DOCUMENT

F Mission Call - Technical Annex

Prepared by	**SCI-F**
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1 INTRODUCTION

1.1 Scope of document

This Annex provides technical guidelines for the proposals in answer to the Call for an F mission, to help proposers to define mission profiles that are compatible with the programmatic boundaries for F missions.

The ceiling cost to ESA (Cost at Completion, CaC) for missions solicited by the Call is 150 M€ (in 2018 economic conditions). The mission will be launched in 2028 together with the ARIEL mission requiring mission adoption in 2022.

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

1.2 Reference documents

[RD1] Small Planetary Platforms CDF Study Report, CDF-178, January 2018

[RD2] ECSS-E-HB-11A DIR1, TRL guidelines, www.ecss.nl, 2016.

[RD3] ESA Tracking Stations (ESTRACK) Facilities Manual (EFM), DOPS-ESTR-OPS-MAN-1001-OPS-ONN Issue 2.0, 2017.

[RD4] Ariane 6 User's Manual, Issue 1 Rev 0, March 2018.

[RD5] Space Debris Mitigation Policy for Agency Projects, IPOL(2014)2.

1.3 List of acronyms

APE	Absolute Performance Error
AU	Astronomical Unit
CaC	Cost at Completion
DV	Delta V (also ΔV)
ECSS	European Cooperation for Space Standardisation
ESA	European Space Agency
ESOC	European Space Operations Centre
GEO	Geostationary Earth Orbit
GTO	GEO Transfer Orbit
HEO	High Elliptical Orbit
HGA	High Gain Antenna
ISO	International Standards Organisation

LEO	Low Earth Orbit
LLO	Low Lunar Orbit
LV	Launch Vehicle
MOC	Mission Operations Centre
MS	Member States
NEO	Near Earth Object
PL	Payload
RF	Radio Frequency
RPE	Relative Performance Error
S/C	Spacecraft
SEL2	Sun-Earth Lagrange point 2
SEP	Solar Electric Propulsion
SES angle	Sun-Earth-Spacecraft angle
SOC	Science Operations Centre
SPP	Small Planetary Platforms
TBC	To Be Confirmed
TBD	To Be Defined
TM	Telemetry
TRL	Technology Readiness Level
TT&C	Telemetry, Tracking and Command Systems

2 F-CALL BOUNDARIES

The general F-Call boundaries are summarised in the Table 1 and further expanded in the following sections.

Item	Requirement	Comment or Guidelines
ESA CaC	≤ 150 M€	Includes all elements to be funded by ESA except for the launch services. Excludes Member State and international partner contributions.
Science objectives	None. The science objectives of this mission are open.	The science instruments shall be defined in relation to the 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 in the definition phase.
Launcher	The F mission will be launched together with the ARIEL spacecraft to the Sun-Earth L2 Lagrange point (SEL2) with an Ariane 6.2 launcher	The likely scenario is to launch in a stacked configuration, with ARIEL mounted on top of the F mission spacecraft. This may constrain elements on the top-floor of the F mission spacecraft. A launch with PLATO is not excluded (with similar constraints)
Spacecraft wet mass	< 1000 kg	Hard constraint imposed by the launch strategy. The proposers are invited to ensure that adequate margins and/or descoping options are available for achieving the mass. The spacecraft wet mass encompasses the platform with its propulsion subsystem(s), the scientific instrumentation, and the daughtercraft if any (smallsats, surface package etc). The launcher adapter is excluded.
Overall science payload mass	< 80 kg	The payload mass limit is related to the overall cost and schedule constraints.

		The actual payload mass can be lower depending on the mission profile (see sections 3 and 4). A fast and reliable payload development schedule, typically 3-3.5 years starting from the mission adoption, will be needed.
Platform TRL	TRL ≥ 5/6 by mission selection in Q1 2020.	ISO scale, see appendix A. As a general rule, the platform equipment shall be at TRL ≥ 7 (space qualified for the mission needs and available) by the mission adoption. TRL 6 is required by the time of the mission selection (TRL 5 may be acceptable on selected items) since technology readiness will be a key element of the decision process.
Science Payload TRL	TRL ≥ 5/6 by mission selection in Q1 2020.	The credibility of the payload development schedule will be an important selection criterion. The proposers are encouraged to make use of existing instruments where possible, eventually with limited adaptations. 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 Preliminary Design Review (PDR) status before the mission adoption. ESA is ready to support the instrument detailed design and pre-developments during this phase for securing the payload development schedule. The 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 from the international partner is available.	The F-mission must be ESA-led. Any international cooperation scheme that respects this constraint is acceptable.
Spacecraft operations	Nominal duration of science operations typically < 2 years	The spacecraft in-orbit operations are ESA-led (with the relevant costs accounted in the CaC), but can include contributions from the Member States or partners. The nominal duration of science operations does not include the cruise phase. Longer cruise phases can be envisaged when combined with shorter science operations (e.g. in situ for small bodies, see examples in section 3)

Table 1: Summary of the F Mission Call general guidelines.

3 POSSIBLE MISSION PROFILES AND GENERAL GUIDELINES

3.1 Examples of mission profiles

Table 2 provides key parameters for a set of potential target orbits or destinations. The list is not meant to be exhaustive, and it provides an indicative assessment of the maximum achievable performance. All figures should be viewed as preliminary and will need consolidation through detailed studies, with the actual achievable performance constrained by the cost ceiling and resulting “design to cost” approach.

Potential target orbit from SEL2	Max. distance to Sun	DeltaV to reach target orbit	Expected Transfer time	Indicative nominal Science Operations duration	Delivered mass at target orbit
Main asteroid belt (inner ring)	<2.5 AU	~7 km/s (with Earth and Moon gravity assists)	~5 years	6 months	~800 kg
Near Earth Object	<1.5 AU	~3 - 5km/s	~2-3 years	~6 months	~700-800kg
Multi target for Near Earth Object up to 3km/s	<1.5 AU	~3 km/s explorer up to 1km/s	~ 2 years to first target and max 6 months to second target	~6 months at main target and 3 months at second target	~800kg
Multi target for Near Earth Object up to 5km/s	<1.5 AU	~5 km/s explorer up to 1 km/s	~ 3 years to first target and max 6 months to second target	~6 months at main target and 3 months at second target	~700kg
Venus	<1.1 AU	~7 km/s into 4-day HEO (500kmx186000km)	2.5-3.5 years	~1.5-2 years	~800 kg
Mars	<1.67 AU	5.3-7.3 km/s into 4sols HEO (~500kmx96,000km)	~2.3-3.3 years	~1.5-2 years	~790-850 kg
Phobos	<1.67 AU	6.5 km/s - 8.5km/s	~3-4 years	~1.5-2 years	~760-810 kg
L2	~1 AU	launcher dispersion correction and disposal required	-	~1.5-2 years	~ 980kg
Heading/trailing	<1.2 AU short transfer, <1.1 AU long transfer	2.1-3.5 km/s	1.5-2.5 years	~1.5-2 years	~750-850 kg
Moon orbit	~ 1 AU	0.5 km/s before reduction, 2 km/s into LLO	0.75-1 years before reduction, 1.25-1.5 years into LLO	~1.5-2 years	950 kg before reduction, 850 kg into LLO
HEO	1 AU	1.3 km/s into $6R_Ex26R_E$ 1.9 km/s into $6R_Ex15R_E$	~10-12 months	~1.5-2 years	860 kg into $6R_Ex15R_E$, 900 kg into $6R_Ex26R_E$

Table 2: Mission Examples

Note: Except for L2 case, all "delivered mass at target orbit" presented in Table 2 have been estimated assuming trajectories and transfer durations with an electric propulsion system (including appropriate margins).

Notes:

- Given the mass ceiling of 1000 kg, the maximum achievable Delta-V is ~ 7 km/s, with an electric propulsion system. For planetary missions, this performance can in principle enable missions to a Near Earth Object (NEO), Mars and its moons and

Venus. Making use of Moon and/or Earth gravity assists, the inner region of the main asteroid belt can be reached, up to 2.5 AU from the Sun.

- Targets orbits beyond 2.5 AUs from the Sun are not reachable due to Delta-V and power constraints imposed by the mass ceiling.
- Sample return mission concepts are unlikely to be feasible within the given cost and schedule constraints.
- The mission analysis will be optimised, and could include, e.g., injection into the transfer orbit before arrival to SEL2 or waiting in orbit around SEL2 for the desired transfer window to open.
- The need for ground commanding of manoeuvres during the science operations phase should be minimised in order to minimize the costs of operations. The science operations strategy will need to be defined based on this constraint.
- The balance between ground commanding and on-board autonomy will need to be optimised depending on the proposed mission concept and transfer/operational orbits.

3.2 Data transmission and link budget considerations

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

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

where:

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

The above formula does not take into account limitations that may result from international regulations.

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 35m ground antennas) are indicated in Figure 1.

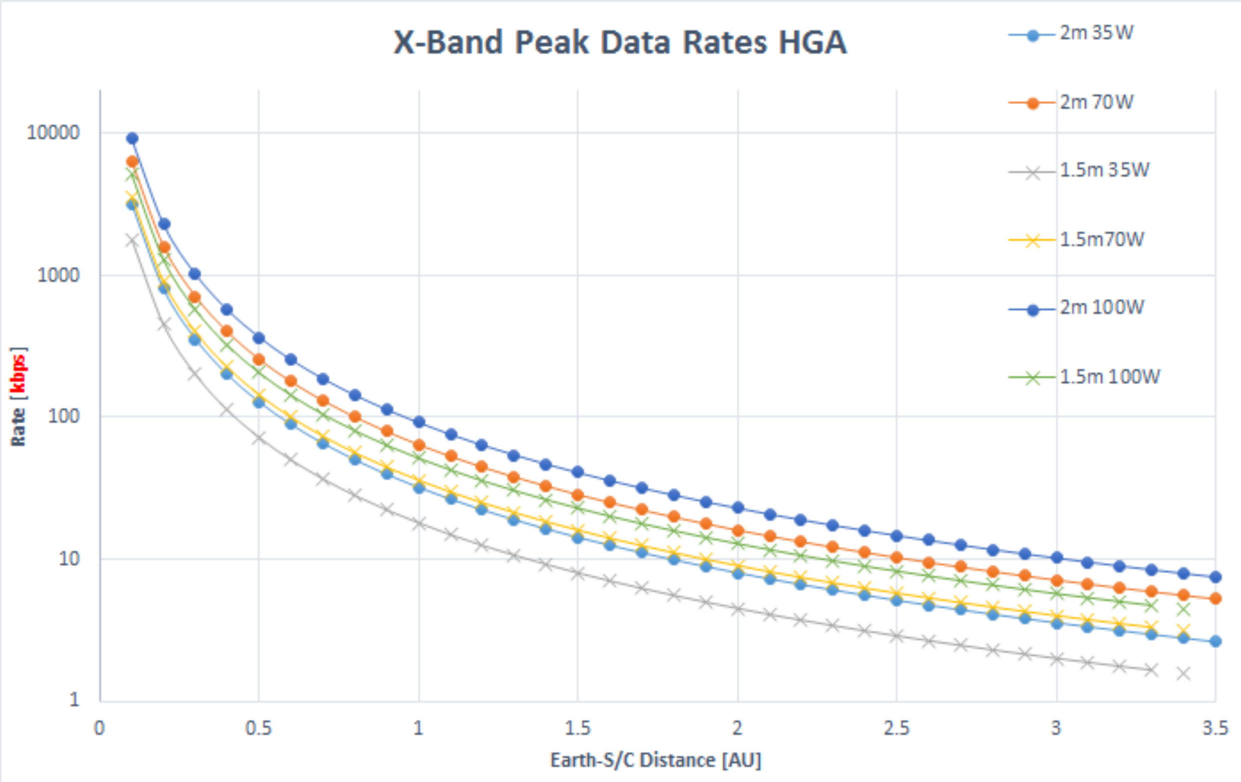

Figure 1: TT&C TM data rates as a function of key parameters

3.3 Ground station characteristics

The reference for ground stations is the ESA ESTRACK network (Table 3 below and RD3). This network is currently evolving, with some 15 m stations being retired from service or handed over to third parties.

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

Table 3: ESTRACK Core Network ground stations

3.4 Space debris regulations

All ESA missions (see [RD5]) have to ensure that no additional orbital debris will contaminate the protected regions (in yellow in Figure 2). This implies the need to implement a propulsion capability for a S/C operating in the LEO or GEO protected regions, for either moving the S/C into graveyard orbits at its end of life, or ensuring its re-entry in the atmosphere within 25 years.

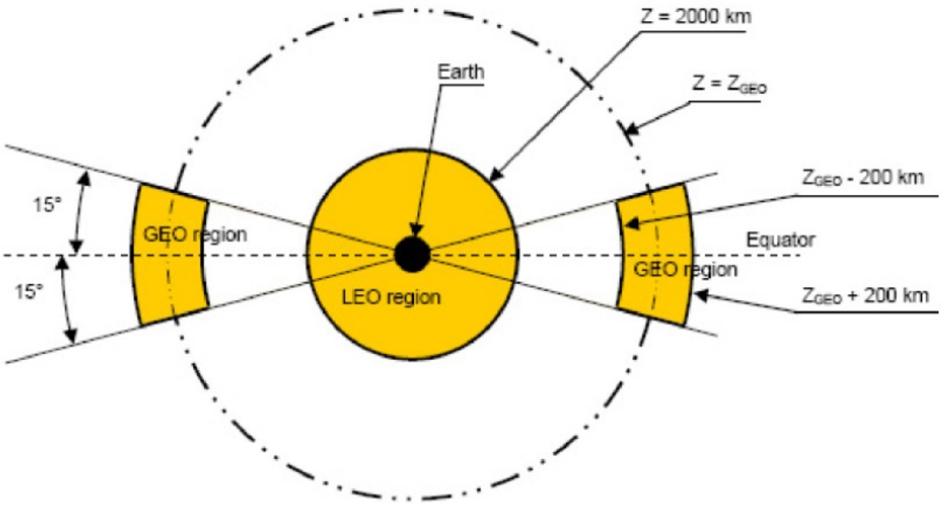

Figure 2: LEO and GEO protected regions

When fragments of the S/C may survive the re-entry (typically for large missions), a controlled re-entry manoeuvre has to be performed to mitigate the risk of ground casualty. For small missions (typically < 1000 kg), an un-controlled re-entry is acceptable, as long as it happens within 25 years.

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

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

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

4 MOTHERCRAFT/DAUGHTERCRAFT CONFIGURATIONS

4.1 Introduction

The F mission Call is open to mission concepts involving multiple spacecraft or smallsats, aiming at innovative missions with multi-point measurement capabilities. The cost and schedule requirements impose clear constraints on such concepts. This section provides recommendations and guidelines to the proposers for enabling feasible proposals with multi-point measurement capabilities. The data provided here are largely resulting from the Small Planetary Platforms (SPP) ESA internal study recently conducted [RD1] analyzing missions to small bodies based on a mothercraft/daughtercraft architecture. Two mission concepts were studied as reference cases: a mission to a non-active body (Near Earth Object, NEO) and a mission to an active body in the main asteroid belt.

Proposals complying with the figures indicated in Table 4 (such as the number of smallsats and related mass depending on the target; payload mass per probe; pointing capabilities, etc.) will likely result in a technically feasible concept, that will in any case need to be confirmed through the Phase 0 and the following definition phase. Conversely, Table 4 should not be viewed as a strict prescription. Furthermore, compliance with the figures indicated in Table 4 does not ensure compliance with cost and schedule constraints, which will need to be confirmed on a case by case basis.

4.2 Mothercraft/daughtercraft mission examples

Table 4 summarises the main parameters for some example missions that could be considered based on the mothercraft/daughtercraft architecture. The mass figures are largely derived from [RD1]. For the mothercraft/daughtercraft configuration, proposers should pay attention to the mass budget and to the payload mass and development schedule. Reasonable mass for daughtercraft is estimated to be in the range 30-40 kg, depending on the functional requirements. A surface package may complement/replace the daughtercraft in some cases, if compatible with the mass constraint and programmatic boundaries (see also 4.3.3).

For missions to small bodies (NEO or main asteroid belt), the proposers should indicate at least two target body candidates that would enable the core science objectives of the mission.

Potential target orbit from SEL2	Max. distance to Sun	Indicative number of daughtercraft (constrained for mass and/or cost reasons)	Indicative Science Payload Mass allocation and distribution	Science Payload Maximum Power in the daughtercraft	Remark
Main asteroid belt (inner ring)	<2.5 AU	2	Total P/L mass typically 20-25 kg e.g. 4-8 kg on Mothercraft, and 6-8 kg per daughter	~ 20 W	Overall mass highly constrained by the DV. Mass figures assume high impulse electrical propulsion for the Mothercraft (3600 s); A small surface package could also be considered, subject to mass compatibility (see also 4.3.3). Power constrained by distance to the Sun.
Near Earth Object	<1.5 AU	2-4	Total P/L mass typically 30-40 kg, e.g. 4-8 kg in Mothercraft and 6-8kg per daughter	~90 W	A small surface package could replace one of the daughtercraft (see also 4.3.3)
Multi target for Near Earth Object up to 3km/s	<1.5 AU	2-3 (one of which is assumed "explorer" to the second target)	Total P/L mass typically 20-25 kg e.g. 4-8 kg in Mothercraft ~ 6 kg per daughter	~90 W	P/L mass assumes 3 daughtercraft, one of which is exploring a second target. The explorer daughter needs 7-8 kg extra mass for the electric propulsion (DV < 1 km/s) and for enabling the communication with the mothercraft (stay ing at the first target). Other concepts/combinations are possible, e.g. 2 daughtercraft only, but both visiting the second target.
Multi target for Near Earth Object up to 5km/s	<1.5 AU	2 (one of which is an "explorer" to the second target)	Total P/L mass typically < 20 kg e.g. 4-8 kg in Mothercraft ~ 6 kg per daughter	~90 W	Comparable to previous case, with higher mass constraint induced by the DV to reach the first target.
Venus	<1.1 AU	2	4-8 kg in Mothercraft 6-8 kg per daughter	~90W	Assumes high impulse (~3600 s) electrical propulsion system. S/C design assumptions to be confirmed by dedicated study.
Mars	<1.67 AU	2-3	4-8 kg in Mothercraft 6-8 kg per daughter	~40W	Assumes high impulse (~3600 s) electrical propulsion system. S/C design assumptions to be confirmed by dedicated study.
Phobos	<1.67 AU	0-2	4-8 kg in Mothercraft 6-8 kg per daughter	~40W	Assumes high impulse (~3600 s) electrical propulsion system. S/C design assumptions to be confirmed by dedicated study.
L2	~1 AU	3	~50 kg in Mothercraft 6-8 kg per daughter	~90W	Overall payload mass is a cost/schedule driver
Heading/trailing heliocentric orbits and Sun-Earth L4/L5	<1.2 AU short transfer, <1.1 AU long transfer	4	4-8 kg in Mothercraft 6-8 kg per daughter	~90W	Long transfer reduces DV, adds +1 year. No Moon-Earth gravity assist assumed to leave L2. S/C design assumptions to be confirmed by dedicated study.
Moon orbit	~ 1 AU	4	4-8 kg in Mothercraft 6-8 kg per daughter	~90W	1500x60000 km capture orbit, then reduction to 300 km LLO considered. S/C design assumptions to be confirmed by dedicated study.
HEO	1 AU	4	4-8 kg in Mothercraft 6-8 kg per daughter	~90W	S/C design assumptions to be confirmed by dedicated study.

Table 4: Example missions with mothercraft/daughtercraft configuration

4.3 Design Guidelines for Mothercraft/Daughtercraft Missions

4.3.1 Mission/System level considerations

For limiting the development and operation costs, the mothercraft design should remain as simple as possible and focused on the two basic functions:

- Carrying the daughtercraft to their target/orbit destination
- Providing their data relay function back to Earth.

Therefore, the mothercraft should carry no (or very limited) scientific instrumentation, with a maximum suggested allocation of 8 kg, and platform performance or resources should be limited (see suggested values in Table 4).

For the same reasons, the daughtercraft should remain as simple as possible, featuring only the minimum required functions and performance to achieve the mission objectives. For example:

- The data relay function should be performed via the mothercraft, avoiding the need for the daughtercraft to incorporate a deep space communication package. Only an inter-satellite-link (ISL) should be considered for communications with the mothercraft and, potentially, between the daughtercraft themselves. The complexity and topology of the ISL will strongly depend on the specific requirements of each mission, the two most important being the size of the selected target and the operational orbits chosen for the mothercraft and the fleet of daughtercraft (e.g range and visibility).
- The reference propulsion system for the daughtercraft should be a basic cold gas system to perform close proximity operations and orbit maintenance (e.g. ~ 15-20 m/s delta-V capability) after separation from the mothercraft. As an option, and subject to mass and cost/schedule compatibility, the daughtercraft may include an auxiliary electric propulsion system for potentially visiting a second target in some mission scenarios (see e.g. the "explorer" case in the "Multiple target NEO options" of Table 4). In that case, it is recommended to not exceed a delta-V requirement of ~1 km/s for a transfer starting from the first target. Such delta-V may require ~ 7-8 kg extra mass for the smallsat propulsion and communication subsystems, possibly to the detriment of the scientific payload.

As a rule, the standard ECSS requirements are applicable to the entire spacecraft, including the daughtercraft. However, deviations from these requirements are allowable for the F mission concept and will be assessed on a case by case basis. As an illustrative example, Table 4 has been established by assuming no single point failures for the mothercraft, while tolerating single point failures for the probes, for maximising the payload mass availability. The underlying assumption is that a mission with multiple smallsats/probes would be conceived with some intrinsic failure tolerance, e.g. by providing a graceful degradation scheme and guaranteeing a core science return in case of a single probe failure. Conversely, robustness against single point failures could be enforced in some cases to the detriment of the useful payload mass.

4.3.2 Resources for the science instruments

This section provides a set of recommendations for the mothercraft and daughtercraft key resources, that will need revision and consolidation depending on mission specifics.

MOTHERCRAFT

The following allocations are recommended for the science instruments in the mothercraft:

- Maximum total mass of ~8 kg including all maturity and system margins of the instruments.
- Maximum total power of ~ 100 W.
- Maximum volume of ~ 5 dm^3, possibly accommodated in a single box or on various faces of the platform. The proposers will need to specify desired pointing direction (nadir, limb, etc.) and field of view so that an assessment of the accommodation feasibility can be performed.

The proposers should specify if the science instruments require deployment, antennas, etc., and if these are considered in the allocated volumes.

- Recommended pointing performance of the mothercraft:
    - Absolute Pointing Error (APE) < 100 arcsec (pointing accuracy, 3 sigma)
    - Relative Pointing Error (RPE) < 20 arcsec over 60 ms (pointing stability).

DAUGHTERCRAFT

The recommended allocations for the science instruments in each of the daughtercraft (all TBC during the Phase 0 study assessment) are:

- Maximum total mass of ~8 kg including all maturity and system margins of the instruments.
- Maximum total power of ~90 W at 1.1 AU from the Sun or ~ 20 W at 2.5 AU from the Sun.
- Maximum volume of ~ 5 U (i.e. ~ 5 dm^3), possibly accommodated on various faces of the platform. The proposers will need to specify desired pointing direction (nadir, limb, etc.) and field of view so that an assessment of the accommodation feasibility can be performed.

The proposers will specify if the science instruments envisaged required deployments, antennas, etc., and if these are considered in the allocated volumes.

- The following pointing performance can be envisaged for the daughtercraft:
    - Absolute Pointing Error (APE) < 360 arcsec (pointing accuracy, 3 sigma)
    - Relative Pointing Error (RPE) < 30 arcsec over 100 ms (pointing stability).

4.3.3 Spacecraft operation cost drivers

Operation costs for missions to small bodies can drive the overall cost determining the mission's feasibility. To contain them, proposers should avoid complex operations or operational scenarios. This will require:

- Compatibility of the space segment with existing ground segment infrastructure and mission operations concepts. The re-use of platform subsystems, at least for what concerns avionics, the communication subsystem, and the Reaction Control System, allows significant cost savings.
- For target bodies with a diameter of ~ 1 km or less, relatively simple, non-bound orbits for the mothercraft (e.g. ping-pong orbit trailing the body) and the fleet of daughtercraft (e.g. hyperbolic arcs around the target) should be considered. This will simplify the spacecraft design (eclipses could be avoided, range and visibility between mothercraft and daughtercraft ease telecommunication architecture, etc.) and the operational concept (no need for an accurate dynamic model or optical navigation for example), reducing the operations costs.
- For target bodies with a diameter larger than ~ 1 km, the mothercraft will likely need to be inserted in an orbit around the target (bound orbit), with significant implications on the spacecraft design (both for the mother and the daughters), their hardware and the operational concept, therefore at a higher cost.
- One of the probes could be a surface element package (of mass < 10 kg). However, it should be kept in mind that a lander delivery may require complex and demanding operations which increase the overall operation costs. Therefore, if a surface element is envisaged, the proposers should consider options for which high precision landing to a particular site is not necessary. Ideally, the lander operation sequence should minimally affect the spacecraft operations, e.g. by relying on autonomous landing and routine operations.
- Distances from the daughtercraft to the surface in the order of ~5-10 km should be considered for proximity operations around the target. Closer distances (down to ~ 1 km) can potentially be achieved in some cases and for specific passes. This is strongly linked to the size of the target and the knowledge of its mass/gravity field, which are two of the parameters that clearly dictate the complexity of the orbits around it. It is useful to keep in mind that the type of operations to be conducted in proximity of the target object is a major cost driver.
- Transfer trajectories and arrival operations at the target have to be designed taking into account the occurrence of solar conjunctions, during which communications will be disrupted (for Sun-Earth-Spacecraft – SES – angles < 3 deg) and orbit determination will be severely degraded. Critical operations are not possible for SES angles below 5 deg.
- Spacecraft operations will be defined and executed under ESA/ESOC overall responsibility as baseline. External contributions to the spacecraft operations are possible, e.g. for enabling more demanding operation scenarios within the CaC ceiling, but will have to be carefully discussed on a case by case basis for fitting the ESOC ground segment development scheme (therefore avoiding additional costs to ESA).

- Mission planning is the most effort-demanding process together with navigation. A lean planning process, with minimal interfaces is a key factor in containing the development and operations costs. Where feasible, operations planning for instruments should be integrated with the overall mission planning process using direct inputs from the science community.
- Mission profiles allowing the main part of the operations to be performed during normal working hours, possibly with repetitive pattern, will help containing operations costs. Critical operations, near real time operations, and short response times should be avoided or limited to the extent possible.

5 COST ASPECTS

The cost breakdown for an F mission will vary with the mission concept and also depend on the expected international and Member States contributions. An indicative breakdown is provided in Table 5.

Item	CaC fraction	Remark
Space segment	~65-70%	Includes all elements expected to be funded by ESA, e.g. platform, payload, smallsat(s) etc. Includes Phase E1. Excludes Member State and international partner contributions (if any)
ESA Project	~15%	Typically, 20% of the space segment costs
Spacecraft Operations (MOC and SOC)	~15-20%	Spacecraft operations will strongly depend on the mission profile
Launcher	0	Not to be accounted in the F mission CaC

Table 5: Approximate expected breakdown of the ESA cost for an F mission

Appendix A - TRL definition (ISO scale)

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

Technology Readiness Level	Milestone achieved for the element	Work achievement (documented)
		Model test reports.
TRL 7: Model demonstrating the element performance for the operational environment	Performance is demonstrated for the operational environment, on the ground or if necessary in space. A representative model, fully reflecting all aspects of the flight model design, is built and tested with adequate margins for demonstrating the performance in the operational environment.	Definition of performance requirements, including definition of the operational environment. Model definition and realization. Model test plan. Model test results.
TRL 8: Actual system completed and accepted for flight ("flight qualified")	Flight model is qualified and integrated in the final system ready for flight.	Flight model is built and integrated into the final system. Flight acceptance of the final system.
TRL 9: Actual system "flight proven" through successful mission operations	Technology is mature. The element is successfully in service for the assigned mission in the actual operational environment.	Commissioning in early operation phase. In-orbit operation report.

Summary definition of the ISO TRL levels (based on ISO 16290).