typically < 2 years	The nominal duration of science operations does not include the cruise phase, nor the disposal (if applicable).

4.2. BOUNDARY CONDITIONS FOR THE M MISSION

Element	Request	Comments or Guidelines
ESA Cost at Completion (CaC)	≤ 550 M€	Includes all elements to be funded by ESA, including the launch services and the contributions to the payload (if applicable). Excludes Member State and international partner contributions.
Science objectives and instruments	Any science objective can be proposed.	The science instruments must be defined in relation with the science objectives. The core science objectives and the proposed concept must allow a rapid technical convergence through a design-to-cost approach in the preparation phase.
Launcher	The baseline assumption foresees use of the Ariane 62 or Vega-C launchers.	Other schemes may be considered subject to feasibility. Procurement of non-European launchers by ESA is excluded.
Spacecraft dry mass	≤1500 kg	Recommended upper limit in view of the cost target
Platform and Science Payload TRL	TRL 5-6 (on the ISO scale, see Appendix B) by mission adoption	The spacecraft (platform and payload) can feature new developments but must rely on substantial heritage. TRL ≥ 5 must be targeted by the mission selection (end of Phase A). Therefore, minor developments are possible provided they can be 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 the mission science goals, the proposer must 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 must achieve a status compatible with the System Requirements Review (SRR, i.e., detailed design including the definition of interface requirements) by the time of the

		mission's adoption, i.e. within ~5 years from the proposal selection. ESA can support the instrument detailed design and pre-developments during the phases 0/A with the goal of securing the payload development schedule. Following the mission selection at the end of the phase A, the Member States must fund the payload pre-development activities in the phase B1 followed by the flight models after the mission adoption. Proposers are invited to address in the proposal the role, responsibilities, and heritage of the payload providers, and their proposed payload development plan, including pre-development needs. The activities needed in phases 0/A and those to be achieved in phase B1 should be, insofar as possible, separately spelled out.
International collaboration	Can be envisaged, contingent on support from the proposed international partner.	Use of an M mission slot to contribute to a mission led by another Agency is possible, contingent on the partner expressing to ESA readiness to support the proposed collaboration scheme. ESA will assess the feasibility of the proposed scheme as part of the feasibility analysis.
Spacecraft and science operations	The baseline approach foresees spacecraft operations under ESA responsibility, with possible contributions from the Member States or partners to the science ground segment. Nominal duration of science operations typically <3 years	Other schemes may be considered subject to feasibility. The nominal duration of science operations does not include the cruise phase, nor the disposal (if applicable).

5. MISSION CONCEPT DEFINITION

5.1. LAUNCH VEHICLES

The proposed launch vehicle must be either Ariane 6 or Vega-C. The following sections provide an overview of the launch mass capability; the performance figures are indicative, and a specific mission analysis will be carried out for selected missions.

5.1.1. Ariane 6

There are two versions of Ariane 6: Ariane 62 and Ariane 64, depending on the number of boosters employed, as described in [RD1]. Given the ESA cost ceiling for this Call, the use of Ariane 64 is not recommended.

5.1.1.1. Ariane 62 performance

The following table provides the foreseen Ariane 62 performance data for selected orbits. The mass of the payload adapter (that can be selected from [RD1]) must be subtracted from the figures in the table to derive the available S/C mass to orbit.

Launch orbit	Orbital parameters	Performance (kg)
SSO	500 X 500 km, i=97.4 deg	7200
LEO Polar	900 X 900 km i=90 deg	7000
Sub-GTO	250 X 22 500 km, i= 6 deg	6000
GTO	250 X 35 768 km, i= 6 deg, ωp=178 deg	4500
HEO	310 X 100 000 km, i= 5.6 deg, ωp=140 deg	3632
Lunar transfer	200 X 400 000 km, i=6 deg	3500
SEL2 transfer	180 X 1 500 000 km, i=6 deg	3300
Escape	V_∞ = 1 km/s, δ = -1 deg	3236
	V_∞ = 2.3 km/s, δ = -3 deg	2688
	V_∞ = 2.6 km/s, δ = -4 deg	2558
	V_∞ = 3 km/s, δ = -4 deg	2383

Table 1: Ariane 62 performance – indicative values.

For escape trajectories:

- The performance decreases significantly if the declination varies from the value in the table;
- For intermediate values of the escape velocity V_∞, the mass performance can be estimated by interpolation.

5.1.1.2. Launch configuration and mechanical interfaces

Single launch, dual launch or launch in Multi-Launch Service (MLS) configuration are possible with Ariane 6. Available payload volumes and standard mechanical interfaces for all configurations are described in [RD1] and [RD2].

The dual launch configuration is not standard with Ariane 62, but it can be implemented with a custom Dual Launch Structure ("light DLS") as in the case of ARIEL and Comet Interceptor. The typical mass of the Dual Launch Structure is 600-800 kg (depending on its height). The mission CaC will have to include the cost of the Dual Launch Structure (which will be offset from the lower costs for the launch vehicle).

The use of Ariane 62 in dual launch configuration can be proposed for both F and M missions. As, no existing ESA dual launch opportunity is pre-identified for either mission, proposers should suggest the possible co-passenger.

The MLS configuration is relevant to small/mini satellites with a maximum mass of 500 kg. Available options are described in detail in [RD2].

5.1.2. Vega-C

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.

5.1.2.1. Vega-C performance

The following figure (taken from [RD3]) provides the performance for a LEO circular Sun Synchronous Orbit (SSO).

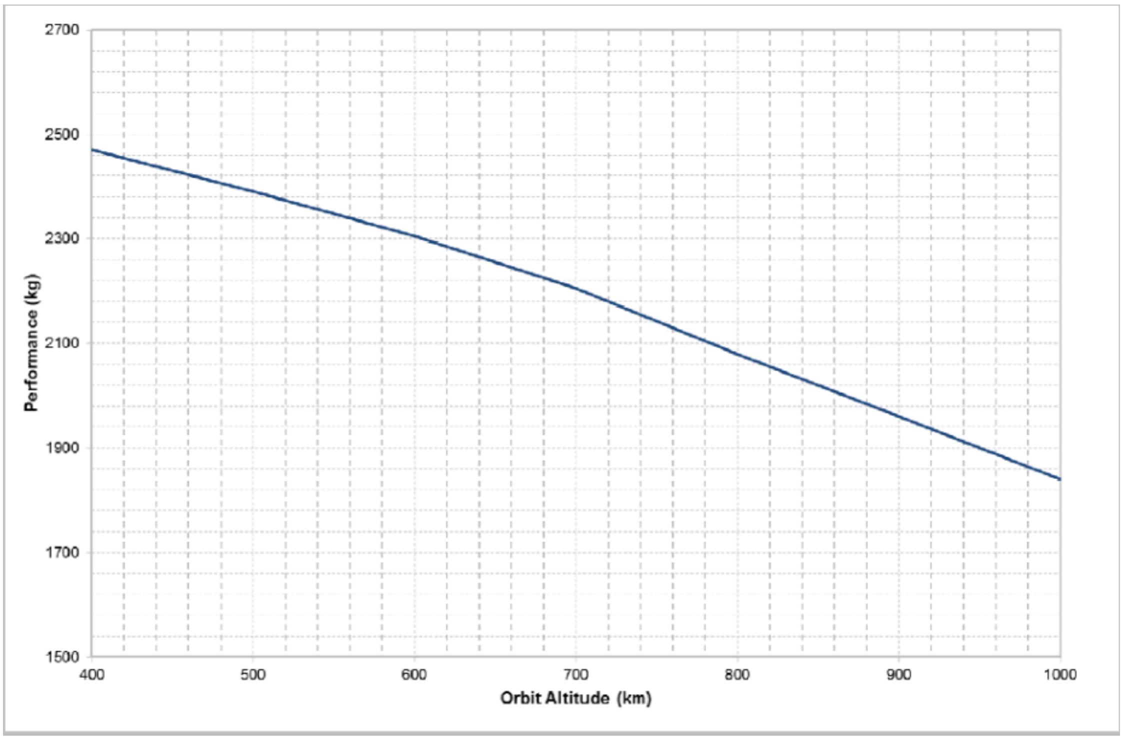

Figure 1: Vega-C performance in SSO

Table 2 provides the VEGA-C performance for other selected orbits.

Launch orbit	Orbital parameters	Performance (kg)
LEO polar	500X500 km i=88 deg	2250
LEO intermediate inclination	700X700 km, i=70 deg	2350
LEO equatorial	600X600 km, i= 5.4 deg	2980
LEO high equatorial	7400X7400 km i=15 deg	600
HEO equatorial	250X5700 km, i= 6 deg	1600
	150X20 000 km i=6.5 deg	600

Table 2: Vega-C estimated performance for selected destinations

Higher energy orbits can be achieved by using an additional propulsion module or through spacecraft propulsion (either electrical or chemical). Figure 2 shows an example of quasi-equatorial apogee altitude vs. mass achievable by using a bi-propellant propulsion module and starting from a HEO quasi-equatorial launch orbit (200X1550 km, 6 deg inclination). The cost of a propulsion stage needs to be included in the mission CaC.

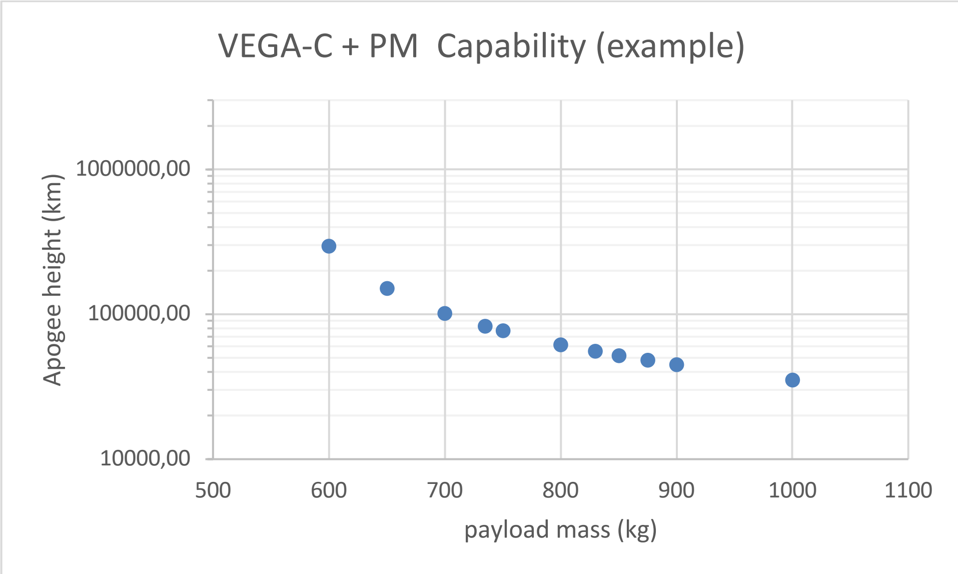

Figure 2: VEGA-C capability with Bi-Propellant Propulsion Module (example – adapter mass to be subtracted to obtain S/C mass).

5.1.2.2. Launch configuration and mechanical interfaces

Single launch, dual launch and launch in Small Spacecraft Mission Service (SSMS) configuration are possible with Vega-C. Available payload volumes and mechanical interfaces are detailed in [RD3] and [RD4].

The SSMS configuration is suitable for the launch of nano, micro and mini satellites.

5.1.3. Other launch vehicles

Launch services from an international partner may be considered contingent on the partner expressing to ESA readiness to support the proposed collaboration scheme.

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 S/C acceleration over time will not remain constant as the S/C 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. 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 an F mission assuming a Vega-C launch

Table 3 provides indicative key parameters for a set of potential orbits. The list of targets is not exhaustive, and other orbits can be considered if proposers can show to meet the call boundary conditions. Table 3 does not address cost and schedule of the possible missions, that will have to also be respected.

Note that:

- Given the current Vega-C performances in many of the potential indicative mission profiles a circular injection orbit of 3000 km altitude is used to minimize the radiation dose, which could jeopardize the health of the payload instruments during the long electric propulsion orbit transfer.
- The maximum spacecraft ΔV is around 4 km/s, using solar electric propulsion with a specific impulse of 1660 s. This rules out F missions with final destinations beyond the Sun-Earth Lagrange Points L1/L2.
- To minimize the cost of operations, ground commanding of manoeuvres during the science operations phase should be minimised, and the trade-off between ground commanding and on-board autonomy should favour on-board autonomy.

F2 Mission Profile Envelope (examples)						
Potential Target Orbit for F2 Mission	**Delivered F2 Mass at Target Orbit**	**Indicative F2 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 ~ 700Kg 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 deducting 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 ~700Kg 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 4.2Km/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 avoid radiation belts during the transfer). SEP raising using SNECMA PPS.1350. Target orbit is in graveyard region ~300 km beyond the geostationary orbit. Inclination reduction to 0 and stationkeeping 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 7000Km altitude orbit (to avoid radiation belts during the transfer).

Table 3: Potential destinations for the F mission using solar electric propulsion.

5.2.1.2. Examples of orbit transfer for the M mission assuming an Ariane 62 launch – interplanetary orbits with electric propulsion

Some examples of electric propulsion transfers are provided in Table 4 for interplanetary targets and with two different electric propulsion engine technologies (Hall Effect Thruster: PPS-5000, Ion Engine: T6).

	PPS-5000			**2x T6 (as on BepiColombo)**		
	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 4: Examples of Electric Propulsion Transfer for the M mission assuming an Ariane 62 launch.

5.2.1.3. Examples of orbit transfer for the M mission assuming an Ariane 62 launch – Sun-Earth L4/L5 Lagrange points with chemical propulsion

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

The L5 point is less demanding in terms of ΔV 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). The fuel demands for reaching L4/L5 can be lowered by increasing the transfer time, as illustrated in Table 5. Transfers are possible in discrete intervals, the shortest of which is 14 months. The next one is 26 months and offers significant benefits both in terms of escape C3 ($C3 = V_\infty^2$) and the ΔV applied at arrival. Longer transfers lead to further, though not significant savings.

Transfer duration [months]	**Departure trajectory C3 ($C3 = V^2_\infty$) [km^2/s^2]**	**Mass at launch (kg)**	**Arrival manoeuvre [km/s]**	**Mass after arrival manoeuvre (kg)**
14	2.016	3110	1.419	1970
26	0.582	3250	0.763	2550
38	0.272	3270	0.521	2770
50	0.157	3280	0.396	2885

Table 5: Approximate performance for transfers to the Sun-Earth L5 point using Ariane 62. The performance ("Mass after arrival manoeuvre") is estimated by assuming an arrival manoeuvre performed using the spacecraft on-board chemical propulsion (with a specific impulse of 317 s). Therefore, the delivered mass includes the dry mass of the chemical propulsion system.

5.2.2. Mass – F missions

To achieve the mission's challenging cost target ESA intends to implement the mission, whenever feasible, using existing platforms (with modifications if/as necessary). The mass of a F mission that is likely to respect the cost and schedule constraints is estimated to be:

- Total science payload mass <= 70 kg;
- Total spacecraft dry mass <= 450-500 kg (including the science payload);
- Spacecraft wet mass <= 750 kg. Fuel mass will depend on the targeted destination and on the propulsion system.

Large ΔVs will require the use of Solar Electric Propulsion (SEP) as explained in section 5.2.1.1.

5.2.3. Mass – M missions

The mass of the M mission that is likely to respect the cost and schedule constraints is estimated to be <= 1500 kg (dry mass). Proposals assuming a higher mass will need to properly explain why they could fit with the Call boundary conditions.

5.2.4. Communications

ESA science missions must comply with ITU frequency allocation requirements (see [ND6]). 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 S/C distance to Earth in the operational orbit:

1. Near Earth or Category A for S/C 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 S/C altitude above Earth surface ≥ 2 Mkm.

The frequency allocations and the maximum bandwidth that can be allocated to a single mission for specific bands are reported in Table 6 (from [ND6]). Actual allocation is likely to be a fraction of the value. The bandwidth limitation leads to a limitation on the maximum data rate that can be downlinked that together with constraints on the ground station visibility and the onboard memory, results in a limit on the maximum science data volume that can be transmitted to ground in a given time.

As an example, typical achievable X-band data rates as a function of distance to Earth, S/C High Gain Antenna diameter and RF power output (assuming ESTRACK 35 m ground antennas) are shown in Figure 3.

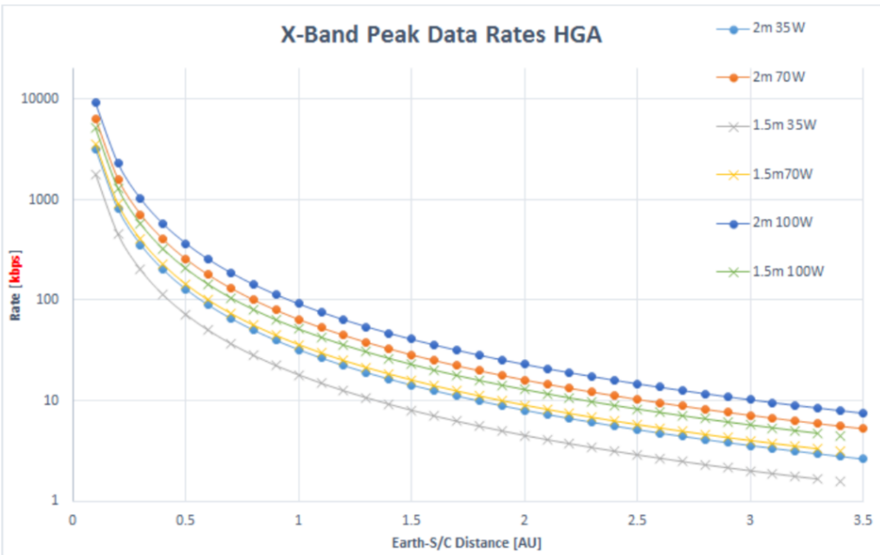

Figure 3: Typical downlink data rate in X-band vs Earth-S/C distance, antenna size and communication power.

Types of mission	Direction	Band	Frequencies (MHz)	Max bandwidth allowed	Examples (data rate)
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	
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 [ND6])	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)
		Ka	37 000 – 38 000	No limitation	Not used yet. Equipment and Ground infrastructure not yet available

Table 6: Allowed frequency bands and associated bandwidths.

5.2.5. Spacecraft Budgets and Margins

This section summarizes the margins required at system level for the proposed missions.

Parameter	Margin	Comments
S/C dry mass	25%	The 25% system margin applies to the nominal total spacecraft dry mass, which must be evaluated by including the maturity margins at equipment or subsystem level. The total spacecraft dry mass must include the total platform dry mass plus the allocated payload mass. The payload margin included in the allocated payload mass must be clearly identified. For the wet mass calculation, the propellant mass must be calculated with the total dry mass at launch including system margin.
Δ V	5%	System margin to be applied to the total Δ V requirement
Power	30%	System margin to be applied to the total power demand of the spacecraft. The power allocated to the payload and the relative margin must be clearly identified.
Pointing	100%	System margin to be applied to the pointing accuracy, knowledge and stability error predictions.
Data Rate	50%	System margin to be applied to the total calculated payload data rate.
Data Volume	50%	System margin to be applied to the total calculated payload data volume to be stored on board.
Communication Link	3 dB	The communication link budget for all mission phases must 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 100 K must include the following system-level margin: - 20% for systems operating between 50 K and 100 K - 50% for systems operating below 50 K - 100% for systems operating below 2 K

Table 7: Required System Contingencies and Margins

5.2.6. Pointing Requirements

Scientific measurement requirements in most cases imply requirements on spacecraft pointing accuracy and knowledge. The flow-down of the pointing requirements from the science observations and instrument performances must be clearly shown in the proposal. In particular, the relation with the instrument image quality and the wavefront error budget must be highlighted, when relevant.

Pointing requirements are specified through pointing error indices introduced in the pointing error engineering handbook [RD8]. The most common of such errors are listed below.

Metric	Index	Name	Definition
Absolute	APE	Absolute Pointing Error	Difference between the target (commanded) parameter (attitude, geolocation, etc.) and the actual parameter in a specified reference frame.
	AKE	Absolute Knowledge Error	Difference between the actual parameter (attitude, geolocation, etc.) and the known (measured or estimated) parameter in a specified reference frame.
Windowed Mean	MPE	Mean Pointing Error	Mean value of the APE over a specified time interval Δt.

	MKE	Mean Knowledge Error	Mean value of the AKE over a specified time interval Δt.
Windowed Variance	RPE	Relative Pointing Error	Difference between the APE at a given time within a time interval, Δt, and the MPE over the same time interval.
	RKE	Relative Knowledge Error	Difference between the APE at a given time within a time interval, Δt, and the MKE over the same time interval.
Windowed Stability	PDE	Performance Drift Error	Difference between MPEs taken over two-time intervals separated by a specified time, Δt_S, within a single observation period.
	KDE	Knowledge Drift Error	Difference between MKEs taken over two-time intervals separated by a specified time, Δt_S, within a single observation period.
Repeatability	PRE	Performance Repeatability Error	Difference between MPEs taken over two-time intervals separated by a specified time, Δt_S, within different observation periods.
	KRE	Knowledge Repeatability Error	Difference between MKEs taken over two-time intervals separated by a specified time, Δt_S, within different observation periods.

Table 8: Definition of Pointing Error Indices

The different pointing error metrics are further illustrated in the time domain in Figure 4.

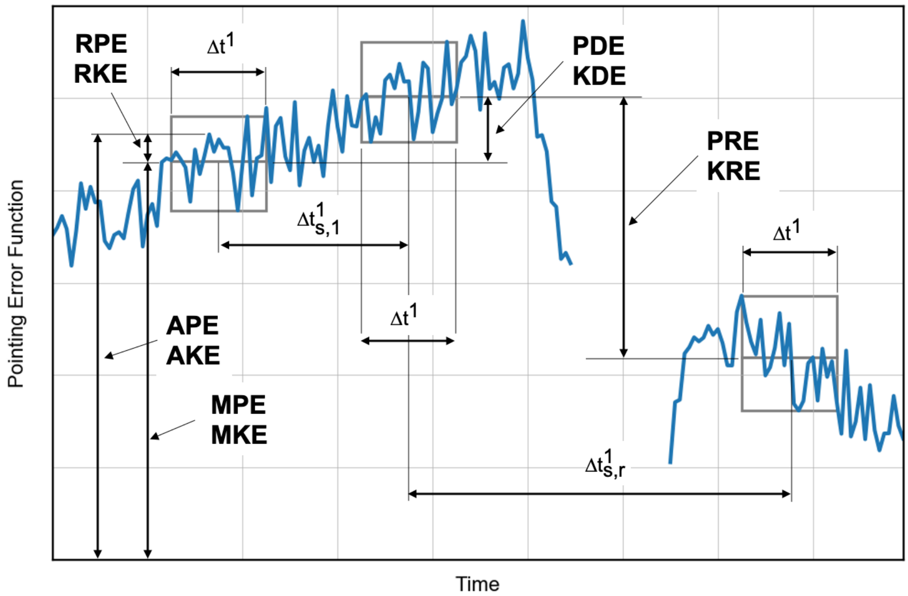

Figure 4: Illustration of Pointing Error Metrics

The proposer must express the critical pointing requirements for the proposed mission, i.e. those driving the science measurement performance and possibly the spacecraft cost. Pointing requirements are generally specified in terms of quantified probabilities. The applicable statistical definitions must be made explicit in the requirement formulation.

Table 9 provides pointing requirements of selected astronomical telescopes, for which the pointing requirements for the instrument's line of sight (LoS) are determined by a directional half cone angle of a given line and a rotation around this line.

ARIEL					
Parameter	**LoS (arcsec)**	**Around LoS (arcmin)**	**Δt**	**Statistical Interpretation**	**Probability (%)**
Absolute Pointing Error (APE) in coarse pointing mode	10.0	1.0		Ensemble	99.7
Absolute Pointing Error (APE) in fine pointing mode	1.0	1.0	-	Temporal	99.7
Relative Pointing Error (RPE) in fine pointing mode	0.23	0.3	10 h	Temporal	99.7
XMM-Newton					
Parameter	**LoS (arcsec)**	**Around LoS (arcmin)**	**Δt**	**Statistical Interpretation**	**Probability (%)**
Absolute Pointing Error (APE)	30.0	1.0	-	Ensemble	95.0
Relative Pointing Error (RPE)	6.0	5.0	2 min	Temporal	95.0
Pointing Drift Error (PDE)	5.0	6.0	1 h	Mixed	95.0
Absolute Knowledge Error (AKE)	10.0	6.0	-	Ensemble	99.7
PLATO					
Parameter	**LoS (arcsec)**	**Around LoS (arcmin)**	**Δt (sec)**	**Statistical Interpretation**	**Probability (%)**
Absolute Pointing Error (APE)	270	9.0	-	Ensemble	99.7
Pointing Drift Error (PDE)	3.0	0.1	3 months	Mixed	99.7
Relative Pointing Error (RPE)	0.8	0.03	2.5 s	Temporal	95.0

Table 9: Examples of Pointing Requirements Formulation

5.3. GROUND STATIONS

The reference for ground stations is the ESA ESTRACK network. This network is currently in evolution, with some 15 metre stations being retired from service or handed over to third parties. Considering the mission timescale, the following stations should 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 10: Available ESTRACK Core Network Ground Stations

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

Stations from the Augmented Network (consisting of commercial antennas, Table 11) 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 11: ESTRACK Augmented Network ground stations.

Stations from the Cooperative Network (consisting of antennas owned by Cooperating Space Agencies) can also be considered (preferably as back-ups only or during critical operations such as LEOP).

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

2	4-Apr-2012	CEBREROS-1 (X / X Ka)	KIRUNA-1 (S / S X)	KIRUNA-2 (S / S X)	KOUROU-1 (S X / S X)	MALARGÜE-1 (X / X Ka)	MASPALOMAS-1 (S X / S X)	NEW NORCIA-1 (S X / S X)
3	TERMINAL	CEB1	KIR1	KIR2	KRU1	MLG1	MSP1	NNO1
4	Longitude	04 deg 22' 03.18" W	20 deg 57' 51.57" E	20 deg 58' 00.77" E	52 deg 48' 16.79" W	69 deg 23' 53.51" W	15 deg 38' 01.68" W	116 deg 11' 29.40" E
5	Latitude	40 deg 27' 09.68" N	67 deg 51' 25.66" N	67 deg 51' 30.34" N	5 deg 15' 05.18" N	35 deg 46' 33.63" S	27 deg 45' 46.40" N	31 deg 02' 53.61" S
6	Altitude [m]	794.095	402.1724	400.6815	-14.6709	1550.00	205.1177	252.2558
7	Antenna Diameter [m]	35	15	13	15	35	15	35
8	S-band Beamwidth [deg]	N/A	Rx: 0.60 Tx: 0.65	Rx: 0.65 Tx: 0.70	Rx: 0.60 Tx: 0.65	N/A	Rx: 0.60 Tx: 0.65	Rx: 0.28 Tx: 0.30
9	X-band Beamwidth [deg]	Rx: 0.064 Tx: 0.074	Rx: 0.16	Rx: 0.19	Rx: 0.16 Tx: 0.18	Rx: 0.064 Tx: 0.074	Rx: 0.16 Tx: 0.18	Rx: 0.064 Tx: 0.074
10	Ka-band Beamwidth [deg]	Rx: 0.017	N/A	N/A	N/A	Rx: 0.017	N/A	N/A
11	Antenna Speed [deg/s]	Az: 1.0 deg/s El: 1.0 deg/s	Az: 15.0 deg/s El: 5.0 deg/s	Az: 12.0 deg/s El: 7.5 deg/s	Az: 15 deg/s El: 5 deg/s	Az: 1.0 deg/s El: 1.0 deg/s	Az: 15 deg/s El: 5 deg/s	Az: 1.0 deg/s El: 1.0 deg/s
12	Azimuth Range [deg]	0 to 540	0 to 720	+/- 400	0 to 720	0 to 540	0 to 720	0 to 480
12	Elevation Range [deg]	0 to 90	-1 to 181	-3 to 182	-1 to 181	0 to 90	-1 to 181	0 to 90
14	Search / Acquisition Aid	IFMS Search	Search	Datron search	Search / Acq aid (X)	IFMS Search	Search / Acq aid (X)	IFMS Search / Perth Steering
15	Tilt Facility	NO	EAST / WEST	EAST / WEST	NO	NO	NO	NO
16	Tracking Mode	Program / ConScan	Auto (S) / Program	Auto (S X) / Program	Auto (S X) / Program	Program / ConScan	Auto (S X) / Program	Program / Slave / ConScan
17	Angular Data Accuracy (autotrack+pointing error)	N/A	80 mdeg	100 mdeg	80 mdeg	N/A	80 mdeg	N/A
18	FUNCTIONALITIES		Common backend for KIR1/2					
19	TM/TC Standards	PCM, CCSDS	PCM, CCSDS	PCM, CCSDS	PCM, CCSDS	PCM, CCSDS	PCM, CCSDS	PCM, CCSDS
20	TM/TC Redundancy	YES	YES	YES	YES	YES	YES	YES
21	Comms Redundancy	YES	YES	YES	YES	YES	YES	YES
22	Ranging	IFMS compliant	IFMS & CORTEX compliant	IFMS & CORTEX compliant	IFMS compliant	IFMS compliant	IFMS & CORTEX compliant	IFMS compliant
23	Doppler	YES	YES	YES	YES	YES	YES	YES
24	Meteo	YES	YES	YES	YES	YES	YES	YES
25	Autotrack Antenna Angles	NO	YES	YES	YES	NO	YES	NO
26	Open Loop recording	YES	NO	NO	YES	YES	NO	YES
27	Delta-DOR	YES	NO	NO	YES	YES	NO	YES
28	Frequency & Timing	MASER	GPS / USO (Crystal)	GPS / USO (Crystal)	MASER	MASER	CESIUM	MASER
29	UPLINK							
30	S-band TX band [MHz]	N/A	2025-2120	2025-2120	2025-2120	N/A	2025-2120	2025-2120
31	S-band Polarization	N/A	RHC, LHC	RHC, LHC	RHC, LHC	N/A	RHC, LHC	RHC, LHC
32	S-band EIRP [dBm]	N/A	101	99	111.2 (SHPA) 104.7 (SLPA)	N/A	102.1 (SLPA)	127.8 (SHPA) 112.1 (SSPA)
33	X-band TX band [MHz]	7145 - 7235	N/A	N/A	7145-7235	7145 - 7235	7145-7235	7145 - 7235
34	X-band Polarization	RHC, LHC	N/A	N/A	RHC, LHC	RHC, LHC, LINEAR	RHC, LHC	RHC, LHC
35	X-band EIRP [dBm]	138 (XHPA) 128 (XLPA) 122 (XSPA)	N/A	N/A	112.8	138 (XHPA) 128 (XLPA) 122 (XSPA)	112.8	138 (XHPA) 128 (XLPA)
36	Ka-band TX band [MHz]	N/A	N/A	N/A	N/A	N/A	N/A	N/A
37	Ka-band Polarization	N/A	N/A	N/A	N/A	N/A	N/A	N/A
38	Ka-band EIRP [dBm]	N/A	N/A	N/A	N/A	N/A	N/A	N/A
39	Modulation Schemes	IFMS compliant	IFMS & CORTEX XL compliant	IFMS & CORTEX XL compliant	IFMS compliant	IFMS compliant	IFMS & CORTEX SQPN compliant	IFMS compliant
40	Subcarrier Freq. [kHz]	8 or 16 kHz	8 or 16 kHz	8 or 16 kHz	8 or 16 kHz	8 or 16 kHz	8 or 16 kHz	8 or 16 kHz
41	TC data Rates [kb/s]	Remnant carrier: 4 kb/s SPL mode: 256 kb/s	Remnant carrier: 4 kb/s SPL mode: 256 kb/s	Remnant carrier: 4 kb/s SPL mode: 256 kb/s	Remnant carrier: 4 kb/s SPL mode: 256 kb/s	Remnant carrier: 4 kb/s SPL mode: 256 kb/s	Remnant carrier: 4 kb/s SPL mode: 256 kb/s	Remnant carrier: 4 kb/s SPL mode: 256 kb/s
42	DOWNLINK							
43	S-band RX band [MHz]	N/A	2200-2300	2200-2300	2200-2300	N/A	2200-2300	2200-2300
44	S-band Polarization	N/A	RHC, LHC	RHC, LHC	RHC, LHC	N/A	RHC, LHC	RHC, LHC
45	S-band G/T [dB/K]	N/A	27.7 (at 5 deg El)	21.4 (at 5 deg El)	29.1	N/A	29.2	37.5
46	X-band RX band [MHz]	8400 - 8500	8025-8500	7600-8500	8025-8500	8400 - 8500	8025-8500	8400 - 8500
47	X-band Polarization	RHC, LHC	RHC, LHC	RHC, LHC	RHC, LHC	RHC, LHC, LINEAR	RHC, LHC	RHC, LHC
48	X-band G/T [dB/K]	50.8 (at 10 deg El)	36.9 (at 5 deg El)	35.6. (at 5 deg El)	41	50.8 (at 10 deg El)	37.5	50.1
49	Ka-band RX band [MHz]	31800 - 32300	N/A	N/A	N/A	31800 - 32300	N/A	N/A
50	Ka-band Polarization	RHC, LHC	N/A	N/A	N/A	RHC, LHC	N/A	N/A
51	Ka-band G/T [dB/K]	55.8 (at 10 deg El)	N/A	N/A	N/A	55.7 (at 10 deg El)	N/A	N/A
52	Modulation Schemes	IFMS compliant	IFMS & CORTEX XL compliant	IFMS & CORTEX XL compliant	IFMS compliant	IFMS compliant	IFMS & CORTEX SQPN compliant	IFMS compliant
53	Carrier Freq Search Range	+/- 1.5 MHz	+/- 1.5 MHz	+/- 1.5 MHz	+/- 1.5 MHz	+/- 1.5 MHz	+/- 1.5 MHz	+/- 1.5 MHz
54	Subcarrier Frequency	2 kHz to 1.2 MHz	1.2 kHz to 2 MHz	1.2 kHz to 2 MHz	2 kHz to 1.2 MHz	2 kHz to 1.2 MHz	1.2 kHz to 2 MHz	2 kHz to 1.2 MHz
55	TM Data Rates	High Speed IFMS compliant: - 1.2 Mb/s (RCD) - 8 Mb/s (SCD) - 16 Mb/s (GMD)	High Speed IFMS: - 1.2 Mb/s (RCD) - 8 Mb/s (SCD) - 16 Mb/s (GMD) Cortex XL compliant: - 256 Kb/s (subcarrier) - 40 Mb/s (Direct PCM) X-band: Up to 100 Mb/s	High Speed IFMS: - 1.2 Mb/s (RCD) - 8 Mb/s (SCD) - 16 Mb/s (GMD) Cortex XL compliant: - 256 Kb/s (subcarrier) - 40 Mb/s (Direct PCM) X-band: Up to 100 Mb/s	High Speed IFMS: - 1.2 Mb/s (RCD) - 8 Mb/s (SCD) - 16 Mb/s (GMD)	High Speed IFMS compliant: - 1.2 Mb/s (RCD) - 8 Mb/s (SCD) - 16 Mb/s (GMD)	High Speed IFMS: - 1.2 Mb/s (RCD) - 8 Mb/s (SCD) - 16 Mb/s (GMD) Cortex SQPN (ATV): - 1 kbps to 500 kbps - PN Chip Rates 10 kcps to 6.25 Mcps	High Speed IFMS: - 1.2 Mb/s (RCD) - 8 Mb/s (SCD) - 16 Mb/s (GMD)
56	Data Coding Scheme	R-S, Convolutional, Concatenated, Turbo	R-S, Convolutional and Concatenated	R-S, Convolutional and Concatenated	R-S, Convolutional, Concatenated, Turbo	R-S, Convolutional, Concatenated, Turbo	R-S, Convolutional, Concatenated, Turbo	R-S, Convolutional, Concatenated, Turbo
57	INTERFACES							
58	TM/TC Connectivity	TCP/IP SLE (TMTCS)	TCP/IP SLE (CORTEX/TMTCS)	TCP/IP SLE (CORTEX/TMTCS)	TCP/IP SLE (TMTCS)	TCP/IP SLE (TMTCS)	TCP/IP, ATV TC SLE (TMTCS/CORTEX)	TCP/IP SLE (TMTCS)
59	Rng/Dop Connectivity	FTP (IFMS)	FTP (IFMS / CORTEX via CSMC)	FTP (IFMS / CORTEX via CSMC)	FTP (IFMS)	FTP (IFMS)	FTP (IFMS)	FTP (IFMS)
60	Meteo Connectivity	FTP (IFMS)	FTP (IFMS)	FTP (IFMS)	FTP (IFMS)	FTP (IFMS)	FTP (IFMS)	FTP (IFMS)
61	Angles Connectivity	N/A	IP Via CSMC	IP Via CSMC	IP via STC	N/A	IP via STC	N/A
62	Pointing Format	STDM	STDM	STDM	STDM	STDM	STDM	STDM
63	Tracking Interface (ESOC)	FDS	FDS	FDS	FDS	FDS	FDS	FDS

Table 12: Some Ground Stations characteristics

6. MISSION IMPLEMENTATION CONSTRAINTS

6.1. SPACE DEBRIS MITIGATION

All ESA missions must ensure that no orbital debris will contaminate the so-called protected regions (in yellow in Figure 5). This implies that:

1. All ESA Space Vehicles including Satellites, Launchers and Inhabited Vehicles must be disposed of;
2. At the end of life, they must be out of the protected regions within 25 years;
3. They must either be moved to non-protected regions or re-enter into Earth atmosphere for break-up and burning;
4. Uncontrolled re-entry is not allowed if the casualty risk > 10^{-4} (the case of large S/C);
5. If drift to non-protected regions or re-entry does not happen naturally, active (propulsive) removal needs to be foreseen.

In most cases, a propulsion disposal manoeuvre at end of life will be needed to comply with the above requirements. Such manoeuvre must inject the S/C into an orbit such that:

- It does not cross the GEO protected region for at least 100 years with a probability >90%;
- It allows spacecraft safe re-entry in the atmosphere within a maximum duration of 25 years.

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

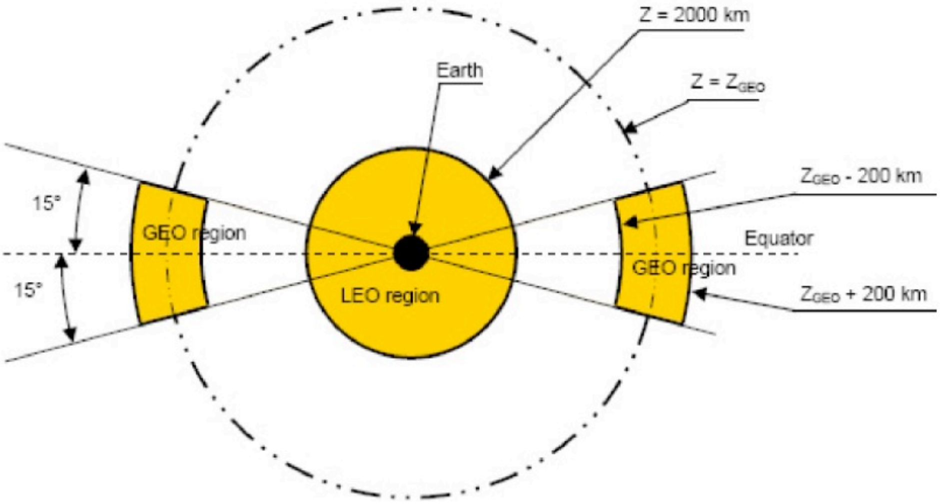

Figure 5: LEO and GEO protected regions.

When fragments of the S/C may survive the re-entry (which can happen for the case for large spacecraft), a controlled re-entry manoeuvre has to be performed to mitigate the risk of ground casualty. The ΔV required for this manoeuvre must be included in the sizing of the propulsion subsystem. This requirement applies to the S/C as well as any other large debris generated by the mission, such as upper stages of the launch vehicle, multi-S/C adapters, ejected covers etc.

However, for the re-entry of modest size objects (mass typically below 1000 kg), an un-controlled re-entry is acceptable, provided it happens within 25 years.

For a LEO mission, as a worst-case estimate, the re-entry 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.

6.2. PLANETARY PROTECTION

ESA planetary missions must comply with the categories and associated requirements reported in [ND3].

6.3. TECHNOLOGY READINESS

6.3.1. Technology Readiness requirements – F mission

The overall spacecraft development must be compatible with a fast implementation schedule, consisting of 2 to 3 years for the preparation phase (phases 0/A/B) and 4 to 5 years for the development phase (C/D).

To achieve this, the platform will have to rely on existing technologies and will likely have to be derived from existing, flight-proven platforms, aiming at maximising reuse. Therefore, TRL 5/6 is required at the time of the mission proposal for all platform elements. TRL 7 is also 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 must be compatible with a fast implementation schedule, with typically 2 years available for focused pre-developments during the preparation phases (phases 0/A/B) and 3 years for the development phase (phases C/D). Therefore, all payload elements should be at TRL ≥ 5 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 will have to present a back-up scenario with TRL ≥ 5 and lower performance.

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

- Reference to heritage must 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.

6.3.2. Technology Readiness requirements – M mission

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

TRL 5/6 is required for all mission elements (platform and payload) by the mission adoption (end of Phase B1). However, for schedule critical elements, TRL ≥ 5 should be reached at the end of the Phase A, since the Technology Readiness will be a major element for the mission selection.

The spacecraft can be a new development but must rely on substantial heritage. The mission should not contain any element with TRL < 4 at the time of the proposal. If elements at TRL 4 are critically needed for achieving the mission science goals, the proposer must 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 must be considered:

- Reference to heritage must 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.

6.4. EXPORT CONTROL

In case the mission is planned with international partners, due consideration must be paid to export control regulations, in particular the US International Traffic in Arms Regulation (ITAR), Export Regulations Administration (EAR) and European export rules. Such regulations may prevent or put major constraints on important mission activities (such as satellite design, assembly, testing and launch).

7. PROGRAMMATIC ASSUMPTIONS

7.1. RESPONSIBILITIES

In most Science missions, payload elements are provided by consortia of scientific institutions under the responsibility and with funding from the Member States. The share of responsibilities between ESA and the Member States on the payload elements must be clearly identified in the proposal.

As a rule, any Member State payload provision must be commensurate with the lead Member State funding capability. For the missions resulting from the present Call, large payloads will be managed by ESA, who will also be responsible for the system engineering and the system AIV (with the relevant costs to be accounted in the ESA CaC) .Direct Member State provisions must be limited to modestly sized payload-elements, which can either be complete, modestly-sized instruments (as is typically the case for instruments on planetary probes) or specific payload elements, e.g., focal plane instruments, filter wheel assemblies, etc. As a rough guideline, “modestly-sized” implies with mass not exceeding approximately 50 kg, although of course this will be dependent on the element’s complexity. Large, complex payloads proposed entirely under the responsibility of the Member States, without the foreseen ESA role in the management and system engineering and AIV, will not be considered feasible for the purpose of the present Call.

ESA can support the detailed design and pre-developments for the payload prior to adoption, with the goal of securing the payload development schedule. Details of the possible funding scheme will be discussed with successful phase-1 proposers.

For most ESA-led missions, the launch services, the mission operations (MOC) and the science operations (SOC) are carried out by ESA, in most cases with contributions from the Member States for the SOC. Different schemes can be proposed, their feasibility will be assessed based on the proposal content.

7.2. MISSION REFERENCE SCHEDULE

Tables 13 and 14 provide the reference schedule to be assumed respectively for the F and M missions.

Event	Date or duration	Note
Start of Phase 0	Q1 2023	
Mission Adoption	Q1 2026	At the end of Phase B
Launch	2030-2031	Will depend on the mission
Nominal in-orbit operations	Typically ~2 years	Operation costs must be included in the ESA CaC

Table 13: Reference schedule for the F mission. The duration of the nominal operations does not include the disposal phase (if applicable).

Event	Date or duration	Note
Start of phase 0	Q1 2023	
Mission selection	2026	At the end of the Phase A
Mission adoption	2029	At the end of the Phase B1
Launch	By 2037	Will depend on the mission
Nominal in-orbit operations	Typically ~3 years	Operation costs must be included in the ESA CaC

Table14: Reference schedule for the M mission. The duration of the nominal operations does not include the disposal phase (if applicable).

7.3. MISSION COST ELEMENTS

The ESA Cost at Completion (CaC) for the M mission is 550 M€, for the F mission 175 M€. The CaC covers all ESA activities following the mission adoption, i.e.,

- 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 15 and 16 provide an indicative cost breakdown for an ESA mission with no international partners, for the F mission with a Vega-C launch and for the M mission with an Ariane 62 launch.

Element	% of total CaC
Space segment under ESA responsibility	42%
Launch Vehicle	26%
Operations (MOC and SOC)	10%
ESA Project	10%
Margin	12%

Table 15: Indicative cost breakdown for a F mission using Vega-C. The space segment includes procurement of the spacecraft together with, if applicable, the management and system activities on the payload.

Element	% of total CaC
Space segment under ESA responsibility	44%
Launch Vehicle	16%
Operations (MOC and SOC)	14%
ESA Project	14%
Margin	12%

Table 16: Indicative cost breakdown for a M mission using Ariane 62. The space segment includes procurement of the spacecraft together with, if applicable, the management and system activities on the payload.

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
CDR	Critical Design Review
CoG	Centre of Gravity
DHS	Data Handling System
DLS	Dual Launch Structure
DSN	Deep-Space Network
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
S/C *or* 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
SSMS	Small Spacecraft Mission Service
SSO	Sun Synchronous Orbit
SVM	Service Module
TBC	To Be Confirmed
TBD	To Be Defined
TM	Telemetry
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.
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.

Technology Readiness Level	**Milestone achieved for the element**	**Work achievement (documented)**
		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.