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Cross Cutting Technologies

Author: Prof. Raffaele Savino
University of Naples "Federico II", Naples, Italy, rasavino@unina.it

Eng. Renato Aurigemma
Euro.Soft Srl (ALI Scarl), Napoli, Italy, r.aurigemma@eurosoftsrl.eu

Dr. Pasquale Dell'Aversana
Astro Industry Srl (ALI Scarl), Napoli, Italy, pasqualedellaversana@astroindustry.com

Eng. Luciano Gramiccia
SRS Engineering Design Srl (ALI Scarl), Roma, Italy, luciano.gramiccia@ed.srs.it

Dr. Luca Ferracina
ATG Europe B.V. - ESA/ESTEC, Noordwijk, The Netherlands, Luca.Ferracina@aoes.com

Dr. José Longo
ESA/ESTEC, Noordwijk, The Netherlands, Jose.Longo@esa.int

Dr. Lionel Marraffa
ESA/ESTEC, Noordwijk, The Netherlands, Lionel.Marraffa@esa.int

Eng. Francesco Punzo
ALI Scarl, Napoli, Italy, francesco.punzo@aliscarl.it

IRENE PRELIMINARY STUDY (ITALIAN RE-ENTRY NACELLE PRELIMINARY STUDY)

ABSTRACT

ESA is supporting a study to develop a small flight demonstrator of a re-entry capsule with deployable heat shield to be embarked as a secondary payload in the inter-stage adapter of a sub-orbital Maxus sounding rocket. This capsule will be ejected during the ascent phase of the rocket payload section, after its separation from the booster, to demonstrate capsule release, heat shield deployment, safe re-entry and recovery of payload. The deployable thermal shield is designed as a low-cost, disposable system, exploiting commercial materials to a large extent. The concept may be applied to any ESA mission in the framework of the Science and Robotic Exploration Program, requiring planetary entry and, possibly also to ESA missions in the framework of Human Spaceflight, requiring planetary entry or re-entry. The technology offers also an interesting potential for aerobraking/aerocapture and for de-orbiting of small satellites or payloads.
The planned Maxus experiment will serve as a technology proof-of-concept whereby larger capsules, with a low-cost, disposable heat shield, could exploit similar technical solutions for Earth observation, short-notice, orbital missions, both for civil and military purposes.
This paper summarizes the results of preliminary activities, regarding the Maxus flight scenario with the related system requirements, the presentation of different possible solutions for the deployable heat shield and a first selection of the avionics and sensors to be embarked.

1. BACKGROUND OF THE PROJECT

The Italian Space Agency (ASI) is supporting a research programme, called IRENE, carried out in Campania region (South of Italy) by a cluster of industries, research organizations and universities, to develop a low-cost re-entry capsule, able to return payloads from the ISS to Earth and/or to perform short-duration, scientific missions in Low Earth Orbit (LEO).
The ballistic capsule is characterized by a deployable "umbrella-like" heat shield, that allows relatively small

dimensions at launch and a sufficient exposed surface area in re-entry conditions, reducing the ballistic coefficient and leading to acceptable heat fluxes, mechanical loads and final descent velocity.
The main features of the IRENE capsule are:

- light weight (100-200 Kg)
- payload recoverability and reusability
- a low-cost, deployable, disposable heat shield composed by:
  - a fixed nose (made by ceramic or other equivalent TPS)
  - a deployable aero-brake (umbrella-like, made by special multi-layered fabric).

The feasibility study of this deployable re-entry system has been carried out in 2011. The main outcome of this activity is summarized in Ref. [1]. In particular, the TPS materials, selected for the nose cone and for the flexible umbrella shield, have preliminarily been tested in the SPES hypersonic wind tunnel at the University of Naples, and in the SCIROCCO PWT (Plasma Wind Tunnel) at CIRA (Centro Italiano Ricerche Aerospaziali) of Capua, Italy. Such successful tests and the preliminary experimental results proved the concept feasibility and the viability of commercial materials for low-cost re-entry nacelles, pushing the Italian and the European Space Agencies to further support the development of the programme.

Fig. 1 - IRENE TPS test in the SCIROCCO Plasma Wind Tunnel at CIRA (From Ref. [1]).

2. MINI-IRENE DEMONSTRATOR FOR SUB-ORBITAL SOUNDING ROCKETS

On the basis of the initial results, the European Space Agency supported a six months "Bridging Phase" to preliminarily address the main issues of a Mini-IRENE demonstrator, to be embarked as a piggy-back payload in a future mission of a sub-orbital Maxus sounding rocket. Similarly to the successful SHARK (Sounding Hypersonic Atmospheric Re-entering Kapsule) Maxus-8 mission in 2010 [2], the Mini-Irene system shall be included as a secondary payload in the interstage adapter of the Maxus and ejected, during the ascent phase of the payload section, after its separation from the booster, at an altitude of about 150 km, to perform 15 minutes of ballistic flight and then re-enter in the atmosphere and hit the ground. The main characteristics of the demonstrator are the "umbrella-like" deployable front structure that, like the larger scale IRENE system, reduces the capsule ballistic coefficient, leading to acceptable heat fluxes, mechanical loads and final descent velocity. Other examples of similar aerobraking reentry capsules investigated in the past include both inflatable and mechanically deployable systems [3-6].
A possible launch of a demonstrator of IRENE from a sounding rocket will require scaling down the most important parameters already identified for a standard, 3-meter capsule (maximum diameter in deployed configuration).
In particular, preliminary designs of the heat shield structure and of the deployment mechanisms have been developed according to the main concepts and requirements identified for each of these subsystems.
The work has been done, considering the mass and volume requirements to comply with the mass budget and volumetric envelope provided by the Maxus interstage.
In detail, the following issues have been addressed:

- Identification of a possible sub-orbital sounding rocket mission with the mini-IRENE technology demonstrator.
- Preliminary aerodynamic and aero-thermodynamic analysis.
- Analysis of the time profiles of the different physical parameters of interest (e.g. pressure, temperature, acceleration).
- Identification of the main mission requirements and corresponding subsystems.
- Trade-off between different configurations and identification of possible solutions for the different subsystems.

2.1. AERODYNAMIC AND AERO-THERMODYNAMIC ANALYSIS

Considering a cylindrical volume with a diameter of 29 cm, total length of 25 cm, total reentry mass in the order of 15-20 kg, different configurations have been investigated. In particular, engineering methods and CFD models have been utilized to evaluate the aerodynamic stability properties and the hypersonic drag coefficient of two sphere-cone configurations with half-angle of 45 and 60 deg respectively.
On the basis of the computed drag coefficients, possible re-entry trajectories have been re-computed considering the same total mass, and therefore, due to

the different drag coefficients and different exposed surface areas, different ballistic coefficients.

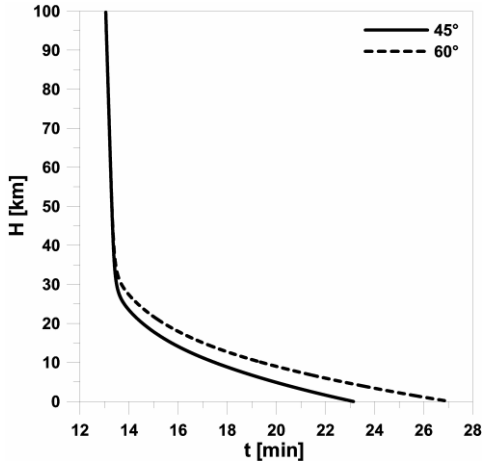

Fig.2 - Altitude versus time

Figures 2 to 5 show results corresponding to the configurations whose main parameters are summarized in table 1.

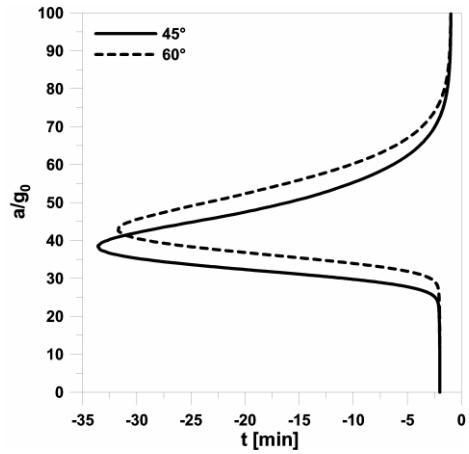

Fig. 3 - Acceleration versus time

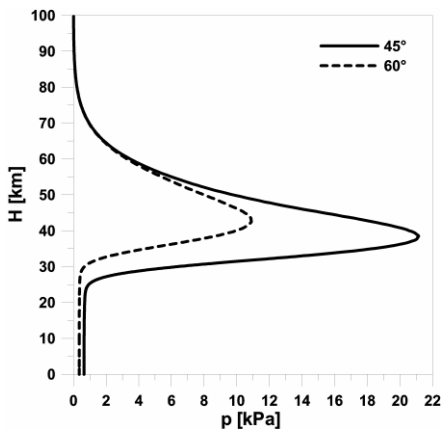

Fig. 4 - Altitude versus stagnation point pressure

Based on viscous CFD computations, heat flux and pressure distributions along the two configurations, have been estimated for maximum heat flux and pressure conditions along the trajectory. Examples of pressure distributions are shown in Fig. 5.

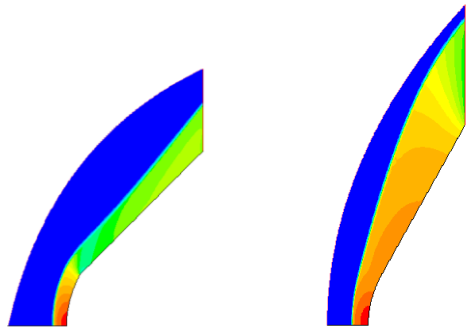

Fig. 5 - Pressure distributions at the condition of maximum dynamic pressure (left: 21 levels, step 1 kPa, maximum 21 kPa, right: 21 levels, step 500 Pa, maximum 10.5 kPa)

Half-cone angle [°]	D [m]	S [m^2]	c_D	$\beta = m/(c_D*S)$
60	1	0.79	1.4	18
45	0.84	0.58	1.03	33

Table 1: Main geometric and aerodynamic characteristics (D is the maximum diameter in deployed configuration). The ballistic coefficient is evaluated for a reentry mass M=20 kg.

It is evident that, due to the steeper reentry trajectory, caused by the larger ballistic coefficient, the capsule with 45 deg half-cone angle experiences the highest heat fluxes and impact pressures, in particular on the nose-cone. It must be also pointed out that, due to the relatively blunter configuration, the capsule with 60 deg half-cone angle exhibits comparable heat fluxes and pressures on the conical surface. It is important to point out that, in the case shown in Fig. 4, the performances obtained with the configuration with 45 deg half-cone angle are not acceptable because, assuming the same length of the poles, this shape is characterized by a smaller maximum diameter and therefore, assuming the same total mass, by a larger ballistic coefficient. Other systems, characterized by a half-cone angle of 45 deg and longer poles, can reach a diameter in the order of 1m, and a ballistic coefficient comparable to the 60 deg half-cone angle configuration. They have been investigated in paragraph 3 (solution 1) and offer the advantage of a better static and dynamic stability.

2.2. MAIN SYSTEM AND MISSION REQUIREMENTS

On the basis of the analysis made above, the main geometrical and mechanical requirements for the small-scale IRENE re-entry demonstrator shall include:

1. Deployable heat shield for a cylindrical capsule having the twofold function of payload container and supporting structure for said heat shield ;
2. Compatibility with Maxus interstage interfaces;
3. Sphere-cone aeroshape, in deployed configuration, with half-cone angle between 45 and 60 deg;
4. Maximum diameters of 29 cm and 100 cm in folded and deployed configuration, respectively;
5. Total mass less than 20 kg;
6. Ballistic coefficient less than or equal to 18 kg/m^2;
7. CG location placed as forward as possible and close to the symmetry axis, to guarantee aerodynamic stability and to reduce the ballistic trim angle of attack;
8. Automatic system for TPS deployment to be performed in the exo-atmospheric phase;
9. Correct attitude attained before heat shield deployment;
10. Structure able to withstand mechanical loads in the launch phase, and aerodynamic loads in the atmospheric reentry phase (10000 Pa stagnation point pressure, 40g aerodynamic deceleration), impact loads at landing (velocity in the order of 20 m/s);
11. Thermal protection system able to withstand heat fluxes in the order of 350 kW/m^2;

In addition, the following avionics and instrumentation sub-system have to be considered:

12. Data Handling for avionics and instrumentation sensors;
13. Telemetry (optional);
14. Beacon for recovery after landing;
15. Sensors including, for instance: temperature, pressure, CCD camera, three axis accelerometer, attitude sensor;
16. Power batteries for a total budget based on 1h mission duration.

3. PRELIMINARY DESIGN OF THE SUPPORTING STRUCTURE

Different solutions for the TPS supporting structure have been considered. During study advantages and disadvantages of each configuration have been investigated.

The basic designs of three different solutions are represented and explained below. They all present two basic concepts in common:

- The deployable part of the heat shield is a tensile structure;
- Deployment is performed in two phases:
  i) coarse elongation, characterized by long strokes (of poles or arms) and realively weak forces;
  ii) tensioning, characterized by shorter strokes (of a sliding structure) and stronger forces.

Solution 1:

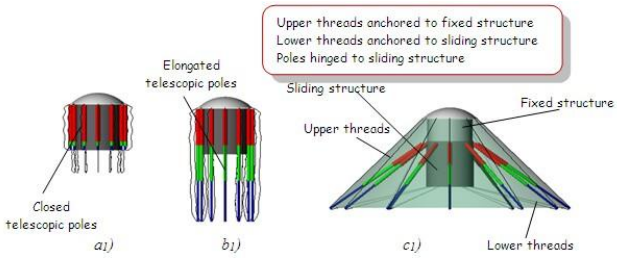

Fig. 6 - Solution #1: a1) folded structure; b1) pole elongation phase; c1) tensioning phase.

Solution #1 allows the largest possible diameter of the unfolded heat shield keeping the construction reasonably simple. In the closed configuration, represented in Fig.6, a_1, 12 telescopic poles are closed, laying parallel to the cylindrical body of the capsule. The first deployment phase, which is represented in Fig.6, b_1, includes the poles elongation and a first moderate unfolding of the TPS fabrics; in the second phase, which is represented in Fig.6, c_1, the heat shield deployment is completed, the TPS fabrics are stretched and the supporting structure becomes rigid. Once the poles have reached their maximum extension, at the end of phase b_1, a locking mechanism keeps their length fixed, even in the presence of the compression force that develops during the subsequent tensioning phase c_1. Such latter phase c_1 includes the sliding of a mobile structure along the main body of the capsule.

Solution 2:

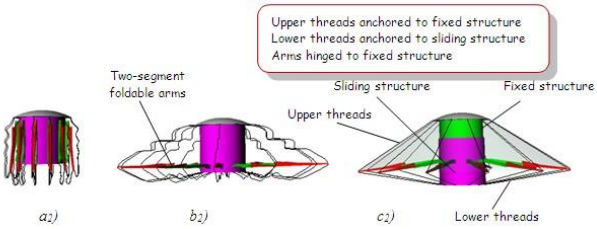

Fig. 7 - Solution #2: a2) folded structure; b2) arm extension phase; c2) tensioning phase.

Solution #2 is conceptually similar to the previous one, with some important differences, however. Here we have arms foldable in two sections, instead of telescopic segments. Another difference with respect to

the previous solution is that the arms are hinged at the fixed structure instead of the sliding structure. Such a choice has been suggested by the need to facilitate unfolding of the flexible part of the TPS. Note that it implies that the sliding structure cannot be a compact one in this case, in order for the arms, which are hinged to the fixed structure, to pass through the sliding structure itself. Also this structure, much like the previous one, is opened in two phases: an arm extension phase, represented in Fig. 7,b_2; and a tensioning phase, illustrated in Fig. 7,c_2.

Solution 3:

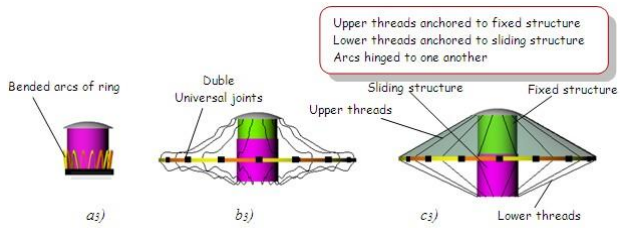

Fig. 8 - Solution #3: a3) folded structure; b3) rim extension phase; c3) tensioning phase.

Solution #3, represented in Figure 8, is radically different. It includes a circular ring which is composed by 12 arcs. Such arcs may be made out of harmonic steel, to work as band springs, and hinged to one another by means of double universal joints. The lower rim of the flexible heat shield is fixed at this ring. The entire ring is tethered to the fixed part of the supporting structure by means of (upper) threads and to the sliding part of the supporting structure by means of other (lower) threads.
When closed, the elastic arcs of the ring are bended, parallel to the cylindrical surface of the sliding structure (see. Fig. 8, a_3). In this status they are like 12 compressed band springs, which need to be constrained somehow, in order to remain in this position.
As soon as these arc springs are released, they open undergoing a composed rotation with respect to one another, which is made possible by the double universal joints. This way the entire ring is formed and the flexible heat shield reaches its maximum diameter (see. Fig. 8, b_3).
In order to give this structure the necessary rigidity, the sliding structure is displaced like in Fig. 8, c_3, the tethers are stretched and the ring becomes stable. Rigidity may be improved if the threads directions are not made to intersect the capsule axis and are oriented much like the radii of a bicycle wheel, instead.

4. PRELIMINARY DESIGN OF THERMAL SHIELD

Most of the materials already tested in the IRENE Phase 0 Study have been considered for a Maxus sounding rocket mission. In particular, also in this case, the heat shield will be composed by two main sections:

- A rigid nose
- A flexible part, to be deployed prior to re-entry.

The flexible part of the heat shield is not requested to function much as a thermal insulator but, rather, mainly as an aero-brake. In facts most thermal insulation means can be placed around the fixed structure (i.e., around the payload/service module, rather than on the deployable part of the thermal shield). In this way the deployable part of the thermal shield can be sufficiently thin and flexible for an easy deployment. Of course such an idea needs to be validated by more detailed analyses and experimental tests that will be carried out in the subsequent phases of the program.
Moreover it will also be necessary to prove that the exposure of the proposed materials to the typical heat fluxes expected during descent would not compromise the tensile strength of the flexible part of the heat shield. This aspect is particularly relevant if a construction will be chosen where the heat shield fabrics are given a structural function as well, as it may be the case if Solution #3 described above is selected.
Only dedicated experiments can give a final answer to these questions because data on resistance of multi-layer fabric structures to hot plasma flows are scarcely available .

5. PRELIMINARY DESIGN OF THE DEPLOYMENT MECHANISM

Two different mechanisms have been considered. At the present stage of definition the main concepts of both the mechanisms can still be thought of as applicable to any of the three solutions outlined above for the TPS supporting structure. Nonetheless, hereafter we will refer to solution #1 of (the one including telescopic poles), to fix our ideas on a well defined case.
Two different mechanisms have been considered to deploy the IRENE aerobrake/heat shield:

Mechanism #1: Including actuator springs and gas dampers
Mechanism #2: Including a gas actuator

Solution 1 exploits harmonic steel compression springs that, once loaded, store the needed mechanical energy to perform the heat shield deployment. In order to avoid abrupt elongation of springs as they are unlocked, a damping system has been devised.

The second deployment mechanism which has been devised in the preliminary study exploits a high-pressure gas working as an actuator spring upon a liquid that performs the displacement. In facts, the gas is initially contained in a bottle, under high pressure (not much differently from the gas reservoirs of compressed air guns).

Fig. 9 - View of the heat-shield structure after the deployment mechanisms have been operated.

6. STRUCTURAL ANALYSIS

A FEM structural analysis has been performed for each of the three identified solutions. The main goal of the activities is the evaluation of the stability and the stress levels in the most critical components of the three solutions. The components so identified, are as follows:

- Poles (sol. #1), arms (Sol. #2) and arcs (Sol. #3);
- Threads
- TPS Fabric layers

In the all three solutions, the components have been modelled using the following FEM element types:

- Beam elements for poles, arms and arcs;
- Truss elements for threads
- Membrane elements for TPS Fabric layers

The capsule structure and the sliding structure have not been modelled in detail, but only their kinematic behaviour has been considered.
Only deployment and loading phases, which are relevant to stress distribution in the structures, have been simulated. Therefore the following load cases have been considered:

- End of the first no-load deployment phase (Only for Solution #1 in which a limited stretching of TPS fabric is caused by the complete extension of the poles)
- End of the tensioning phase, with the sliding structure in the final position
- Maximum pressure applied all over the TPS Fabric (10 kPa applied homogenously)

In all of the three FEM models, four layers of fabric (thickness = 0.39 mm) have been used as the TPS fabrics.
It must be taken into account that the investigated structures are only preliminary solutions, whereby the stress levels calculated in the simulations, especially if localized in small zones, can be strongly reduced after a structural optimization procedure.
A first attempt choice for the admissible stress levels for the structural components, considering also the operating temperature, is the following:

- Titanium structure 400 MPa at 400°C
- Fabric : 40 MPa at 900 °C

According to the previous results, the solution #1 shows a better behavior for the following aspects:

- Better aerodynamic stability, due to smaller cone angle (45° instead of 60°)
- Largest diameter of the deployed structure due to the deploying mechanism kinematics
- Better fabric tension distribution after the deployment phase due to deploying mechanism kinematics
- Lower fabric deflection under the re-entry pressure loads

So, a series of mechanical sensitivity analyses have been performed in order to optimize the fabric behavior and to reduce the deployment work that must be provided by the power system.
As a result of the optimization analysis, though carried out only in static field, the loads needed for the deployment and the stability of the supporting structure and the flexible TPS, are compatible with the admissible limits.

7. PRELIMINARY DEFINITION OF AVIONICS AND INSTRUMENTATION

7.1. PAYLOAD (SENSORS)

The most important parameters to be quantified during the re-entry are the aerothermodynamic loads, represented by the surface pressure distribution and the the surface heat flux.
The measurement of such data is important to validate the proposed design concepts and solutions for the aerothermodynamic performance of the “IRENE family” re-entry vehicles.
The performance of the thermal protection system and umbrella-like structure, shall also be monitored. To this purpose, the pressure and temperature evolution during the reentry flight shall be recorded.
The main assumption is that the payload consists only of the following sensing elements and their respective data acquisition and storage electronics and power supply:

- Thermocouples
- Pressure transducers
- Linear and angular acceleration sensors
- Strain gages
- Telecommunications subsystem
- On Board Camera and Sound Recording
- On Board Data Handling

Figure 10 and 11 show the points selected to place thermocouples and pressure sensors, respectively. The data from such sensors will be used as inputs for the aerothermal and pressure analyses.

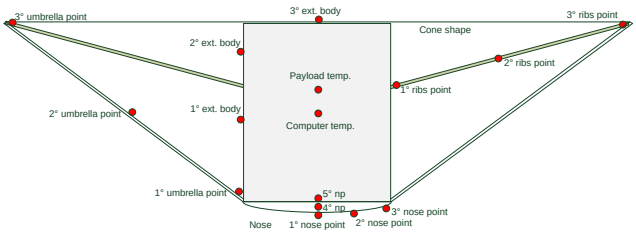

Fig. 10 - Thermocouples location.

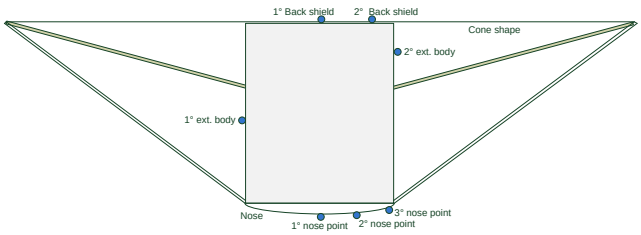

Fig. 11 - Pressure sensors location.

An evaluation of the poles deformation during the capsule's re-entry phase, will be supplied by an appropriate number of miniature **strain gauge sensors** along the poles.
The **GNC** system will be composed of n.1 triaxial accelerometer and n.3 MEMS gyroscopes. Different configurations may be identified during the project evolution.
The localization of the capsule will be based on a satellite emergency locator system **(beacon)**, operating on the 406 MHz band, and a homing signal acquired by the recovery team at 120 MHz.

7.2. OBDH

The **data handling** system could be based on a COTS, preferably flight proven (but not necessarily "space qualified"), able to acquire and store, on a ruggedized memory unit, all the data measured by sensors and transducers.
The figure below shows a possible solution for the avionics architecture, based on an integrated and modular acquisition system.

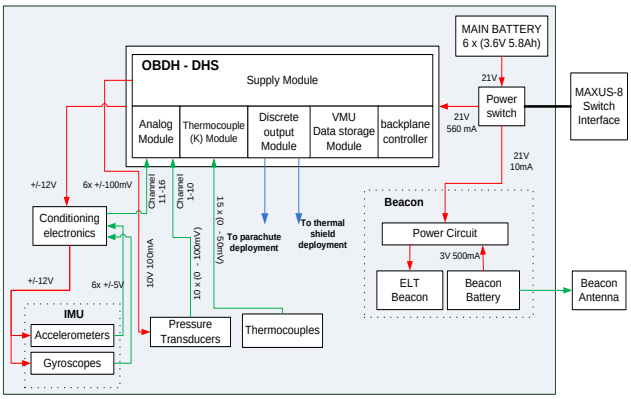

Fig. 12 - Preliminary Electronics Design.

The following equipment list was obtained at the time of the preliminary study's final iteration:

Item	Dimensions (l x h x w) (mm)	Power (W)	Voltage	Operating Temp. (°C)
OBDH	118x101x101	15	18 - 40 Vdc	-40 - +85
Harness	NA	-	-	-
GPS module & antenna	included in OBDH	0,01	12 Vdc	-40 - +85
Beacon & antenna	140x100x100	5	self powered	-20 - +55
Batteries	26x26x51 (x6)	5,8 Ah (x 6)	3,6 V (x 6)	-60 - +85
Temperature sensors (n.16)	NA	NA	NA	-
Pressure sensor	NA	NA	NA	-
Acceleration sensors	25x21x30	0,35 W	12 Vdc	-40 - +85
Camera and Sound recording	42x60x30	1,1 Ah	-	20 - +55
Angular rates	33x19x33	0,3	±10 - ±15	-40 - +85
Strain Gages	12.23x21x30	2,4 mA	12 Vdc	-75 - +205

Table 2: Equipment list.

7.3. Power System

Power subsystem will generate all regulated voltages to supply the other spacecraft subsystems. Phase "0" IRENE study output shows that lithium batteries are the best option as power source for the capsule.

8. MASS BUDGET

A mass budget for the entire IRENE capsule is detailed below.

Item	Mass (Kg.)
Fixed Structure (composites)	2.50
Sliding Structure (composites)	0.45
Two Plates	1.19
Poles (n.12)	1.92
Threads	0.05
Hinges (n.12)	0.12
Deployment Mechanism	2.70
TPS – Fabric	1.40
Nosecone (Silica Foam)	1.20
OBDH	1.26
Harness	0.10
GPS module antenna	0.10
Beacon & antenna	1.00
Batteries	0.31
Temperature sensor (n.16)	0.03
Pressure sensor	0.05
Acceleration sensors	0.05
Camera and Sound recording	0.09
Angular rates	0.03
Strain Gages	0.04
SubTotal	14.54
Margin (25%)	3.64
Total	**18.18**

Table 3: Mass Budget.

9. FUTURE DEVELOPMENTS

A development plan for the MINI IRENE project up to launch, presently planned in 2016, includes the already performed Preliminary Study (referred to as "Bridging Phase", BP) which concludes the Phase 0 studies previously performed on behalf of ASI.
This preliminary study allowed to define the mission base-line, preliminary configurations of the deployable structure and mechanisms, the thermal shield protection of the capsule and the avionics instrumentation.
The second phase of the project MINI IRENE (Phase A and B) shall last one year, from April 2013 to March 2014. It will include the analysis, the design of all the capsule subsystems and the construction of a ground demonstrator of the thermal shield of the capsule.
The third phase of the project (Phase C, D, E) will include the construction, qualification testing of all parts of the capsule and the assembly of the flight demonstrator.

10. APPLICABILITY

The deployable system may be applied to other missions, requiring planetary entry or re-entry.
The technology offers also an interesting potential for aerobraking/aerocapture and for de-orbiting of small satellites or payloads from LEO.

More generally, the convenience of the proposed idea stems from its simplicity and low cost that make the IRENE concept suitable for a number of different launchers and missions. In fact, the low ballistic parameter of the open configuration implies relatively moderate thermo-mechanical loads during the atmospheric descent phase and thus the exploitation of commercial materials is possible, dramatically reducing the TPS costs. Accordingly, the whole variable-geometry heat shield is designed to be completely independent, to separate the low-cost part from the usually more valuable payload. So, the structure displayed in Fig. 9 (including the TPS fabrics not shown in the figure) represents a completely independent, disposable macro-system, which can be regarded as a module. A payload of suitable size can readily be accommodated inside the central, cylindrical payload bay of this module and then retrieved, at the end of the mission. No refurbishment is required and a number of identical modules of this kind can be built, further reducing the cost of each single mission. Presently, the IRENE concept is intended for small-sized payloads and is suitable for launchers ranging from the Maxus to the Vega size but, up to a certain extent, no major obstacles are envisaged that may prevent the development of similar systems with a larger size.
In principle, the IRENE basic idea, is applicable to any mission that has to be set up with a very short notice, as it may be the case for Earth observation campaigns with civil or military purposes.
Finally, one should not miss to note that some of the proposed technical solutions may be valuable not only for variable geometry TPS' but also for other space, deployable systems.

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