# Concepts of Active Payload Modules and End-Effectors Suitable for Standard Interface for Robotic Manipulation of Payloads in Future Space Missions (SIROM) Interface

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*Abstract*—**The increasing variety of space missions, combined with their rising complexity and need for more environmentally-friendly, yet cost-effective, solutions, is putting the traditional spacecraft and rover designs to the test. In fact, the majority of nowadays spacecraft and planetary rovers are mostly monolithic, one-of-a-kind, single-use systems, hardly offering any possibility for their future servicing, upgrade or re-use.**

**The EU-funded H2020 project Standard Interface for Robotic Manipulation of Payloads in Future Space Missions (SIROM) aims to bridge this gap by developing an integrated and inherently optimized multi-functional standard interface for mechanical, data, electrical and thermal transfer. The interface, in combination with a custom end-effector and active payload modules (APMs), will allow to design modular and re-configurable systems that could be easily serviced and upgraded via a dedicated robotic system for in-orbit or planetary environment. With respect to the existing state-of-the-art, the interface and modules in SIROM are being developed considering the need for scalability, reusability, compatibility with robotic manipulation and suitability for both environments.**

**Within this context, the paper aims to analyze the feasibility of APM and end-effector concepts, within the system requirements of the project, and identify their most suitable preliminary concepts. The analysis is performed in terms of functionalities and architecture, and in case of APMs, considers a remote sensing and power storage system as payloads for orbital and planetary scenarios, respectively.**

**The methodology used for the evaluation of APM and end-effector concepts is a top-down methodology generally used for the design and sizing of payloads of space missions. It consists of: (a) definition of payload objectives and its desired capabilities, (b) identification of candidates, (c) estimation of their characteristics based on analogy, scaling or component budgeting, and (d) evaluation and selection of a reference concept. Moreover, in case of the end-effector analysis, interactions and configurations with APM concepts were also taken into consideration.**

**The results of the analysis point out the feasibility of APMs and end-effectors, within the system requirements of the project, and outline concepts that could be used in the future steps of the project as a guideline in the detailed design of APMs and end-effectors.**

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## 1. INTRODUCTION

Driven by the increasing demand for global connectivity, monitoring and space exploration, the number and complexity of space missions are only expected to increase in the near future. However, in order to keep mission costs at bay, a shift in the current paradigm of space missions and morphology of spacecraft design is needed. The current spacecraft design, where reliability is achieved through redundant and highly reliable components on highly integrated platforms, is not a successful model, especially for long-term missions (e. g. ranging from 12-15 years) in geostationary equatorial orbit (GEO), soliciting spacecraft operators to insure their space assets before a launch [1]. The very nature of spaceflights has heavily constrained every space mission since the launch of the first artificial satellite in terms of mass, lifetime, and ultimately cost. Therefore, the adopted mission paradigm resulted mostly in one-of-a-kind, single-use, highly integrated, space systems that rarely leaves any room for error or flexibility. In order to pursue the commercialization of space, more cost-effective and flexible systems are needed that could be serviced in-orbit [2]. Furthermore, the proliferation of the space debris population calls for a more sustainable use of the space environment [3], by recycling or re-using current space assets. Likewise, future space exploration missions call for more flexibility, that would allow failures, and modularity, so that not everything needs to be brought in one mission [4]. All these facts point toward the need to develop cost-effective on-orbit servicing (OOS) and assembly (OOA) missions, that could potentially reduce the life cycle cost of a system and increase its capability, or provide more flexibility in case of failures or unexpected events. To achieve this, a new, more "cooperative" morphology of spacecraft is needed. Among proposed solutions, the most attractive one, from an economical sustainability point of view, is that of modular spacecraft, composed of multiple heterogeneous modules that can be easily connected to or disconnected from the whole via one or more standard multifunctional interfaces, integrated in

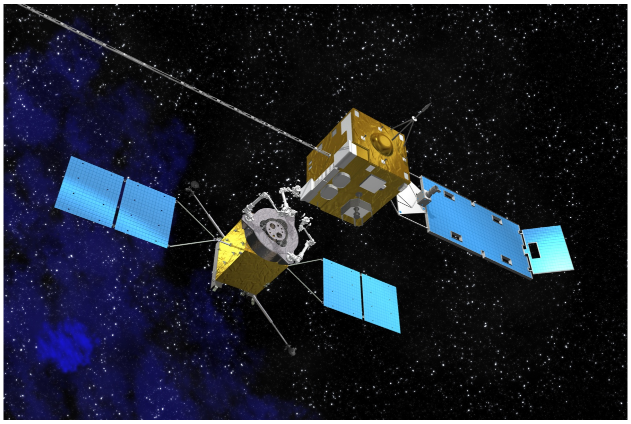


**Figure 1. The artist's concept of a robotic on-orbit servicing scenario (credit: NASA/Goddard Space Flight Center 2017).**

**Figure 2. Real world test of a multi-robot planetary exploration scenario using modular payload-items (PLIs) performed by DFKI at the Utah desert USA. The main rover is visible in the center, PLIs on the right and an agile robot in the upper-right of the figure (credit: DFKI 2016).**

each module. This is especially true in case of future robotic on-orbit servicing missions, where a standard interface would greatly simplify any task at hand [5].

A standard multifunctional interface is defined in this paper according to the "Guidelines for strategic research cluster on space robotic technologies of the Horizon 2020", section E, page 22, as: "a combination of devices that allow to couple active payload modules (APMs) to a manipulator, among themselves and to spacecraft".

In recent years numerous multifunctional interfaces have been proposed [2], [4], [6], [5], [7]. However, to the best of our knowledge, none has considered the modularity in both the orbital and planetary contexts with only one standard interface. The Standard Interface for Robotic Manipulation of Payloads in Future Space Missions (SIROM) project aims to bridge this gap by developing a compact, integrated and inherently optimized multi-functional standard interface for mechanical, data, electrical and thermal connectivity. The interface, in combination with custom end-effectors and APMs, would allow the design of modular and re-configurable systems that could favor low cost on-orbit servicing and modular robotic exploration of planetary bodies. Therefore, two distinct mission reference scenarios were defined within the SIROM project for the characterization of the standard interface, APMs and end-effectors.

The orbital reference scenario consists of a robotic spacecraft performing an on-orbit servicing task on a cooperative, modular spacecraft in low Earth orbit (LEO). The OOS task of the robotic spacecraft, illustrated in Figure 1, shall in particular consist of a replacement of antiquated/faulty payload, such as an optical sensor. The exchangeable modules shall be integrated with two standard interfaces and transferred via a robotic arm from the servicer to the client spacecraft while considering the latter to be firmly docked with the former.

The planetary reference scenario is defined on the basis of future autonomous long-range planetary rover exploration missions, such as the future NASA Mars Sample Return mission. It consists of a main rover (containing the majority of scientific equipment), used to explore the surface of a planet (e. g. Mars), smaller, more agile robots and payload modules that shall be used to extend the capabilities of all robots in terms of autonomy, navigation, and/or communication, as illustrated in Figure 2. The payload modules shall be transported by the primary rover to a desired location, deposited for a certain amount of time, and subsequently recovered/used when needed. The manipulation of modules shall be performed with a robotic arm mounted on the primary rover.

With those reference scenarios in mind and with an intention to facilitate the development and testing phases of the SIROM project, the overall physical properties of the new interface were constrained to a cylinder having a diameter of $120\,\mathrm{mm}$, height of $30\,\mathrm{mm}$ and mass of $1.5\,\mathrm{kg}$, making it suitable for a large range of future missions, especially those involving small spacecrafts and robotic modules. At the same time the interface is envisioned to be easily scalable, opening the market for an even larger range of payloads, and therefore, missions.

The SIROM project is one of the six operational grants (OGs) founded in the 2016 call for the European Union (EU) Horizon 2020 (H2020) Strategic Research Cluster (SRC) in Space Robotics Technologies. It is a European attempt to address the needs for reducing costs and increasing standardization of space missions to allow access to space to a larger number of customers. This makes it an excellent opportunity to develop and demonstrate state-of-the-art interface technologies which will be essential in many future missions, ranging from OOS to robotic exploration missions.

Within this context, this paper aims at describing the work performed by DFKI and University of Strathclyde during the preliminary design phase of the SIROM project. To this end, the paper details an analyses of the feasibility of APM and end-effector concepts, within the system requirements of the project, and identifies most suitable concepts for the preliminary design of APMs and end-effectors to be performed in the next steps of the project. The analysis is performed in terms of functionalities and architecture, and in case of APMs, considers a remote sensing and power storage system as payloads for orbital and planetary scenarios, respectively. The interface considered in this paper is a preliminary concept developed by the University of Strathclyde and is to be considered only a case study and one of the iterations of the interface being developed within the consortium.

The structure of the remainder of the paper is as follow: Section 2 presents the state-of-the-art of robotic and spacecraft interfaces, payload modules and end-effectors of space manipulators. Section 3 describes the overall methodology used for the evaluation and selection of the most suitable concepts for active payload modules and end-effectors. Section 4 provides an overview of the characteristics of a preliminary design of the standard interface (IF). Section 5 details envisioned APM concepts and their baselines for orbital and planetary scenarios, in terms of their functionalities and architecture, based on system requirements. Section 6 illustrates the analysis and selection of the most suited end-effector concept for future studies, based on system requirements and end-effector interaction with the previously defined APMs. Finally, Section 7 provides the summary of the results of the paper and concluding remarks.

## 2. Background and State-of-the-Art

The concept of robotic on-orbit servicing and assembly of spacecraft dates back to the early 1980s, after the first ever successful use of the Shuttle Remote Manipulator System (SRMS), or Canadarm, in an orbital environment during the STS-2 Space Shuttle mission [8]. Several Shuttle missions followed aiming to capture, repair or deploy malfunctioning satellites (e. g. the STS-41C launched in April 1984 to repair the stranded SolarMax spacecraft and the STS-51A launched in November 1984 to recover two strayed satellites Palpa B2 and Westar 6) culminating with the on-orbit servicing of the Hubble Space Telescope (HST) and on-orbit robotic assembly of the International Space Station (ISS). Moreover, several technology demonstrator missions were executed from 1997 to 2007, such as the Engineering Test Satellite (ETS)-VII, NASA's DART and DARPA's Orbital Express missions, to demonstrate unmanned on-orbit servicing and refueling of in-orbit spacecrafts [4]. All of these missions served to prove that spacecraft can be serviced even if not designed for such tasks. However, the modularity of the SolarMax spacecraft and HST underlined the importance of a "cooperative" spacecraft design for on-orbit servicing [5]. However, modularity comes at a cost of additional structural mass which could negatively impact the overall mission cost when compared with a typical highly integrated spacecraft [5]. Moreover, advanced modularity could also have a negative effect on the total life-cycle cost of a spacecraft and its scientific return. Therefore, these issues need to be carefully taken into consideration and traded-off during the design phase of a mission, to find an optimal level of modularity that would have a positive impact on the overall mission when compared with a traditional one [9].

### *Modularity of Spacecraft and Planetary Robots*

The modularity of spacecraft or planetary robots in this paper defines the level of subdivision of a system in standardized and easily replaceable units, interconnected between them or with the main bus via a relatively simple standard interface [5]. These units can contain any number of replaceable system components such as inertial reference units, payload, electronics, power distribution units, batteries, etc. [2].

Over the years, different levels of spacecraft modularity have been implemented, ranging from highly integrated, specialized systems, to highly modular ones, comprised entirely of large number of small modules [5].
Typical spacecraft generally consist of many individual components, which integration and interfaces are highly optimized towards mass and cost reduction. Therefore, they are not easily serviceable on-orbit, if at all [5].
One step closer toward an advanced modularity is represented by the minimally modular spacecraft, such as those being part of families of commercial communication spacecraft. They are generally composed of two to three large modules that allow parallel integration and testing (I&T) and provide significant cost savings but not necessarily on-orbit servicing [5].
The serviceable modularity or modularity at the component level represents an even higher level of modularity then the previous one. Examples of spacecraft with this level of modularity are the HST and ISS, equipped with serviceable components and standard interfaces. However, these components are not grouped into serviceable modules, meaning that any OOS task would need to be performed at a component level with tools and procedures specifically developed for each component separately.
This complication can be avoided by developing systems with a subsystem level of modularity, consisting mainly of components integrated into modules which can be easily removed/replaced on-ground as well as in-orbit. Examples of such type of spacecraft are the Multimission Modular Spacecraft (MMS), the SolarMax spacecraft, and the Reconfigurable Operational spacecraft for Science and Exploration (ROSE). They contain components grouped into serviceable modules, integrated onto the main bus via a standardized interface, thus allowing a great deal of flexibility both on-ground, during I&T activities, and in-orbit, while keeping the complexity of those tasks at the minimum [5].
The intelligent Building blocks for On-orbit Servicing (iBOSS), Autonomous Assembly of a Reconfigurable Space Telescope (AAReST), DARPA's Satlets and Self Assembling Wireless Autonomous and Reconfigurable (SWARM), are designed with an even greater spacecraft modularity in mind. In these concepts the overall spacecraft system is composed of small interconnected modules, each providing only a fraction of functionality of a traditional spacecraft, comparable to cells in a living organism. Modules are envisioned to be interconnected via intelligent plug-and-play interfaces, allowing almost total in-orbit reconfiguration and assembly, with the highest level of flexibility in mind [5]. The type and number of individual modules will be based on an optimization process that will depend not only on engineering metrics, such as the cost and mass, but also on other less quantifiable metrics, such as future market uncertainties/projections and influence of stakeholders [10], [11], [9].

In line with the typical spacecraft design, planetary rovers are generally composed of many individual, highly integrated components, not meant for serviceability nor repairability, but with ruggedness and redundancy in mind. In fact, currently deployed Mars rovers Spirit, Opportunity and Curiosity, are highly specialized, single mission systems, conceived to be mobile laboratories to singlehandedly carry-out all the required exploration tasks. However, these systems are proving to be inappropriate for future large-scale exploration missions of planetary surfaces, where coordinated, modular, multi-robot systems will play a pivotal role.
The payload-items (PLIs) developed at DFKI-RIC Bremen, visible in Figure 2, represent one existing solution for such systems able to support robot-to-robot interactions in multi-robot scenarios through the usage of an in-house developed

electro-mechanical interface (EMI) [12].

The SIROM project shall provide a way to extend the modularity of both orbital and planetary systems by developing a single interface that will support advanced modular architectures of spacecraft and planetary rovers, similarly to what is already achieved by iBOSS and PLIs in orbital and planetary scenarios, respectively.

*Robotic and Spacecraft Interfaces*

To support the advanced modularity concepts mentioned in the previous subsection, over the years there has been a great variety of interfaces developed for space missions [13]. Among them, the four that deserve particular attention are: SINGO [14], Phoenix Satlet [15], DFKI's EMI [16], and iSSI (intelligent Satellite System Interface) [17].

SINGO is a fail-safe mechanical connector, powered by a single motor, which design allows two connectors to engage and disengage even if one does not cooperate. The connector is made of four jaws that can bite those of the counterpart: one will bite from outside-in, while the other from inside-out [14].

The Phoenix Satlet interface is part of the DARPA's Phoenix project [18] which focuses on joining satlet modules to harvested sub-systems, such as an antenna, of defunct satellites, to create a new functioning space system [6]. The interface integrates electrical and mechanical functionalities for on-orbit attachment of tools. The mechanical interface is made of a male-female interface that uses expanding directional clamps to latch. The electrical connector is located in the center of a circular interface allowing arbitrary rotation of connected modules. The interface offers three types of engagement operations: mechanical (an internal mechanism pushes out four locking balls that fully align and lock onto a receptacle, guaranteeing a rigid connection), electrical (a custom 20 pin annular electrical connector assures power and data connection), and torque drive (a drive socket spring pushes a torque drive socket towards the mated interface only after the electrical connector is extended outward) [15].

EMI allows a higher level of modularity for planetary robots through the usage of PLIs. This interface integrates a gender-principle approach to allow one side to be designed without any moving parts and openings where dust could enter. A passive male part is used on the upper side of modules, where dust is likely to accumulate, while an active female part is integrated beneath modules and protected from the external environment [12].

The iSSI interface, developed for the iBOSS project, is the only one that currently integrates four different functionalities in one single block: mechanical, thermal, data and electrical. In fact, the interface has a fail-safe hermaphroditic roto-lock mechanism, power contacts, a fiber optic data lens, and an annulus for thermal conductive exchange [17].

The SIROM interface aims to integrate all of the functionalities of the iSSI and EMI interfaces within smaller and lighter modules, while enabling its usage in both orbital and planetary scenarios.

*Robotic End-Effectors*

In robotics, an end-effector is a device designed to be placed at the end of a manipulator to interact with an environment. The structure of an end-effector along with the nature of its hardware and control software, heavily depends on tasks the robot (the end-effector is mounted on) will be performing. Therefore, end-effectors may consist of a gripper, tool or only of an interface for a connection with an external module.

End-effectors developed for the space environment are mainly meant for the manipulator arms of the ISS.
One such manipulator arm is the European Robotic Arm (ERA) planned to be installed onto the Russian segment of the ISS in the near future. Its end-effector can latch to a grapple fixture on a payload or base-point on the station allowing it to "walk" on the exterior of the ISS. Electrical connectors on the base-point provide ERA with power, data and video links to/from the ISS. The integrated service tool in the end-effector can be used to mechanically drive a mechanism in the grappled object [19].
Another very versatile ISS manipulator arm is Dextre, also known as the Special Purpose Dexterous Manipulator (SPDM) or the ISS handyman. It is a dual arm manipulator system, resembling a headless torso, capable of performing delicate tasks on the ISS. The two arms of Dextre are equipped with ORU/Tool Changeout Mechanisms (OTCMs) which include built-in grasping jaws, a monochrome TV camera, lights and an umbilical connector to provide power, data, and video to/from a "grasped" object [20]. The central body of Dextre is equipped with a Power and Data Grapple Fixture (PDGF) at one end, that can be grasped by the Canadarm2, and a Latching End-Effector (LEE), identical to that of Canadarm2, at the other which enables it to attach to other PDGFs on the ISS or the mobile base system [21].
The Self-Adapting Robotic Auxiliary Hand (SARAH) is a another dedicated tool/end-effector designed to be used by the SPDM [22]. It consists of three under-actuated fingers mounted on a common structure. The fingers can envelop various shapes including cylindrical and spherical geometries. The key novelty of its design is that it is a completely self-contained, passive, mechanism requiring only the existing SPDM-OTCM drive mechanisms to actuate the fingers.

Spacehand, developed by the German Aerospace Agency (DLR) for use in higher Earth orbits, such as the GEO, is another end-effector worth mentioning. It is a four tendon driven fingered robotic hand, having the size of an extravehicular activity (EVA) glove, with actuation and electronics completely integrated in the hand, developed specifically for dexterous tasks, such as the removal of multi-layered insulation (MLI) cover [23].

The end-effector of the manipulator arm of the rover SherpaTT developed at DFKI [24] consists of an electro-mechanical interface [16] and a six axis force torque sensor (FTS). The FTS is used to stop the manipulator in case of overloads and allow a force-feedback controlled operation. The FTS can also be used to allow a force guided stacking of payloads. The EMI helps to connect the end-effector with one unit of a multi-robot system and ensures power and data transfer from the main rover to the module being manipulated.

The end-effector planned in the SIROM project shall go a step beyond the existing end-effectors since its development and integration with the standard interface focuses on the applicability in both planetary and space environments.

## 3. METHODOLOGY

*APM Concepts Definition and Evaluation*

An APM in this document is defined as a payload container, used to fulfill certain mission objectives (e. g. interaction with an environment through a sensor, or providing an enhanced computational or electric power to a mobile system).

Payload definition and sizing is what generally determines the capabilities and limitations of any space system. The rest of the spacecraft/rover is defined only to support the payload within mission parameters [25].
With this in mind, the methodology used for the selection and evaluation of payload concepts is a simplified version of a top-down, iterative methodology, used for the payload design and sizing of space missions, documented in [25]. Given its top-down, iterative nature, this methodology is especially useful when no previous reference design exists [25].
The basic steps of the used methodology consist of: (a) a definition of payload objectives and its desired capabilities, (b) an identification of candidates, (c) an estimation of their characteristics based on analogy, scaling or component budgeting, and on (d) an evaluation and selection of a baseline.

*End-Effector Concepts Definition and Evaluation*

The methodology used for the selection and evaluation of end-effectors is similar to the one described for APM concepts. However, the methodology employed in this case is also heavily influenced by possible interactions of end-effectors with APM concepts, and constrained by the characteristics of the manipulator arm of the SherpaTT rover (see Figure 2), which shall be used for testing purposes of the planetary scenario.

## 4. STANDARD INTERFACE CONCEPT

To achieve generalized modularity of spacecraft and planetary rovers, a preliminary concept of a modular and scalable interface, that could be applied in both scenarios, has been designed by the University of Strathclyde and its main characteristics are outlined hereafter.

The presented design is to be considered only a case study and one of the first iterations of the interface being developed within the consortium. In fact, the concept interface (IF) is mostly suited for orbital scenarios, given the lack of any dust cover. However, the overall design has been made compliant with the main system requirements and is therefore used as a reference in the rest of the paper.

The final design of the SIROM interface is currently under development within the consortium; however, its description is outside the scope of this paper.

*System Requirements*

The system requirements of the SIROM interface were defined based on the orbital and planetary reference scenarios, delineated in Section 1, while keeping the development and on-ground testing costs in mind.
More specifically, those requirements impose that the SIROM interface shall:

- support standardized mechanical, data, electrical and thermal IFs;
- support modular spacecraft or rover architectures;
- be fail-safe;
- have an architectural flexibility in terms of:
  - scalability with low complexity, mass and volume;
  - internal redundancy;
  - compatibility with robot servicing;
  - symmetry, and at least one axis of rotation;
  - connection of nearly arbitrary types of modules, without restriction on the relative module orientation;
- have dimension and weight limits of $120 \times 120 \times 30\,\mathrm{mm}$ (L × W × H) and $1.5\,\mathrm{kg}$, respectively;
- provide an electrical power $\geq 150\,\mathrm{W}$;
- provide a data rate $\geq 100\,{}^{\mathrm{Mbit}}\!/_{\mathrm{s}}$ (preferably via the SpaceWire (SpW) or CAN protocols);
- transfer a thermal load $\leq 50\,\mathrm{W}$;
- allow $5\,\mathrm{mm}$ tolerance when mating with another standard IF.

*Characteristics of the Concept Interface*

Based on those requirements, a concept interface was developed at the Design, Manufacture & Engineering Management (DMEM) Department of University of Strathclyde.
The prototype is $650\,\mathrm{g}$ and $120\,\mathrm{mm}$ in diameter; its body is $18\,\mathrm{mm}$ thick, which with four $12\,\mathrm{mm}$ high pins, makes the overall interface $30\,\mathrm{mm}$ high. It is actuated by one DC brushed geared motor, able to provide a maximum torque of $50\,\mathrm{Nmm}$, enough to ensure $0.55\,\mathrm{MPa}$ of contact pressure between the thermal patches of two interfaces in contact [26], necessary for an efficient thermal transfer.

The concept interface is shown in Figure 3. The green boxes highlight the central mechanism, that ensures an axial lock during mating, one of the four pins and one of the four chamfered holes, that ensure the guidance and tolerance requested during mating. The red boxes show one of the four pairs of male-female, 9-pole, D-micro SpW connectors envisioned for the data transfer. The blue boxes locate the position of one of the four pairs of +/- flat pins envisioned for the electric power transfer, while the yellow box identifies one of the four thermal patches for the heat transfer.
The reason to equip the IF with redundant pins-holes, data and power connectors, and thermal patches, is due to the decision to enable four degrees of axial-symmetry, allowing four different mating positions with a counterpart IF that increases flexibility, ease of operations and redundancy [26].
The overall performances of the designed prototype IF are summarized in Table 1 [26].

**Table 1. Nominal performances of the prototype IF**

| Mechanical | Thermal | Data | Electrical |
|---|---|---|---|
| $4640\,\mathrm{N}$ (radial)<br>$440\,\mathrm{N}$ (axial)<br>$128\,\mathrm{Nm}$ (torque) | $29\,\mathrm{W}$ for $10\,\mathrm{K}$ of thermal gradient | $400\,{}^{\mathrm{MB}}\!/_{\mathrm{s}}$ | $600\,\mathrm{W}$ |

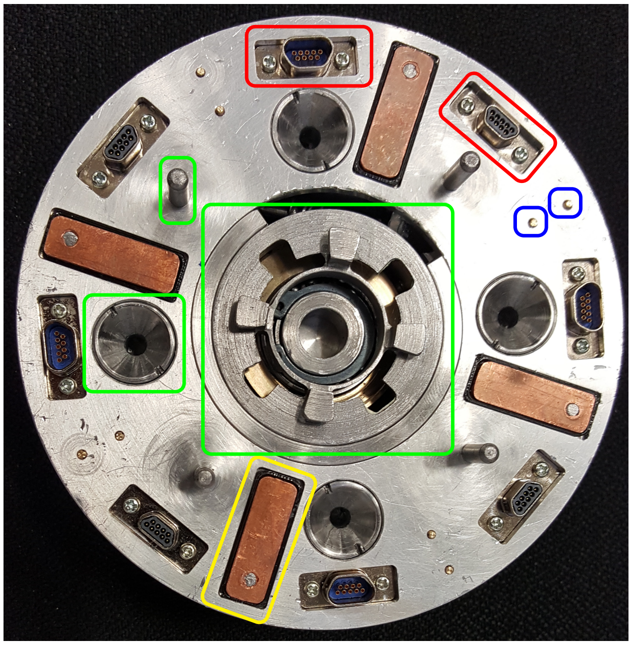


**Figure 3. Prototype of the standard interface. Green, red, blue and yellow boxes indicate the mechanical, data, power and thermal functionalities, respectively [26].**

Specifically, considering the manufacture of the IF out of a commercial steel and with a safety factor of two, the four steel pins, each $4\,\mathrm{mm}$ in diameter, ensure that the IF is able to transfer $4640\,\mathrm{N}$ of lateral (i. e. radial) force and $128\,\mathrm{Nm}$ of torque; the four teeth in the central mechanism of the IF ensure an axial lock during mating and are designed to sustain up to $440\,\mathrm{N}$ of force.

As per the design constraint, the IF has to allow a transfer of $30\,\mathrm{W}$ of thermal power. The numerical simulations and tests performed during the design stage determined that with a surface of $720\,\mathrm{mm}^2$ and contact pressure between the patches of $0.55\,\mathrm{MPa}$, a $29\,\mathrm{W}$ of thermal power would flow between the two sets of connectors that are at a thermal gradient of $10\,\mathrm{K}$.

In general, a single SpW connector is limited by hardware design to $50\,{}^{\mathrm{MB}}/\mathrm{s}$. Therefore, to satisfy the data rate constrain, the IF is equipped with eight pairs of connectors working in parallel, assuring a data rate of up to $400\,{}^{\mathrm{MB}}/\mathrm{s}$.

Each pair of +/- connectors on the IF has been designed to transfer $150\,\mathrm{W}$ of electrical power, for a total of $600\,\mathrm{W}$ considering all four pairs.

The $5\,\mathrm{mm}$ tolerance is achieved via pins and holes with a $5\,\mathrm{mm}$ chamfer visible in Figure 3.

The expected insertion forces to be generated during mating will largely depend on the sliding forces generated by the mechanical and data interfaces. The four pins of the mechanical interface will generate very low sliding force depending mainly on the accuracy of the manipulator and are therefore neglected. The eight data connectors are expected to require a total connection/disconnection force between $1.12$ and $13.36\,\mathrm{N}$, according to the specifications of the manufacturer[2].

One of the key requirement for the envisioned IF, and a crucial factor for any space application, is the fail-safeness. A product is fail-safe when it can maintain a certain degree of functionality even in case of failure of one (or more) main component(s).
The mechanical fail safeness of the designed IF has been achieved by developing a locking mechanisms that can disengage even if the counterpart does not cooperate, or if one of the two mechanisms stops cooperating after the two are mated.
The fail-safeness of other interfaces has been achieved by redundancy: data, thermal, and power connectors are replicated eight, four and again four times, respectively. In case of one (or more) connector fails, continuity of the flow and performances are ensured by switching to other, working, connectors. Moreover, the user can also choose how many and which pairs of connectors to use simultaneously, based on the scope of a mission and requirements of a payload [26].

## 5. APM Concepts

To validate the feasibility of APM concepts within the system requirements of the project, their functional and architectural analysis has been performed during the preliminary design phase of the SIROM project, and is outlined in this section. The outcome proves the feasibility of the imposed system requirements and outlines the most suitable concepts for the orbital and planetary scenarios that could be used as a reference for a more detailed design and development of APMs.

*System Requirements*

The system requirements for APM concepts have been defined in line with the reference scenarios and those of the SIROM interface, as detailed in Sections 1 and 4, respectively.
Specifically, it can be summarized that an APM shall:

- include at least two standard interfaces;
- include an optical sensor in the visible spectrum (i. e. in the $0.7 - 0.4\,\mu\mathrm{m}$ range) for Earth observation and/or in-orbit inspection, in case of the orbital scenario, or a lithium-ion polymer battery pack for on-board energy storage having a nominal voltage of $44.4\,\mathrm{V}$ and capacity of $5 - 20\,\mathrm{Ah}$, in case of the planetary scenario;
- have a mass $\leq 5.5\,\mathrm{kg}$ (without interfaces);
- have dimensions (without interfaces) $\leq 0.15 \times 0.15 \times 0.15\,\mathrm{m}$ and $0.15 \times 0.15 \times 0.24\,\mathrm{m}$ (L $\times$ W $\times$ H) for orbital and planetary APMs, respectively;
- comply with the capabilities of standard interfaces;
- endure the environmental conditions of the LEO (in case of the orbital scenario) and Moon/Mars (in case of the planetary scenario).

The APM housing configuration and integration of the prototype interface, illustrated in Section 4, has not been specified by the defined system requirements and has been therefore

[2] www.axon-cable.com/publications/D-LINE_P155_180_Space.pdf

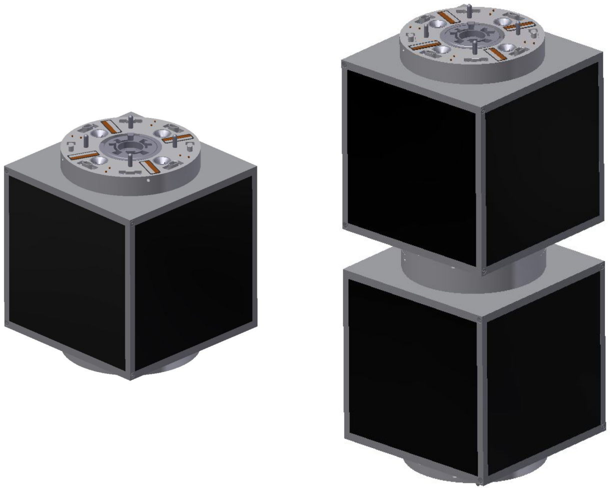


**Figure 4. CAD models of an APM housing with external interfaces in single and stack configurations, respectively.**

**Figure 5. CAD models of an APM housing with internal interfaces in single and stack configurations, respectively.**

identified in what follows.

Taking into account the characteristics of the prototype interface, two possible configurations have been identified between APM housings and standard interfaces as illustrated in Figures 4-5.

In both configurations, the APM housings are assumed to have maximum dimensions of the orbital APM and a CubeSat-like structure developed at DFKI for PLIs (see Figure 6).

The housing configuration with the externally mounted interfaces is presented in Figure 4 and has the following characteristics:

- an internal volume of $\sim 3.11 \times 10^{-3}\,\mathrm{m}^3$ (equivalent to that of a 3U CubeSat);

- an external contact area of $\sim\ 2 \times (1.13 \times 10^{-2}\,)\mathrm{m}^2$,

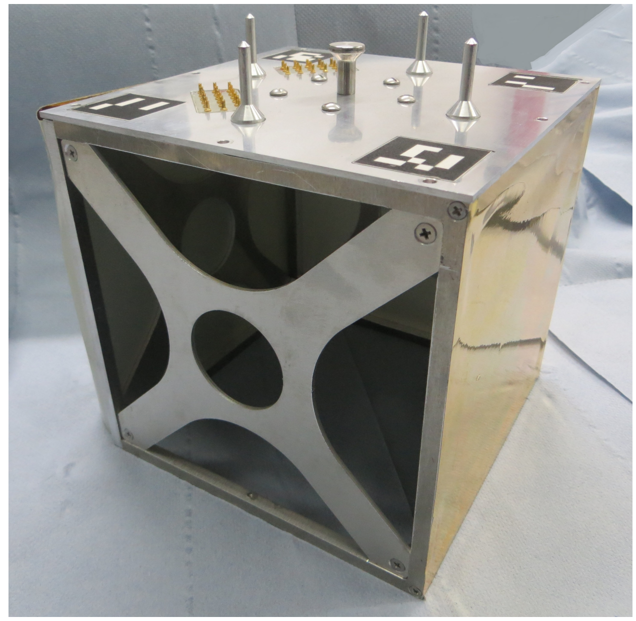


**Figure 6. Skeleton of a payload-item developed at DFKI [12].**

represented only by surfaces of the two interfaces;

- an unusable volume between the stacked APMs of $\sim 6.71 \times 10^{-4}\,\mathrm{m}^3$;

- a characteristic minimum internal width of $0.146\,\mathrm{m}$.

The housing configuration with internally mounted interfaces is presented in Figure 5 and has the following characteristics:

- an internal volume of $\sim 2.39 \times 10^{-3}\,\mathrm{m}^3$ (equivalent to that of a 2U CubeSat);

- an external contact area of $\sim 2 \times (2.25 \times 10^{-2}\,)\mathrm{m}^2$, represented by the two faces of the APM containing interfaces;

- no unusable volume in stack configuration;

- a characteristic minimum internal width of $0.086\,\mathrm{m}$.

From the Figures and above specifications it is clear that the configuration with external interfaces offers a simpler integration and the highest volume available for the payload. The disadvantages are mainly related to the smaller surface area upon which the external forces could be distributed, and confined only to the standard interfaces, which would be directly exposed to the outside environment and more likely be subject to damage, especially in case of the planetary scenario. Another disadvantage is represented by the unusable volume in stacked configuration, as visible in Figure 4. These disadvantages are nonexistent in the housing configuration with internally mounted interfaces. However, this is achieved at the expense of higher mounting complexity and $\sim 23\,\%$ reduction of the internal volume. With this in mind, and considering that the housing configuration with internally mounted IFs would represent the worst case scenario, with regards to the available internal volume, this was selected for the definition and evaluation of payload concepts.
Therefore, the major limitations of the modules are repre-

sented by their physical characteristics (e. g. dimensions and mass), which are in the range of those of CubeSat-sized systems. Nevertheless, due to advancements made in recent years in the miniaturization of space systems, the selection of CubeSat-sized sub-systems has been found to be not nearly as limited as expected, especially for Earth observation purposes [27].

*Orbital Payload Concepts*

The orbital APM will have the function of performing remote sensing in the visible spectrum of the Earth or immediate vicinity of a spacecraft. Therefore, it will need to have the capability of capturing, processing and storing images before transmitting them through an interface to a communication module or any other part of a modular spacecraft. With this in mind, candidate payloads have been evaluated with the goal of identifying a reference concept that could be used for a more detailed definition of orbital APM concepts.

*Candidates Definition*—The identification of candidate payloads and their characteristics has been performed using an analogy with existing CubeSat systems, given the similar available volume inside the orbital APM housing.
Potential candidates for the reference concept of the orbital APM have been identified using on-line databases of CubeSat components (e. g. [28], [29]). The sensors were mainly chosen based on their characteristic dimensions and interface capabilities. Off-the-shelf components could have been a compelling alternative; however, they were not considered at this stage, in order not to distort the feasibility perception of envisioned concepts given the additional bulk and reduced capacity that representative space-worthy technologies generally involve.

The list of potential payload candidates for the orbital APM therefore contained: Crystalspace C1U CubeSat Camera[3], Crystalspace Satellite Monitoring Camera[4], GOMSpace C1U NanoCam[5], Malin Space Science Systems (MSSS) ECAM-C30[6] and SCS Space Gecko Imager[7].
Detailed characteristics of the selected candidate systems are presented in Table 2.
All of the selected payloads are space-qualified and most of them can be customized with different sensors and lenses. However, for evaluation purpose, only their standard configurations have been considered.

*Evaluation and Baseline Analysis*—The evaluation of payload candidates and definition of the reference concept has been performed by first comparing the characteristics of the identified candidates against the system requirements (outlined in Subsection 5), and then using the characteristics of a selected reference payload to assess mass and power requirements of the baseline orbital module.
Following the outlined procedure, the Crystalspace and GOM C1U NanoCam have been identified as the most suitable payloads for the orbital APM concept. Among the two, the GOM C1U NanoCam is found to be the most exigent in terms of mass, volume and power requirements. Therefore, it was selected for the definition of mass and power estimates of the overall orbital APM concept.
The MSSS ECAM-C30 system was also found to be a compelling candidate, however it was discarded considering that an additional module will need to be entirely allocated to its digital video recorder (DVR), having considerable dimensions (i. e. $122 \times 218 \times 31\,\mathrm{mm}$ (L $\times$ W $\times$ H)), required for processing and storing images from the ECAM-C30.

The GOM C1U NanoCam is a 3 Megapixel ($2048 \times 1536$) color complementary metal-oxide-semiconductor (CMOS) modular system designed for standard 1U CubeSat structures and capable of on-board data processing and storage. It offers a CubeSat Space Protocol (CSP)-enabled CAN, I2C, and TTL level serial interfaces but no SpW interface. Its overall mass ($M_{P/L}$), increased by a safety margin of $5\,\%$, amounts to $0.291\,\mathrm{kg}$.

The mass of individual subsystem of the orbital APM was therefore estimated as a percentage of the payload mass ($M_{P/L}$) [30] (see Table 3). Specifically, considering the payload to be the $20\,\%$ of the overall mass of the module [30], the dry mass ($M_{dry}$) of the module was estimated to be $1.455\,\mathrm{kg}$.
The structural mass of the module ($M_{sam}$) was estimated to be $0.341\,\mathrm{kg}$, based on the $M_{sam}$ of DFKI's PLIs, developed for a similar purpose [12], with an additional safety margin of $2\,\%$ to account for hardware integration [30].
The masses of thermal ($M_{th}$) and power ($M_{ep}$) subsystems have been estimated to be $16\,\%$ of the $M_{P/L}$ [30], i. e. equal to $0.047\,\mathrm{kg}$, to account for the cabling and any supporting equipment.
This is found to be a safe hypothesis, given the nature of the APMs and the existence of integrated standard interfaces which should provide APMs with electrical power and thermal management without the need for additional integrated subsystems.
The mass of the Command and Data Handling (C&DH) subsystem ($M_{cadh}$), the control block of the module, has been assumed to be $0.07\,\mathrm{kg}$, a typical value for a CubeSat-sized component of this kind.

Based on the individual masses of subsystems delineated previously, their total mass ($M_{SS}$) was estimated to be $0.796\,\mathrm{kg}$, resulting in a system level margin ($M_{mar}$) of $0.66\,\mathrm{kg}$, i. e. of $45.4\,\%$, well above the recommended $25\,\%$ [30], that can be allocated to the payload or any other subsystem should it be necessary during a more detailed development of the orbital APM module.

The power estimate of the reference module was obtained in a similar fashion.
Starting from the peak power of the selected payload, and considering it responsible for a conservative $30\,\%$ of the overall power consumption [30], a preliminary average power requirement of the module has been estimated equal to $4.16\,\mathrm{W}$, well within the capabilities of the prototype interface outlined in Section 4.

*Planetary Payload Concepts*

The planetary APM will be a rechargeable power storage unit able to provide more flexibility and autonomy to planetary rovers and/or other APMs. The module will therefore need to attend to all the necessary functionalities to assure a specific bus voltage and current during its charge/discharge, as well as safe states of its individual battery packs and/or cells. With this in mind, the most suited candidate payloads have been found and evaluated, and a baseline was selected and used for

[3]http://crystalspace.eu/products/crystalspace-c1u-cubesat-camera/
[4]http://crystalspace.eu/monitoring-camera/
[5]https://gomspace.com/Shop/payloads/earth-observation.aspx
[6]http://www.msss.com/brochures/c30.pdf
[7]http://www.scs-space.com/pdf/SCS_Space_Gecko_Brochure_2016-11-14.pdf

**Table 2. Characteristics of candidate systems for remote sensing in the visible spectrum**

| Name | Dimensions (mm) | Mass (kg) | Avg. Pwr. (W) | Data Rate & IF (Mbps) | Storage (GB) |
|---|---|---|---|---|---|
| **Crystalspace C1U Camera** | $45 \times 25 \times 45$ | $\leq 0.05$ | 0.08 | NaN; SpW,CAN, UART, I2C, SPI | NaN |
| **Crystalspace Monitoring Camera** | $88 \times 82 \times 39$ | 0.27 | 0.5 | NaN; SpW, CAN, UART, USB, I2C, SPI | NaN |
| **GOM C1U NanoCam** | $86 \times 92 \times 97$ | 0.277 | 0.9 | $\leq 1$; CAN, I2C, TTL | 2 |
| **MSSS ECAM-C30** | $58 \times 44 \times 78$ | 0.356 | 2.1 | 100 to DVR; SpW | None |
| **SCS Gecko Imager** | $97 \times 60 \times 96$ | $\leq 0.48$ | NaN | NaN; LVDS, SPI, I2C | 128 |

**Table 3. Mass budget of the reference orbital APM concept**

| Subsystem | Mass (kg) | Comments |
|---|---|---|
| Payload ($M_{P/L}$) | 0.291 | with 5 % margin |
| Structure ($M_{sam}$) | 0.341 | with 2 % margin |
| Thermal ($M_{th}$) | 0.047 | $0.16 \times M_{P/L}$ |
| C&DH ($M_{cadh}$) | 0.07 | typical value |
| Power ($M_{ep}$) | 0.047 | $0.16 \times M_{P/L}$ |
| Total subsystems ($M_{SS}$) | 0.796 | |
| Margin ($M_{mar}$) | 0.66 | 45.4 % of dry mass |
| Total dry mass ($M_{dry}$) | 1.455 | $M_{P/L}/0.2$ |

the characterization of the overall reference concept of the planetary module.

*Candidates Definition*—The identification of candidate payloads and their characteristics was performed as in case of the orbital payloads by using the analogy with existing CubeSat systems, given the restricted dimensions of the APM housing and available internal volume ($\sim 4.18 \times 10^{-3}\,\mathrm{m}^3$). Therefore, using on-line databases of CubeSat components (e. g. [28], [29]) it was possible to identify candidate payloads for the reference concept of the planetary APM based on their dimensions and mass. Once again, off-the-shelf components could have been a compelling alternative, but were not considered in order not to distort the perception of the feasibility of envisioned concepts given the additional bulk and reduced capacity that representative space-worthy technologies generally involve.
The most suitable power storage unit candidates were: Clyde Space CubeSat Battery[8], EXA Pegasus Class BA01/D[9], GOM NanoPower BPX[10], GOS CubeSat Battery[11], IMT Battery Pack[12], MAI FlatPack Battery[13] and SIL Intelli-Pack Battery[14].
All of the mentioned battery packs are space-qualified and have been specifically designed with limited dimensions in mind. Furthermore, most of them are customizable; however, in order to make a fair evaluation of the candidates, only their standard configurations have been here considered.
Detailed characteristics of those configurations are presented in Table 4.

*Evaluation and Baseline Analysis*—Comparing the characteristics of the candidates with the system requirements, it was possible to select the EXA Pexasus Class BA01/D and IMT battery packs as possible baselines for the payload of the planetary APM. The reason for choosing those two systems was due to their high energy density compared to other options. In fact, considering the internal volume limitations of the planetary APM housing, four sets of six EXA Pexasus Class BA01/D battery packs can be connected in parallel to achieve a nominal voltage of $44.4\,\mathrm{V}$ and a capacity of $28.8\,\mathrm{Ah}$, in a package having a mass of $4.32\,\mathrm{kg}$. In case of the IMT battery pack, two sets of six packs in series can be connected in parallel to achieve a nominal voltage of $44.4\,\mathrm{V}$ and a capacity of $24.8\,\mathrm{Ah}$ having mass of only $2.88\,\mathrm{kg}$.
From the previous estimates, it is obvious that among the selected baselines, the EXA Pexasus Class BA01/D battery pack is the most exigent system in terms of mass and volume. Therefore, it was selected for the mass budget characterization of the module, as detailed in Table 5.

[8] https://www.clyde.space/products/49-40whr-cubesat-battery
[9] https://www.cubesatshop.com/product/ba0x-high-energy-density-battery-array/
[10] https://gomspace.com/Shop/subsystems/batteries/nanopower-bpx.aspx
[11] http://www.orbitalsystems.de/produkte/gos-cubesat-battery-pack/?lang=en
[12] http://www.imtsrl.it/battery-pack.html
[13] http://maiaero.com/components/
[14] http://spaceinformationlabs.com/products/intelli-pack-battery-technology/

**Table 4. Characteristics of candidate systems for power storage**

| Name | Dimensions (mm) | Mass (g) | Nom. Voltage (V) | Nom. Capacity (Ah) | Telemetry Interface |
|---|---|---|---|---|---|
| **Clyde Space CubeSat Battery** | $95 \times 90 \times 27$ | 447 | 7.6 | 5.2 | I2C |
| **EXA Pegasus Class BA01/D** | $89 \times 95 \times 14$ | 180 | 7.4 | 7.2 | NaN |
| **GOM NanoPower BPX** | $92 \times 86 \times 41$ | 500 | 14.8 | 5.2 | I2C |
| **GOS CubeSat Battery** | $91 \times 96 \times 24$ | 325 | 7.4 | 10.4 | NaN |
| **IMT Battery Pack** | $96 \times 90 \times 25$ | 240 | 7.4 | 12.4 | NaN |
| **MAI FlatPack Battery** | $96 \times 83 \times 41$ | 710 | 16.8 | 2.74 | NaN |
| **SIL Intelli-Pack Battery** | $140 \times 119 \times 90$ | 2268 | 33.6 | 6.6 | I2C |

The EXA Pexasus Class BA01/D is a $14\,\mathrm{mm}$ thick double-sided battery array weighing $180\,\mathrm{g}$. Therefore, a power storage system of four sets of six packs is assumed to be connected in parallel having a total mass ($M_{P/L}$) of $4.536\,\mathrm{kg}$, including a $5\,\%$ safety margin.

Given the demanding characteristics of the payload in question in terms of mass, the total dry mass of the module ($M_{dry}$) was assumed to be equal to the maximum allowable mass of the planetary APM, i. e. equal to $5.5\,\mathrm{kg}$.
The mass of the structure of the module ($M_{sam}$) was estimated to be $0.511\,\mathrm{kg}$, under the assumption that the dimension of the planetary APM housing is 1.5 times that of the orbital APM.

A single BA01/D battery pack does not include any onboard electronics and therefore has no means to maintain an optimum temperature of batteries within the APM. Nevertheless, batteries can optionally be provided with a Carbon Nanotubes Thermal Transfer Bus (CN/TTB) capable of transferring heat[15] to the batteries without the need for active heaters. At the same time, the bus can also provide excellent radiation shielding allowing cost and mass reduction of the overall system [31]. Therefore, the mass of the thermal subsystem ($M_{th}$) was assumed to be equal to that of the orbital module, i. e. equal to $0.047\,\mathrm{kg}$, to account for any cabling and supporting equipment.
The mass of power subsystem of the module ($M_{ep}$) was also assumed to be equal to that of the orbital APM with an added mass of two electric power system (EPS) boards, for a total mass equal to $0.367\,\mathrm{kg}$.

Based on previous values the total estimated mass of all subsystems of the planetary module ($M_{SS}$) was evaluated to be $5.461\,\mathrm{kg}$, thus resulting in a system level margin of $< 1\,\%$, below any recommended value [30]. Nevertheless, this value is found to be within the system requirements and it could be lowered by either choosing an alternative battery pack (e. g. the IMT Battery Pack) or by a more in depth analysis of the mass budget of the planetary APM, which is out of the scope of the present paper.

[15] That could be originating from the standard interface, for example.

## 6. END-EFFECTOR CONCEPTS

As in case of APM reference concepts, a functional and architectural analysis of end-effector concepts was performed during the preliminary design phase of the SIROM project, to validate the feasibility of an end-effector (EE) concept, within the system requirements of the SIROM project, and identify the reference end-effector concept that could be used to develop its detail designs within the next steps of the project.

*System Requirements*

The system requirements of end-effector concepts have been defined based on the reference scenarios outlined in Section 1 while also considering the operational limitations of the testing facilities. Therefore, it can be summarized that an end-effector shall:

- support mechanical, electrical and data IFs of: the KUKA Lightweight Robot (LWR) (to be provided by the German Aerospace Center (DLR)) during tests of the orbital scenario, and manipulator of SherpaTT (to be provided by DFKI) during tests of the planetary scenario.

- present an electromagnetic compatibility with coupled APMs;

- be able to operate in the orbital and planetary environments;

- include one standard IF;

- be based on the end-effector of the manipulator arm of the SherpaTT rover (see Figure 7);

- not exceed the footprint of the end-effector of the manipulator of the SherpaTT rover, i. e. $\leq 150 \times 150\,\mathrm{mm}$ (L × W).

*End-Effector Concepts*

With the system requirements in mind, the reference end-effector concept shall consist of a housing, standard interface and mechanical misalignment system (e. g. Schunk AGE-F-XY-063-2).

Table 5. **Mass budget of the reference planetary APM concept**

| **Subsystem** | **Mass** (kg) | **Comments** |
|---|---|---|
| Payload ($M_{P/L}$) | 4.536 | with $5\,\%$ margin |
| Structure ($M_{sam}$) | 0.511 | with $2\,\%$ margin |
| Thermal ($M_{th}$) | 0.047 | from Table 3 |
| Power ($M_{ep}$) | 0.367 | including 2 EPS boards |
| Total subsystems ($M_{SS}$) | 5.461 | |
| Margin ($M_{mar}$) | 0.039 | $< 1\,\%$ of dry mass |
| Total dry mass ($M_{dry}$) | 5.5 | Max allowable mass |

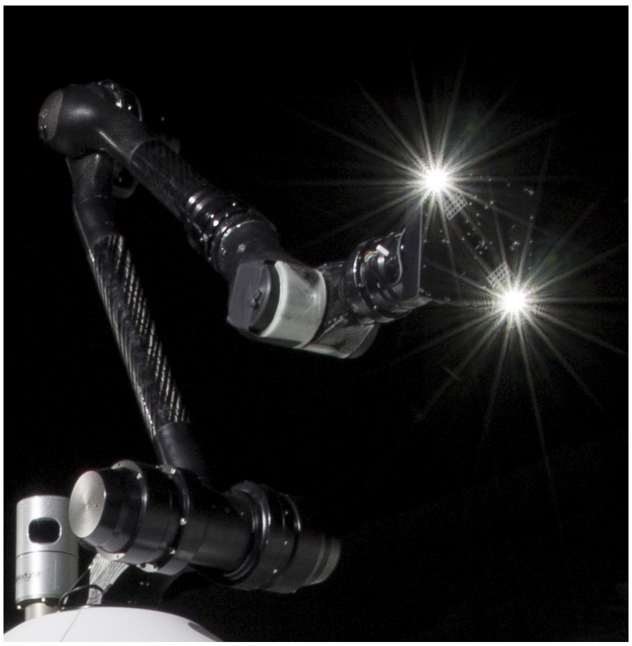
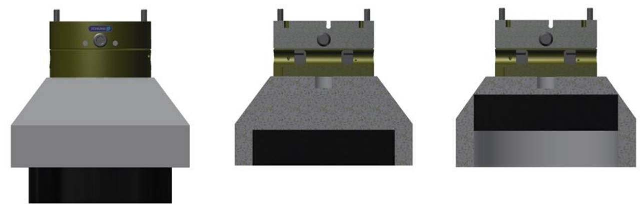


**Figure 7. End-effector of SherpaTT rover (credit: DFKI 2015).**



(a)EE Version 1 (b)EE Version 2 (c)EE Version 3

**Figure 8. Candidate end-effector concepts used for the EE reference analysis.**

The housing shall include all the necessary assemblies, like electronics for the interface, connectors and an FTS.
Considering the housing of a reference end-effector similar to that of the end-effector of the SherpaTT manipulator [32], and the concept interface described in Section 4, three candidate EE versions have been identified, as illustrated in Figure 8.
They all differ only in the way the interface (represented in the figure by black rectangles) is integrated within the EE. More details about each version are outlined hereafter.

In the Version 1 of the EE (see Figure 8(a)), the interface is mounted on its external surface. The advantage of this configuration consists of a large housing where electronics and other necessary parts can be enclosed. The disadvantage consists of the overall large dimensions of the EE, especially its height ($152\,\mathrm{mm}$), which reduces the maximum height of an APM to only $223\,\mathrm{mm}$.

In the Version 2 of the EE (see Figure 8(b)), the interface is integrated within its housing such that the external surfaces of the IF and housing are planar. This configuration is the most compact one and ensures the highest stability during the manipulation of a module with internally mounted interfaces (see Figure 5), since the forces can be redistributed over a larger surface. However, this comes at a cost of reduced internal volume available for electronics and other parts which would require more compact solutions having higher cost and requiring more difficult integration procedures.

The Version 3 of the EE (see Figure 8(c)) is similar to the second one, since an interface is also integrated within the EE housing, and is specifically envisioned to be compact while providing a more stable connection between the manipulator and APM, having externally mounted interfaces (see Figure 4). However, in this configuration the interface is mounted $30\,\mathrm{mm}$ deeper inside of the EE housing reducing the available internal volume of the housing compared to the second version, worsening its disadvantages.

Details of the identified EE candidates are outlined in Table 6, where the end-effector of the SherpaTT manipulator is included only as reference.
The column titles of the table are defined as follows:

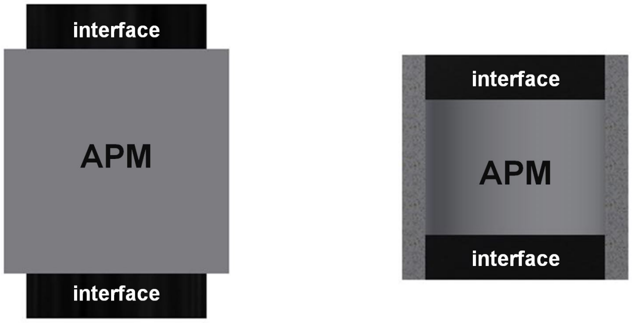
- assembly space: the overall internal dimensions of the end-effector housing available for electronics, cabling and other necessary parts;

- dimensions: the overall external dimensions of the end-effectors, including the interfaces;

- resilience: the ability of an EE to compensate a misalignment in any directions;

- contact area after docking: the surface of the contact area between an EE and a docked APM;

- max load to grasp: the maximum force sustainable by an EE;




(a)APM Version 1 (b)APM Version 2

**Figure 9**. **Schematic representation of APM configurations used for the EE reference analysis.**

- max load/contact area: the maximum pressure sustainable by an EE;
- measurement detection system: the type of system used for the detection of the relative pose[16] of an APM with respect to an EE.

*End-Effector-APM Combinations and Baseline Selection*

As mentioned in Section 5, the APM housing configuration and integration with the prototype interface has not been specified by the system requirements, leaving two possible configurations (see Figures 4 and 5 or Figure 9) to be taken into consideration during the functional analysis of APM and end-effector concepts.
Therefore, using the three identified EE concepts and the two APM housing configurations, it was possible to determine in total five feasible combinations, illustrated in Figure 10, to be evaluated for the identification of the reference concept of the end-effector.

In the Combination 1 (see Figure 10(a)), the interface is positioned outside of both the APM and end-effector. This configuration allows for the end-effector and APM to have a maximum height of $152\,\mathrm{mm}$ and $223\,\mathrm{mm}$, respectively.

In the Combination 2 (see Figure 10(b)), the interface is positioned outside of the APM and inside the end-effector, so that the lower side of the interface is planar to the bottom of the end-effector. This configuration allows for the end-effector and APM to have a maximum height of $122\,\mathrm{mm}$ and $253\,\mathrm{mm}$, respectively.

In the Combination 3 (see Figure 10(c)), the interface is positioned outside the APM and inside the end-effector at a distance of $30\,\mathrm{mm}$ from its external surface. In this configuration the maximum heights of the end-effector and APM are $122\,\mathrm{mm}$ and $283\,\mathrm{mm}$, respectively.

In the Combination 4 (see Figure 10(d)), the interface is located inside the APM, planar to its top and bottom surfaces, and outside of the end-effector. In this configuration the maximum heights of the end-effector and APM are $152\,\mathrm{mm}$ and $223\,\mathrm{mm}$, respectively.

Finally, in the Combination 5 (see Figure 10(e)), the interface is integrated inside both the APM and end-effector, thus favoring the $122\,\mathrm{mm}$ and $253\,\mathrm{mm}$ maximum heights of the end-effector and APM, respectively, while ensuring lowest torques on the end-effector and high grasping stability during the operation of a manipulator arm.

Taking into considerations the outlined end-effector concepts and their combinations with APMs, the best possible EE concept was determined to be the version two (see Figure 8(b)) illustrated in more detail in Figure 11 featuring: internally mounted interface (flush with external surface of the EE), compact dimensions (i. e. $150 \times 150 \times 122\,\mathrm{mm}$), moderate assembly space (i. e. $1.43 \times 10^6\,\mathrm{mm}^3$), high contact area (i. e. $2.25 \times 10^4\,\mathrm{mm}^2$), moderate load to contact area (i. e. $5.56 \times 10^3\,{}^{\mathrm{N}}\!/\!_{\mathrm{m}^2}$) and moderate wrist torque (due to compact dimensions).

[16]Defined as the position and orientation.

## 7. CONCLUSIONS

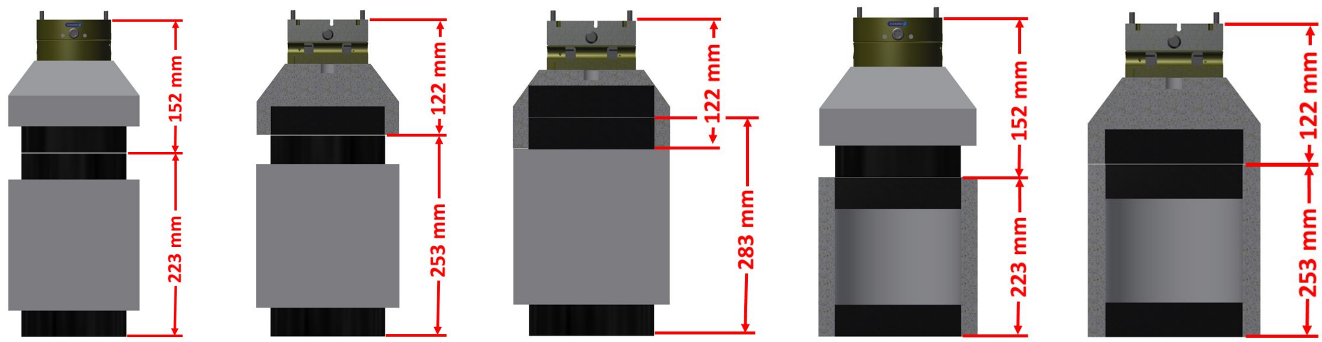
In this document the work performed by DFKI and University of Strathclyde during the preliminary design phase of the SIROM project is outlined.
More specifically, frameworks for the definition and evaluation of APM and end-effector concepts within the project have been established based on the methodology used in the design and sizing of payloads of space missions. Based on these frameworks, the feasibility of APMs and end-effector concepts, within the system requirements of the SIROM project, has been validated and the most suitable concepts for future studies have been outlined.
The configuration space of APMs has been confined to optical remote sensors in the visible spectrum and electrical power storage systems in case of the orbital and planetary scenarios, respectively.
The APM housing configuration with internal interfaces has been assumed as a baseline for the evaluation of APM concepts, given its advantages in stack configuration despite the slight reduction of the available internal volume.
The APM concepts were defined and evaluated in terms of their payload systems, due to the fact that in any spacecraft, and more generally in any space mission, a payload definition and sizing is what generally determines its capabilities, limitations and thus size, cost and risk.
The reference concepts for the orbital and planetary APMs integrate the GOM C1U NanoCam as an optical sensor, and a 24 EXA Pexasus Class BA01/D battery pack, respectively.
In a similar way, the configuration space of end-effector concepts has been confined to that dictated by the orbital and planetary test scenarios, as well as by the system requirements. Three end-effector concepts have been defined, and the one consisting of an internally mounted IF, flush with the external surface of the end-effector, has been deemed as the most suited concept for further studies.
The results of this analysis prove the feasibility of APMs and end-effectors, within the system requirements of the project, and define characteristics of concepts that could be used by the SIROM consortium partners as a reference for the detailed design and development of APMs and end-effectors that will ultimately test the practical usage of APMs and the SIROM interface currently being developed within the consortium.

## ACKNOWLEDGMENTS

The authors would like to thank all the supporting staff of the University of Strahclyde and DFKI GmbH along with partners of the SIROM Project: SENER Ingenieria y Sis-

**Table 6. Characteristics of end-effector candidates**

| EE N. | Designation | Assembly space ($mm^3$) | Dimensions ($L \times W \times H$) (mm) | Resilience | Contact area after docking ($mm^2$) | Max load to grasp (N) | Max load/con. area ($^N/_{m^2}$) | Meas. detection system |
|---|---|---|---|---|---|---|---|---|
| **1** | End-effector SherpaTT | $0.93\times10^6$ | $154\times154\times75$ | NaN | $2.37\times10^4$ | 250 | $1.05\times10^4$ | camera and markers on IF |
| **2** | End-effector version 1 | $1.76\times10^6$ | $150\times150\times152$ | $\pm4$ mm in $x$,$y$ axis | $1.13\times10^4$ | 125 | $1.11\times10^4$ | camera on EE |
| **3** | End-effector version 2 | $1.43\times10^6$ | $150\times150\times122$ | $\pm4$ mm in $x$,$y$ axis | $2.25\times10^4$ | 125 | $5.56\times10^3$ | camera on EE |
| **4** | End-effector version 3 | $1.09\times10^6$ | $150\times150\times122$ | $\pm4$ mm in $x$,$y$ axis | $5.64\times10^4$ | 125 | $2.22\times10^3$ | camera on EE |




(a)Combination 1 (b)Combination 2 (c)Combination 3 (d)Combination 4 (e)Combination 5

**Figure 10. Feasible EE-APM combinations considering three EE and two APM configurations.**

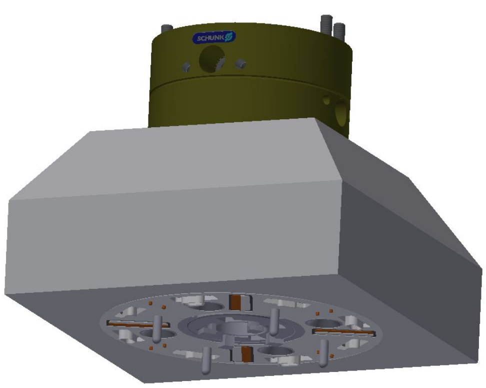


**Figure 11. CAD model of the reference concept of the end-effector.**

temas S.A. (Spain), Airbus DS Ltd (UK), Airbus DS GmbH (Germany), Thales Alenia Space S.p.A (Italy), Leonardo S.p.A. (Italy), TELETEL S.A. (Greece), Space Applications Services N.V. (Belgium) and MAG SOAR S.L. (Spain).
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