DOCUMENT

ASTEROID IMPACT MISSION: PAYLOAD INTERFACE DOCUMENT

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1 PURPOSE OF DOCUMENT

The purpose of this document is to describe interfaces between the Asteroid Investigation Mission (AIM) spacecraft and its payload in the context of the AIM Phase-A/B1. It contains information on all payload aspects influencing the assessment of the S/C during that phase.

2 REFERENCES

Reference Documents (RD) are applicable to this document only when specifically stated in the text and called up with references to specific parts of the document that are to be applicable. Otherwise the documents below are listed for information and as an aid for understanding.

[AD3] Didymos Reference Model

[RD1] AIM-3P CDF-149(A) Report

[RD8] AIM Advisory Team Final Report[1] https://www-n.oca.eu/michel/AIMReport_Final.pdf

[RD9] Deep Space Instrument Design for Thermal Infrared Imaging with MERTIS, Infrared Remote Sensing and Instrumentation XIX, 2011, I. Walter et al.

[RD10] FANTINA Part II: Payload design and development, Marco Polo Proposal, 2012, A. Herique et al.

[RD11] ExoMars WISDOM a GPR designed for shallow and high-resolution sounding of the Martian subsurface, Proceedings of the IEEE, Manuscript ID: 0023-SIP-2010-PIEEE. D

[RD12] Optical Deep Space Link for AIM System Architecture, Link Analysis and Preliminary Design Concept, ESA DOCUMAS study CCN, 2014[2]

[RD13] OPTEL-μ: A Compact System for Optical Downlinks from LEO Satellites, Dreischer et al.[2]

[RD14] ISIS ISIPOD CubeSat Deployer Product Overview [download]

[RD15] The thermal emission imaging system (THEMIS) for the Mars 2001 Odyssey mission, Christensen et al.

[1] This document is not directly applicable to the AIM Phase A/B1 i.e. it discusses options and performances that are not

[2] This document will be made available to the Contractor at kick-off.

3 ACRONYMS

AIM	Asteroid Impact Mission
CDF	Concurrent Design Facility
COPINS	Cubesat Opportunity Payload Intersatellite Network Sensors
DLR	Deutsches Zentrum für Luft- und Raumfahrt
DTM	Digital Terrain Model
EU	Electronic Unit
FOV	Field Of View
GNC	Guidance Navigation & Control
HFR	High Frequency Radar
JAXA	Japanese Space Exploration Agency
LFR	Low Frequency Radar
LT	Laser Terminal
OHU	Optical Head Unit
PID	Payload Interface Document
S/C	Spacecraft
SoW	Statement of Work
TBC	To Be Confirmed
TBD	To Be Defined
TIRI	Thermal InfraRed Imager
UHF	Ultra High Frequency
VIS	Visual Imaging System

4 AIM STRAWMAN PAYLOAD OVERVIEW

The AIM spacecraft will carry an Optical Communication Terminal (Optel-D), a Thermal InfraRed Imager (TIRI), two radar systems HFR and LFR, an asteroid microlander and three CubeSat-based payloads hosted on a CubeSat deployer. A Visual Imaging System (VIS) is also included as part of the AIM spacecraft system. A short description of these payloads is given within this section.

The Visual Imaging System on AIM will be used both for GNC and to perform scientific measurements. It will be the only element mentioned in this document that will be part of the spacecraft system, and hence under the design responsibility of the S/C prime.
In its scientific role it will be imaging the target asteroid system from multiple fixed positions and from various distances during the course of the AIM asteroid observation phases. Its objective will be to provide information on the binary asteroid dynamics and (especially for the smaller secondary component) asteroid volume, shape, and surface topology data. The VIS shall be designed to provide panchromatic images in the visible range (goal: three colours in the visible range) [RD8]. The resolution of surface images shall be better than a meter/pixel (goal: the DTM shall have a resolution of 1 m lateral and 0.3 m vertical resolution - pixel resolution must be factor 3-5 times better, depending on

method.). The baseline design in the AIM3P CDF study [RD1] assumed that 0.5 m/pixel scale would be achieved from a nominal observation point 10 km above the surface of the asteroid,. The VIS design shall be able to have the entire binary asteroid system within its FOV from the nominal observation point. A lossless compression factor of 2 plus a simple pixel selection based on threshold achieving further data rate reduction by asteroid image extraction were assumed in the data budgets.

The AIM thermal imager (TIRI) primary goal is to investigate the thermal properties of the asteroid surface that are relevant to the characterisation of the soil structure and cohesion. It shall be able to view the entire binary system simultaneously to the VIS camera. The baseline design is based on an evolution of the MERTIS instrument on the BepiColombo ESA mission [RD9]. Such design for AIM TIR features a spectrometer based on a 160x120 pixel uncooled bolometer array and an additional radiometer. Both sensors will share one optical path. The AIM TIRI baseline design has a FOV of 10.3° and would use a pushbroom principle for imaging.

The AIM high-frequency radar (HFR) is based on the WISDOM instrument design developed for the ExoMars mission [RD11]. Its main objective is to obtain information on the structure of the asteroid´s outermost surface and sub-surface layers, up to a depth of 10 meters. The monostatic radar is a Step Frequency (SF) radar operating from 300 MHz to 3 GHz. It shall be operated within 10 km range to the surface scattering the radar signal.

The AIM bistatic Low-Frequency Radar (LFR) primary goal is to obtain data on the asteroid internal structure. The baseline design heritage comes from Rosetta´s CONSERT transmission radar experiment, and the assumed design is similar to the FANTINA-R [RD10] instrument proposed in the context of the Marco Polo-R mission assessment phase. The instrument consists of two subsystems, one on the main S/C and one on the lander. Since the subsystem accommodated by the lander will not interface the main S/C, only the interface of the counterpart on the orbiter is given in this document. The LFR system operates within a 20 MHz bandwidth at a center frequency of 60 MHz, possessing a maximum signal level of 30 dBm at the S/C. A priori, the LFR operation preparation requires a shape, motion and orbitography model. The accuracy should be 10 m for the shape and 100 m for the orbit of the secondary asteroid.

The MASCOT-2 lander is an evolution of its predecessor, MASCOT, developed by DLR for JAXA´s Hayabusa-2 mission to be launched in 2016. MASCOT-2 design is also largely based on the Marco Polo-R FANTINA lander design [RD10]. As in the case of FANTINA, it would accommodate the second component of the bistatic LFR, a visual camera and an accelerometer. It will be deployed on the secondary component of the asteroid system, on a landing area providing safe landing and operation conditions e.g. vertical speed well below escape velocity right after touchdown, and illumination conditions that alternate sun and shadows at regular intervals. Its operative lifetime once deployed on the surface will be at least 3 months.

The objective of the Optel-D optical downlink system of 2.5 Gbps downlink rate capability is to perform an In-Orbit-Demonstration of its capability to transmit data from large distances in deep space. The Optel-D baseline design is an evolution of the Optel-μ equipment. It operates in the 1550 nm C-band (TBC) and will enable the transmission of AIM payload data to ESA´s Optical Ground Station while the spacecraft is operation close to the asteroid. Other optical ground stations would also be used for larger distances if they became available. The downlink system will in principle be operated for no more than one hour a day.

Finally, one 3U cubesat deployment system based on existing developments with flight heritage [RD14] will be mounted on the AIM spacecraft. This will carry the Cubesat Opportunity Payload Intersatellite Network Sensors (COPINS) during the AIM mission. Once in the asteroid vicinity, and with the objective of carrying out complementary measurements (e.g. surface remote or in-situ sensing or impact plume imaging) this will deploy at least one and up to three payloads, based on either 1-U, 2-U or 3-U cubesat configurations. After deployment, the S/C will only support COPINS operations during maximum 1 month.

5 VISUAL IMAGER (VIS)

In line with the AIM Phase A/B1 Statement of Work the VIS camera shall be designed by the Contractor as part of the spacecraft GNC subsystem. The data below addresses the minimum required data handling capability that needs to be guaranteed by the spacecraft to support the VIS asteroid research observations.

Data rate	251.64 kbit/s
Data volume over lifetime	7.55 Gbit

Table 5-1 VIS data rate & volume

6 THERMAL IMAGER (TIRI)

6.1 Mechanical interfaces

The thermal imager consists of the electronics box housing the imaging system, a long cylindrical baffle and a smaller nozzle-like baffle. The longer main baffle is mounted on the optical port used for the main scientific measurements, while the short baffle is mounted to the deep space sensor calibration port. The optical ports lie on perpendicular sides of the Electronic Unit box. On the front of the top face of the Electronic Unit (180x180 mm) lies the calibration port. The measurement port lies on the front face of the Electronics box. The optical ports must remain unobstructed by any S/C structural elements.

Total mass (inc. 10% margin)	3.6 kg

Table 6-1 TIRI mass budget (10% margin included) [RD9]

Electronics Unit	180 mm x 180 mm x 130 mm (cuboid)
Main baffle	200 mm (length) x 75 mm (diameter)
Calibration baffle	90 mm (length) x 75 mm (diameter)

Table 6-2 TIRI equipment dimensions [RD9]

6.2 Electrical interfaces

Type of power supply regulated / unregulated: TBC

Peak Power (cold case heating)	19 W
Peak Power (nominal operation)	13 W

Table 6-3 TIRI power consumption (~~no~~ margin included) [RD9]

6.3 Data handling interfaces

Max. data rate generated	0.06 kbit/s
Data volume over lifetime	16.82 Mbit

Table 6-4 TIRI data handling interface

6.4 Attitude control interface

Measurements are assumed to be conducted from a fixed observer position relative to the asteroid system. Each measurement will last 12 h during which the entire asteroid binary system shall be within the FOV of TIRI.

Required pointing accuracy	± 0.53°
Required positioning knowledge of S/C	± 50 m

Table 6-5 TIRI attitude control interface

6.5 Thermal interfaces

The thermal environment given in Table 7-6 must be ensured by the S/C during the entire lifetime.

Max. operational temperature	30	C°
Min. operational temperature	- 30	C°
Max. non-operational temperature	45	C°
Min. non-operational temperature	- 50	C°

Thermal stabilisation range	0.01	C°

Table 6-6 TIRI thermal interface [RD9] [RD15]

7 MONOSTATIC HIGH FREQUENCY RADAR (HFR)

7.1 Mechanical interfaces

The HFR is composed of an electronics unit and an antenna structure. The antenna structure, containing the two Vivaldi antennas must be fixed on the outer side of the S/C. The antenna structure may be abstracted as a box with the dimensions given in Table 7-1. The interface of S/C to the antennas is a surface with dimensions 140 mm x 410 mm.

Electronics Unit	0.7 kg
Antenna structure	0.5 kg
Cables	0.2 kg
20% margin	0.3 kg
Total mass	1.7 kg

Table 7-1 HFR mass budget (20% margin included) adapted from [RD10]

Electronics Unit	145 mm x 155 mm x 50 mm (cuboid)
Antenna Structure	260 mm x 620 mm x 240 mm (cuboid)

Table 7-2 HFR equipment dimensions [RD10]

7.2 Electrical interfaces

Type of power supply: TBC.

Peak Power (full power mode)	16 W
Peak Power (low power mode)(RF OFF)	7 W

Table 7-3 HFR power consumption (~~no~~ margin included) [RD10]

7.3 Data handling interfaces

Max. data rate generated	80 kbit/s
Data volume over lifetime	10 Gbit

Table 7-4 HFR data handling interface

7.4 Attitude control interface

Each measurement will last 12 h during which the radar antenna must be oriented towards the secondary asteroid given in Table 7-5.

Required pointing accuracy	+- 5°
Required positioning knowledge of S/C	+- 100 m

Table 7-5 HFR attitude control interface

7.5 Thermal interfaces

The thermal environment given in Table 7-6 must be ensured by the S/C during the entire lifetime.

Max. operational temperature	50	C°
Min. operational temperature	- 40	C°
Max. non-operational temperature	50	C°
Min. non-operational temperature	- 50	C°

Table 7-6 HFR thermal interface

8 BISTATIC LOW FREQUENCY RADAR (LFR)

8.1 Mechanical interfaces

The LFR consists of an electronics unit and an antenna structure. The baseline orbiter antenna design consists of four antenna elements mounted at the corners of the spacecraft in plane like shown in Figure 8-1. Four monopoles are foreseen to transmit LHCP waves from the orbiter to the lander at the surface of the asteroid. The angle between the spacecraft body and the antenna is 135°. Figure 8-2 shows the dimensions of this configuration in a top view of S/C central body and antenna arms. The deployment mechanism for the antenna rods will be included within the payload.

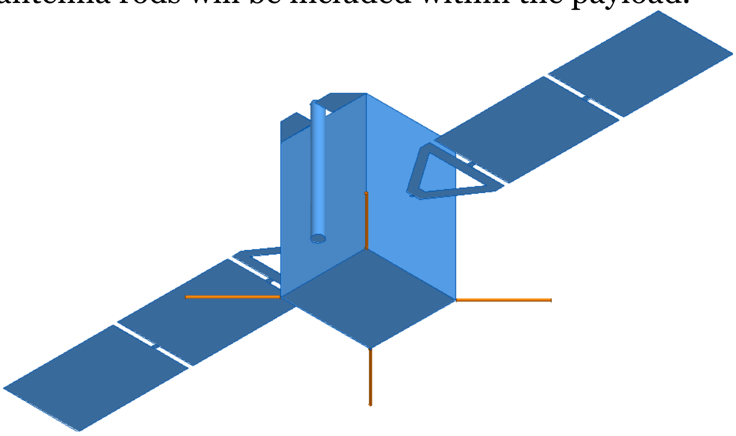

Figure 8-1 LFR Antenna arrangement on the AIM spacecraft

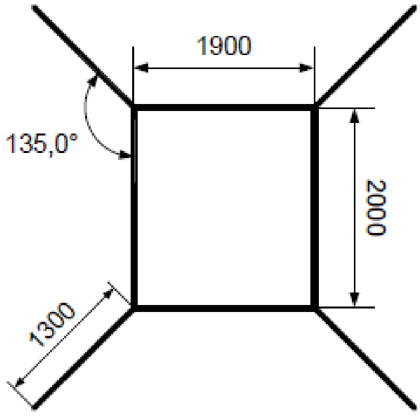

Figure 8-2 Top view of antenna arrangement [RD10]

Electronics Unit	0.3 kg
Antenna structure	0.3 kg
Shielding	0.3 kg
10% margin	0.3 kg
Total mass	1.2 kg

Table 8-1 LFR mass budget on spacecraft (10% maturity margin included) [RD10]

Electronics Unit	220 mm x 220 mm x 60 mm (cuboid)
Antenna Structure	4 x 1300 mm (rod)

Table 8-2 LFR equipment dimensions on AIM spacecraft [RD10]

8.2 Electrical interfaces

Type of powwe supply: TBC

Peak Power	13 W
Mean Power (stand-by-mode)	< 4 W

Table 8-3 LFR power consumption (~~no~~ margin included) [RD10]

8.3 Data handling interfaces

Max. data rate generated	20 kbit/s
Data volume over lifetime	0.64 Gbit

Table 8-4 LFR data handling interface

8.4 Attitude control interface

During measurements the antenna structure must be oriented towards the target asteroid with the precision given in Table 8-5. The duration of one measurement will be approx. 12 hours.

Required pointing accuracy	+- 5°
Required positioning knowledge of S/C	+- 10 m

Table 8-5 LFR attitude control interface

8.5 Thermal interfaces

The thermal environment given in Table 8-6 must be ensured by the S/C during the entire lifetime.

Max. operational temperature	50	C°
Min. operational temperature	- 40	C°
Max. non-operational temperature	50	C°
Min. non-operational temperature	- 45	C°

Table 8-6 LFR thermal interface

9 MASCOT-2

9.1 Mechanical interfaces

The MASCOT-2 system is made of two separate elements, a MASCOT-2 lander and a Mechanical and Electrical Support Structure (MESS). The functionality provided by the interface elements between the lander and the main spacecraft shall enable a support of the lander during the cruise phase. The MESS acts as housing for the FANTINA Lander and is thus the main load bearing element between the lander and the AIM S/C structure. It includes a Separation & Push-off Mechanism consisting of a non-explosive actuator releasing a bolt, which mounts the lander to the main spacecraft. A spring pushes the lander away upon deployment.

The MASCOT-2 lander in its stowed configuration housed inside the MESS is depicted in Figure 9-1. MESS has a cube-like shape with dimensions given in Table 9-1.

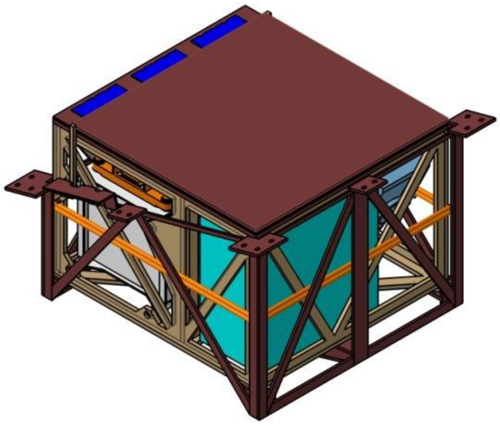

Figure 9-1 MESS including stowed MASCOT-2

Dimensions	337 mm x 301 mm x 200 mm
Mass (inc. 10% margin)	13.0 kg

Table 9-1 MESS including stowed MASCOT-2 system mass (10% margin incl.) and dimensions [RD10]

9.2 Electrical interfaces

The Electrical Connector is the electrical interface between the AIM S/C and the lander during the launch and cruise phases. it bundles power (to PDCU and heater) and data lines (temperature probes, separation sense line). To prevent the depletion of the primary batteries of MASCOT-2 when it is mounted within the S/C, the carrier shall keep the battery charged. This will require a power supply of approx. 1 W, from the time of integration into the S/C until deployment. During deployment, the MASCOT-2 deployment mechanism will require approx. 60 W for less than 1 sec in order to trigger the release.
Type of power supply: unregulated 46 V-52 V, nominal 50 V.

9.3 Data handling interfaces

All communications between MASCOT-2 and the ground station are relayed by the -S/C using an Inter-Satellite Link that is part of the S/C telecommunication design. Data volumes over a measurement cycle and over its entire lifetime are given below.

Data volume over measurement cycle	50 Mbits
Data volume over lifetime	850 Mbits

Table 9-2 MASCOT-2 data handling interface (after deployment)

9.4 Attitude control interface

The following ACS requirements are valid during the deployment phase of MASCOT-2:

Required pointing accuracy	+- 0.25°
Required positioning knowledge of S/C	+- 50 m

Table 9-3 MASCOT-2 attitude control interface during deployment

9.5 Thermal interfaces

The MESS isolates thermally the MASCOT.2 lander and ensures thermal conditioning by the S/C during all mission phases before lander deployment. The thermal environment given in Table 8-6 must be assured by the S/C for the MESS:

Max. operational temperature	30	C°
Min. operational temperature	- 30	C°
Max. non-operational temperature	50	C°
Min. non-operational temperature	- 35	C°

Table 9-4 MESS thermal interface

10 OPTICAL LASER TERMINAL (OPTEL-D)

10.1 Mechanical interfaces

The Optel-D system is composed of the electronic unit (EU), the optical head unit (OHU), the laser terminal (LT) and its harness. The volume of these elements will be approx. 5.6 l in total.

Optical Head Unit	20.1	kg
Laser Unit	7.0	kg
Electronics Unit	3.2	kg
Harness	2.4	kg
Total w/o margin	32.7	kg
20% margin	6.6	kg
Total	39.3	kg

Table 10-1 Optel-D mass budget (20% margin included)[RD12]

10.2 Electrical interfaces

Type of power supply: Unregulated 24 – 46 VDC. -

Peak Power (nominal operation)	129 W

Table 10-2 Optel-D Power consumption (~~no~~ margin included) [RD12]

10.3 Data handling interfaces

Only standard housekeeping data js generated by the Optel-D in the baseline design. This housekeeping data must be transmitted to Earth.

10.4 Attitude control interface

The Optel-D system includes an internal pointing mechanism, which enables high precision pointing when provided with attitude data from the S/C.

The Sun-Spacecraft-Earth angle shall be at least 19 degrees during OPTEL-D operations to avoid Sun blinding of the optical system.

Required pointing accuracy (APE) from S/C	+- 70 arcsec
Required accuracy of pointing knowledge (APK) provided to the Optel-D system by S/C	+- 10 arcsec
Relative Pointing Error (RPE)	< 0.6 arcsec over 1 s

Table 10-3 Optel-D ACS interface

10.5 Thermal interfaces

The thermal environment given in Table 10-4 must be realised for all components of the Optel-D by the S/C.

Max. operational temperature	30	C°
Min. operational temperature	0	C°
Max. non-operational temperature	50	C°
Min. non-operational temperature	- 30	C°

Table 10-4 Optel-D Thermal interface [RD12]

11 CUBESAT OPPORTUNITY PAYLOADS (COPINS)

11.1 Mechanical Interfaces

The AIM Cubesat Opportunity Payload Intersatellite Network Sensors (COPINS) CubeSat deployer dimensions and mass are given below for each of the two units.

Outer Envelope (pre-deployment, doors closed)	≤414.1 x 130.3 x 180.8 mm
Mass (inc. 10% margin)	2.1 kg

Table 11-1 CubeSat deployer maximum mass and dimensions (including 10% margin) [RD14]

Two separate CubeSat deployer ¨pods¨ will be accommodated on the AIM S/C. The mounting orientation must be such that the deployer side containing the exit door is facing outside the S/C. The mounting plane will be the back plate (100x100 mm face) or the side panels (100x300 mm face).

Natural Frequency	> 90 Hz

Table 11-2 CubeSat Deployer Natural Frequency Requirement

The CubeSats themselves will not have direct mechanical interfaces to the S/C. They will adhere to the standard CubeSat dimensions. The total internal volume of the CubeSat deployer is 3 litres, constraining to a maximum 3-Unit configuration. Design would include rails on the long axis for integration with the deployer.

1U Dimensions	100 x 100 x 100 mm (1 litre)
1U Mass	1.5 kg
2U Dimensions	100 x 100 x 200 mm (2 litres)
2U Mass	3 kg
3U Dimensions	100 x 100 x 300 mm (3 litres)
3U Mass	4.5 kg

Table 11-3 Individual CubeSat maximum mass and dimensions

11.2 Electrical interfaces

The CubeSats will interface with the main spacecraft pre-deployment using an umbilical via the deployer and CubeSat external port. This will be for battery charging and functional checkout / on-board software maintenance and will require 1W power.

There will also be an electrical interface for the door of the deployer. The CubeSat deployer will electrically interface with the main spacecraft for telemetry signal to the spacecraft door switch status (OPEN/CLOSED) and pusher plate switch status (STOWED/EXTENDED). The typical actuation values for the door release mechanism are 28V (±4V) and 1.75A for 250 ms [RD14]. The connector used will be a standard electrical Micro D or other serial connector.

Typical Power	1W
Peak Power (250 ms)	49W

Table 11-4 CubeSats and deployer power consumption

11.3 Data handling interfaces

(After deployment)
All communications between COPINS and the ground station are relayed by the -S/C using an Inter-Satellite Link that is part of the S/C telecommunication design. Peak data rates and data volume over the lifetime are shown in Table 11-5.

Max. data rate generated	1 kbit/s
Data volume over lifetime	1 Gbit

Table 11-5 CubeSats and deployer data interface

11.4 Attitude control interface

(During deployment).

Required pointing accuracy	+- 0.25°
Required positioning knowledge of S/C	+- 50 m

Table 11-6 COPINS attitude control interface during deployment

11.5 Thermal Interfaces

The thermal environment of the CubeSats and deployer during transit will be provided by the AIM S/C.

Max. operational temperature	30	°C
Min. operational temperature	- 30	°C
Max. non-operational temperature	50	°C
Min. non-operational temperature	- 35	°C

Table 11-7 CubeSat Thermal Interface

Max. operational temperature	30	°C
Min. operational temperature	- 30	°C
Max. non-operational temperature	50	°C
Min. non-operational temperature	- 35	°C

Table 11-8 CubeSat deployer Thermal interface [RD14]

12 PAYLOAD SUMMARY

Table 12-1 presents a summary on the *minimum* data volume that would need to be produced during the mission to meet the asteroid observation goals.

Full use of downlink capacity by daily contacts compatible with the baseline ground stations i.e. ESA DSN and OGS, and the efficient use of on-board data storage, together with a suitable payload operation plan, are assumed to maximise data return.

Observation phase	Minimum Data volume [Mbit]	Measurements
Early Characterisation Phase	1258	1 measurement set of VIS
Detailed Characterisation Phase Period 1.	11261	1 measurement sets of VIS + 1 measurement sets from TIR + HFR
Detailed Characterisation Phase Period 2.	2523	2 measurement sets of VIS + 2 measurement sets from TIR
Detailed Characterisation Phase Period 3.	3784	2 measurement sets of VIS + 3 measurement sets from TIR
TBD	3690	Minimum data volume (MASCOT-2 incl. LFR) + COPINS
Total	21260	

Table 12-1 Generated data volume by AM payloads

Table 12-2 shows the mass budget for the AIM payloads:

Payload	Mass [kg]	Margin [%]	Mass [kg] incl. margin
TIRI	3.3	10	3.6
HFR	1.4	20	1.7
LFR	1.1	10	1.2
OPTEL	32.7	20	39.3
MASCOT-2	11.7	10	13.0
COPINS	11.8	10	13.2
Total	62.0		72.0

Table 12-2 Payload mass budget