Qubesat for Aerothermodynamic Research and Measurement on AblatioN

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ABSTRACT

The preliminary design of the QARMAN re-entry CubeSat developed by the von Karman Institute is presented in this paper from de-orbiting to payload choices. It represents an ideal cost-efficient platform for re-entry flight test and validation of thermal protection system (TPS) materials with a demonstration flight scheduled for June 2015. The CubeSat comprises a standard double-unit platform with sensors for atmospheric research and a functional unit for essential satellite operations. A third unit accommodating an ablative heat shield is added to protect the vehicle against the extreme aerothermal conditions of the re-entry. The challenging aspect of the project lies on the constraining mass and form factor from the CubeSat standard, 3kg and 34x10x10 cm^3. Finally, the preliminary design of the vehicle results in a payload of 400 g collecting data all along the re-entry trajectory including the maximal heat flux conditions.

1. INTRODUCTION

A low cost nano-size re-entry spacecraft will permit huge opportunities for independent research institutions to test materials or subsystems in real flight conditions. A very attractive part of such a mission is to provide an affordable low cost platform to validate ground testing and numerical simulations that are developed to understand the complex problem of the atmospheric re-entry of spacecraft.

The purpose of this paper is to develop a first realistic approach in the preliminary design of such vehicle. This study is related to the QB50 project [1] led by the von Karman Institute for Fluid Dynamics (VKI). The scientific objective of the QB50 project is to study the temporal and spatial variations of a number of key constituents and parameters in the lower thermosphere (90-320 km) with an international network of 50 double CubeSats (Fig. 1), miniaturized satellites weighing 2 kg in a 20x10x10 cm^3 volume. The 50 CubeSats in a circular orbit will be separated by a few hundred kilometres and will carry identical sensors that will perform in-situ, long duration (~3 months) and multi-point measurements. QB50 will also allow to study the re-entry by measuring a number of key parameters during the mission and by comparing predicted and actual CubeSat trajectories as well as orbital lifetimes.

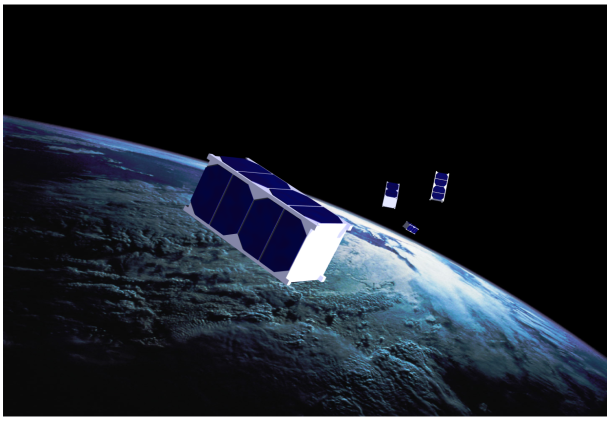

Figure 1: Artistic impression of the QB50 network

2. VKI RE-ENTRY CUBESAT CONCEPT

The difference between the proposed re-entry CubeSat platform and the other CubeSats of the program is the necessity of a deorbiting system and an extra unit for the thermal protection system (TPS) in regard of the reentry constrains.

The reentry conditions with the natural orbital decay will result with a long time in orbit (~3 months) and a high heat flux/heat load on the vehicle. A deorbiting manoeuver is mandatory to ensure the feasibility of the mission. At the difference of a standard vehicle, a drag augmentation system has been found to be the most adequate solution [2] compare to traditional chemical propulsion systems for mass, volume and reliability constrains. In its side, the extra unit mandatory for the thermal protection system (TPS) permits to increase the survivability of the vehicle up to the expected end of life altitude of 50 km. It has to be noted that no other CubeSats of the QB50 program is supposed to survive below 90 km with the two unit configuration. The proposed platform for the vehicle is the 3-Unit (3U) CubeSat standard; such standard platform imposes an external volume that does not exceed 34x10x10 cm^3 in dimensions and 3 kg in mass.

As opposed to conventional re-entry vehicles, the standard form factor is a challenge and the design approach avoids the typical re-entry vehicle shapes optimised for stability and aerothermodynamics constrains. Moreover, it is mandatory to conceder the need of the vehicle's destruction before it reaches the ground thus avoiding any problem of collision with ground assets. The continuing subsystem functionality up to the end of life altitude is the second important constraint in the mission design.

As primary objective, the vehicle should reach the expected end of life altitude (50-70 km) where measurements performed at the critical points of the trajectory can be achieved, which are the maximum heat flux point, the maximum dynamic pressure point and the telecommunication blackout phase. The choice and feasibility of the telecommunication system has also to be considered to be able to transmit the measured data before the vehicle loses its functionality. The most challenging part of the feasibility study is to be able to size an effective TPS that would fit within the external dimensions of a standard 3U CubeSat and that could manage the thermal environment until the targeted altitude, by keeping the payload bay in a suitable temperature (50°C with margins). To be able to size such a TPS, one needs to have a very good knowledge of the aerothermodynamic environment around the vehicle all along the re-entry. For this purpose, an accurate estimation of the trajectory is essential. For the sake of simplicity, the trajectory is chosen to be ballistic (no lift coefficient) to avoid the necessity of a guidance and navigation control. Once the trajectory is propagated, the critical altitude for which data can still be acquired from the sensors on-board has to be determined. The critical altitude is defined as the lowest altitude where the spacecraft continues to collect and transmit flight data before it loses its functionality due to elevated temperature. Finally, the necessary power and mass budgets are calculated to conduct the intended experiments by the sensors on board of the spacecraft. The feasibility analysis as been performed and presented in [3] including the demonstration of the high scientific return potential.

3. CONCEPTUAL DESIGN

3.1. Deorbiting and stability system design

As a constraining subsystem in term of mass and volume, the deorbiting device will drive the whole mission and specifically the heat load and heat flux profiles. These parameters represent the sizing parameters of the TPS design and thus they have to be controlled. For the sake of the scientific objectives of the mission, the maximum heat flux point of the trajectory should be comprised between 1.5 and 2.2 MW/m^2 to represent a test case for Low Earth Orbit (LEO) re-entry missions and specifically return missions from the International Space Station. Considering the boundaries of the maximal heat flux profile, we can only decrease the value of the heat load to lower constrains on the TPS sizing and so reducing the mass of it.

As previously mentioned, the deorbiting system has been chosen to be a drag augmentation device called AeroSDS for Aerodynamic Stabilization and Deorbiting and System as it also stabilise the vehicle during the mission without needs of additional systems. Regarding this option, other parameters leading the optimization of the deorbiting system can be presented, namely the altitudes to deploy and the one to jettison the AeroSDS.

As we want the vehicle to be stabilized as soon as possible, the altitude of deployment has to be as soon as the vehicle is on orbit. For feasibility purpose, an initial commissioning phase (i.e. battery recharge and system checks) will be performed immediately after the beginning of the mission as well as an initial detrembling performed with magneto-torquers.

The two parameters to take attention to within the AeroSDS optimization are then the shape of the deorbiting system and the altitude to jettison it. One could mention the possibility to keep the AeroSDS during the whole mission but the heat fluxes and the dynamic pressures encountered on the AeroSDS below 100 km appears to be too much constraining for the CubeSat form and mass factors. In this condition, as presented in [3], a smaller stability system has been found to be the most adequate solution to deal with the continuum part of the re-entry (below 100 km). It consists on a small geometry downstream the vehicle and fastens to it with a tether, which will add a small drag increment 1.3 meters downstream of the vehicle (Fig. 2), thus stabilizing the vehicle by moving the center of pressure of QARMAN downstream the center of gravity.

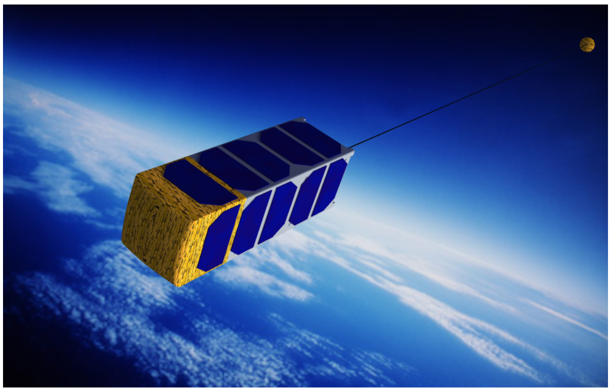

Figure 2: QARMAN vehicle with the stability system for the continuum part of the trajectory.

Fig. 3 presents the methodology employed to optimize the AeroSDS. The idea is to use an optimizer, in our case one using a genetic algorithm [4], to evaluate the different combinations of AeroSDS geometries and altitudes to jettison the deorbiting system through a full trajectory propagation and evaluation with realistic aerodynamic coefficients.

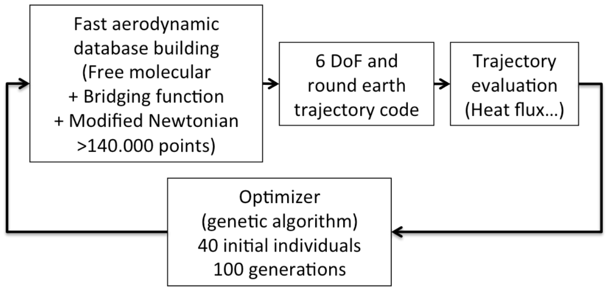

Figure 3: Methodology of the AeroSDS optimization

As the 6 Degrees of Freedom (6 DoF) trajectory code and the evaluation of the heat flux [5] and heat loads remains standardized, the construction of the aerodynamic database for every single AeroSDS configuration remains a time consuming and difficult process with traditional means (i.e. Computational Fluid Dynamic tools). To permit to gain in efficiency and cover the full range of possibilities within the evaluation, a fast aerodynamic database builder has been developed based on engineering methods to cover the full ranges of attitudes and flight conditions for the different vehicles configuration.

The flight regimes can be split in three, starting by the free molecular regime for high altitudes (above ~120km) where the flow is seen as a stream of particles. The continuum regime (below ~80km) where the flow is considered as continues and where applies traditional fluid mechanics relations. Finally, we have a region between the two previously mentioned named transitional regime where models differs largely depending on the geometry of the vehicle.

- The continuum part of the aerodynamic database builder is lead by the Modified Newtonian Theory (MNT) [6] based on integration of the pressure coefficient (Cp) on elementary faces (Eq. 1) that represent the shape of the vehicle.

$$\mathrm{Cp} = Cp_{max} sin^2(\alpha) \qquad (1)$$

  Quantity $\mathrm{Cp}_{\max}$ is the pressure coefficient at the stagnation point of the vehicle derived from the exact shock-wave theory, the Rayleigh Pitot tube formula,

$$\mathrm{Cp}_{\max} = \frac{2}{\gamma M_\infty^2}\left[\left(\frac{(\gamma+1)^2 M_\infty^2}{4\gamma M_\infty^2 - 2(\gamma-1)}\right)^{\gamma/(\gamma-1)}\left(\frac{1-\gamma+2\gamma M_\infty^2}{\gamma+1}\right) - 1\right] \quad (2)$$

  where M_∞ is the Mach number of the free stream and $\gamma = 1.4$.

- The free molecular part of the aerodynamic database relies on the Maxwellian gas-surface interaction kernel [7] (Eq. 3), a local inclination panel method (as for the continuum part).

$$c_p = \frac{\sigma_n}{2}\sqrt{\frac{T_w}{T_\infty}}\left(\frac{e^{-w_x^2 s^2}}{s^2} - \frac{w_x\sqrt{\pi}}{s}[1+erf(-w_x s)]\right) - w_x(2-\sigma_n)\left(\frac{e^{-w_x^2 s^2}}{s\sqrt{\pi}} - w_x[1+erf(-w_x s)]\right) - \frac{2-\sigma_n}{2s^2}[1+erf(-w_x s)] \qquad (3)$$

  where w_x is the direction cosine between the free stream velocity vector and the surface normal, s the speed ratio calculated as $s = \frac{V_\infty}{\sqrt{2RT_\infty}}$, σ_n the accommodating coefficient kept equal to 1, V_∞ and T_∞ the velocity and temperature of the free stream, T_w the temperature at the wall and R the specific gas constant depending on the altitude.

- The transitional regime needs a bridging function between the continuum and the free molecular parts of the trajectory. As it is highly dependent on the vehicle shape, the 3U CubeSat standard with a ratio of length on width equal to 3 leads us to choose the lifting body configuration [8] rather than the ballistic one for the aerodynamic database even if the trajectory profile is different. The following equations describe this bridging function.

$$C_N = C_{N_C} + \left(C_{N_F} - C_{N_C}\right)\overline{C_N} \qquad (4)$$
$$C_A = C_{A_C} + \left(C_{A_F} - C_{A_C}\right)\overline{C_A} \qquad (5)$$

  where C_N is the normal aerodynamic force coefficient, C_A is the axial aerodynamic force coefficient, the subscripts "*F*" and "*C*" denote the free molecular flow and continuum regions, respectively. In addition,

$$\overline{C_N} = exp[-0.29981(1.3849 - log_{10}Kn)^{1.7128}] \ (6)$$

  if $log_{10}Kn < 1.3849$, otherwise $\overline{C_N}$=1.0.

  and,

$$\overline{C_A} = exp[-0.2262(1.2042 - log_{10}Kn)^{1.8410}] \ (6)$$

  if $log_{10}Kn < 1.2042$, otherwise $\overline{C_A}$=1.0.

  with Kn the Knudsen number defined as the ratio between the mean free path (the mean distance between two particles at the considered altitude) and the characteristic length of the vehicle (here 34 cm).

The shape of the AeroSDS has been chosen to be pyramidal as presented in Fig. 4. The parameterization of the AeroSDS has been done with the angles for the inclination of the panel (from 90 to 135° from the CubeSat side panel) and the total surface of the deorbiting system (from 0.5 to 2 m^2).

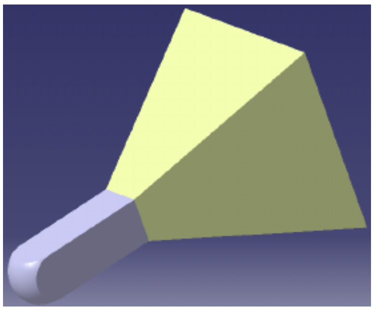

Figure 4: Sketch of the AeroSDS (pyramidal part) on the CubeSat

The result of the optimization is presented in Fig. 5 with the nominal design and the optima chosen.

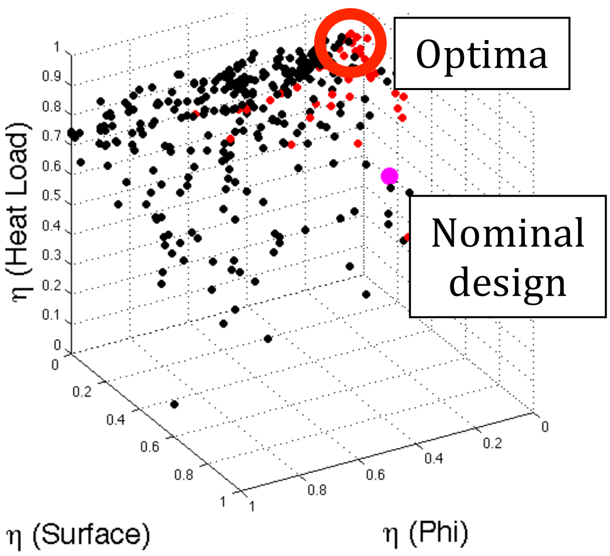

Figure 5: 3d plot representing the result of the optimisation with the nominal and the chosen optima

The Fig. 4 is plotted according to evaluated parameter that we desire to reduce. The Surface and the Phi (inclination of the AeroSDS's panels) are getting close to the actual minimum of their range but a physical limit avoid the heat load evaluated parameter to get under a certain limit due to the constrains on the heat flux range.

The final optima chose (~0.55m^2, 110° with a jettison at ~120km of altitude) permits to gain in mass and volume and maintain the stability of the vehicle all along the trajectory with the combination of the AeroSDS and the stability system for the continuum part of the trajectory.

3.2. Trajectory

For deeper analysis, the 6 DoF trajectory code [9] was used once again for the analysis. In this case, the analysis was made by taking in account the uncertainty on the different parameters of the trajectory propagation. According to [10], the uncertainties on the atmospherically models and on the aerodynamic coefficients have been set to the range ±10%. The result of a Monte Carlo simulation with the trajectory code and whose inputs have permitted to get the heat flux profile with its dispersion. As the uncertainties will affect the time of the re-entry, Fig. 6 shows the time axis as normalized.

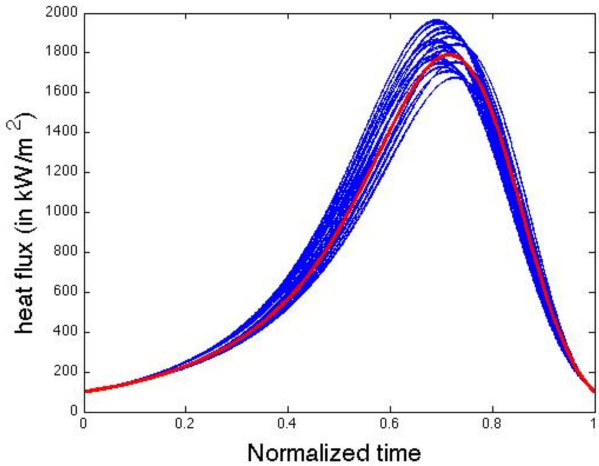

Figure 6: Heat flux profile with dispersion versus normalized time

Fig. 6 will permits in the next part to conceder the dispersion of the heat flux as a margin leading criteria for the TPS sizing.

3.4. Thermal analysis

The critical altitude is defined as the altitude where the sub-systems of the re-entry CubeSat demonstrator would stop functioning due to elevated temperature inside the payload bay. As the vehicle should not reach the ground, this critical altitude must be reached as soon as we don't need to perform any more measurements. This boundary has been chosen to be equal to 50 km as it is just after the peak heating part of the trajectory and so the most interesting and challenging part of the mission.

The TPS material has been chosen to be the ablative Cork P50 material [11] for the heat shield part in order to keep the conceptual design simple and realistic as wildly available and affordable. Future studies or application may conceder different TPS material and specifically for industrial TPS testing, as the need will be to test different heat shields.

Reference [11] permits to withdraw a design method for the TPS sizing. The idea is to directly link the local cold wall heat flux (heat flux measured at a wall kept at 300K) and the recession rate of the virgin material used as heat shield. The recession rate is the velocity at which the TPS material will pyrolyse and so loosing insulation thickness. The test of reproducibility have been performed in von Karman Institute on the cork P50 samples among a range of heat fluxes, stagnation pressure and local radius and the suitability of the theory has been shown as presented in Fig. 7.

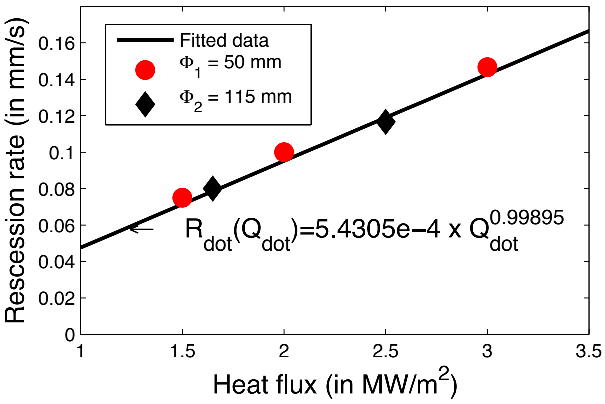

Figure 7: Recession rate in function of cold wall heat flux for the cork P50.

For each point of the heat shield's surface, a one dimensional thermal response of the Cork P50 is calculated and the thickness needed to have the payload bay's temperature equal to 340 K. The next step of the program permits to integrate all the thicknesses calculated to form the definitive thickness mapping of the heat shield. Finally the side parts of the heat shield are smoothed to acquire margins in structural and thermal constrains. The margin on the side of the heat shield doesn't represent an obstacle on the destruction of Qarman neither in the internal volume as one can observe in the following Fig. 8. Effectively, to ensure the destruction of the vehicle several method are possible and the chosen one induce to size the stagnation line thickness in a way that at the desire altitude (50 km) the rest of the heat shield will be too thin to resist to the aerothermodynamic loads. Other systems are possible but the idea is to limit the margins on the stagnation area.

The result presented in Fig. 8 permits to have a heat shield fitting within its dedicated unit. With a mass of 360 grams and internal volume available, the presented design of the TPS permits to gain more margins for the payload mass/volume budget and more confidence for the feasibility of the mission.

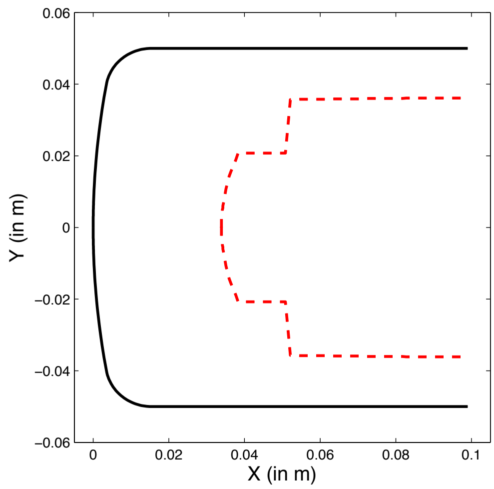

Figure 8: Cut of the heat shield (solid line: the external surface; dash line: inner surface)

The rest of the vehicle will sustain much lower constrains (maximum 300 kW/m^2 at the edges of the CubeSat). In this conditions, the standard structure will be maintain for the vehicle and additional layer of Nextel 312 [12] are included adding 217 grams to the total mass budget. Additional data on the sizing are available on [3].

3.5. Scientific instrumentation

QARMAN will carry on board a set of payloads, to investigate the challenging physics of the re-entry flight. The measurement techniques have to be chosen carefully due to the strict platform constraints and complex physical phenomena. The sensors will be developed in house or bought from already existing technologies.

The final selection of the aerothermodynamic payloads will be made after further scientific investigation. The Tab. 1 shows a possible set of sensors and their mass/power/data budgets including margins. The analysis shows that the aerothermodynamic payloads are suitable with the allowed budgets and that the mission is feasible. The data acquisition unit will be developed by VKI and is currently estimated as 1 PCB with 200 g and 0.2U volume.

Table 1: Possible set of sensors

Challenge	**Parameter to measure**	**Sensor**	**Mass [kg]**	**Energy /Sensor [mW h]**	**Data Size /Meas. [bit]**
TPS Ablation	Recession	2 x Recession Sensor	0.004	1.67	10
TPS Efficiency	Temperature Distribution	8 x TC	0.031	1.67	14

Stability	Pressure	4 x Pressure Sensor	0.060	840	10
Rarified Flow	Low Pressure / Vacuum	1 x Vacuum Sensor	0.011	756 756	10
Shear Force, Transition	Skin Friction	4 x Preston Tube	0.120	2520 16.67	10
Off-Stagnation Temperature Evolution	Temperature	10 x TC	0.021	16.8 1.67	14
ATD Environmen t	Species	1 x Spectrometer	0.084	6250	28
Total			**0.391**	6250	**338**

3.6. Data transmission

During the re-entry, a plasma sheet is formed in front of the vehicle and at a given time, the electronic density can become high enough to block any radio signal to pass through it, avoiding communication with the ground stations (i.e., the telecommunication blackout). As mentioned in the introduction, one of the most interesting and challenging parts of the mission in terms of the scientific return is the range of altitude 50-70 km; which is within the telecommunication blackout. The re-entry CubeSat demonstrator is not designed to survive to lower altitudes and the only solution is downloading the data during the actual re-entry flight.

To overcome the communication blackout issue, a non-conventional methodology should be used. The most feasible solution is to transmit data through the area with lower electronic density that would be located downstream of the spacecraft (through its wake). This antenna location helps the signals to be recovered in space and so, the utilization of other satellite(s) is essential as the space shuttle did with the TDRS (Tracking and Data Relay Satellite System) during the re-entry to avoid radio blackout. Fig. 9 shows the typical relay communication system proposed for the present case.

From the inherent nature of the vehicle's trajectory, no pinpointed re-entry is expected. A network/constellation of communication satellites has to be considered to be able to recover the signal all along the re-entry and wherever/whenever it happened. Considering the main purpose of the re-entry vehicle, it is beneficial to be able to transmit as much data as possible, so the link budget is the key parameter for reducing the transmitting distance and increasing the antenna gain. The Iridium constellation (66 satellites at 780 km of altitude) has been chosen to be able to have a better budget link and to have increased scientific return than a Geostationary Earth Orbit satellite/constellation option.

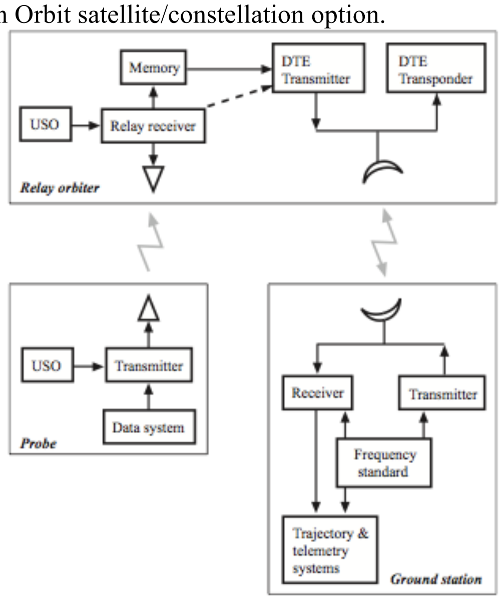

Figure 9: Relay communication system, from [8]

A coverage analysis has shown that the re-entry vehicle is within the communication range of at least 4 Iridium satellites during the entire trajectory. No Doppler shift problem during the data transition is expected considering the satellite speed (7.5 km/s) and close orbit inclination (less than 20 degrees of orbital inclination difference).

5. CONCLUSION AND PERSPECTIVES

A critical review of the different subsystems and key points of the Re-Entry CubeSat demonstrator mission is presented in this paper. Resulting from the system analysis the power budget is presented in Tab. 2 for the reentry part of the mission; the most power constraining one as it should use only its batteries. In its side, the mass and volume budget are shown in Tab. 3. The preliminary design yields to the feasibility of the proposed mission with respect to its constraints. Further studies, less conservative, will allow a heat shield's mass reduction leading to more payload mass available.

Table 2: Power budget for the reentry phase

System		DC [%]	[W]
EPS		100	0.20
OBC		100	0.23
ADCS	Gyroscope	0.66	0.66
	Accelerometer	0.00	0.00
	Magnetorquer	0.00	2.43
	GPS	0.00	0.10
Comm.	UHF/VHF	0.00	0.00

	IRIDIUM	1.25	0.00
Payload Sensors		100	6.28
Total			**8.62**

Table 3: Mass and Volume budget

Subsystem	**Mass [g], [%]**		**Volume [10cm], [%]**	
Heat Shield				
Front surface	360	20	0.63	25
Side-Panels	217	20	n/a	0
Functional Unit				
Structure (2U)	468	20	n/a	
OBC	161	10	0.17	10
EPS + Batteries	248	10	0.33	25
Solar Panels	336	5	n/a	
Communication	263	10	0.46	10
Payloads				
Acquisition PCB	240	20	0.25	25
Sensors	469	20	0.25	25
AeroSDS	300	20	0.5	25
Total	**3062**		**2.59**	

One of the aims of the demonstrator is to show the feasibility of a standard platform for in-orbit technology demonstration and reentry experiments. For this purpose, the utilization of standard subsystems is essential to reduce the development costs of similar re-entry testbed in the industrial and scientific sector.

The potential of the proposed preliminary design of QARMAN (Fig. 10) is a tremendous opportunity for the atmospheric re-entry and material sciences. This platform should be considered as a low cost platform for test and validation of sub-systems and concepts.

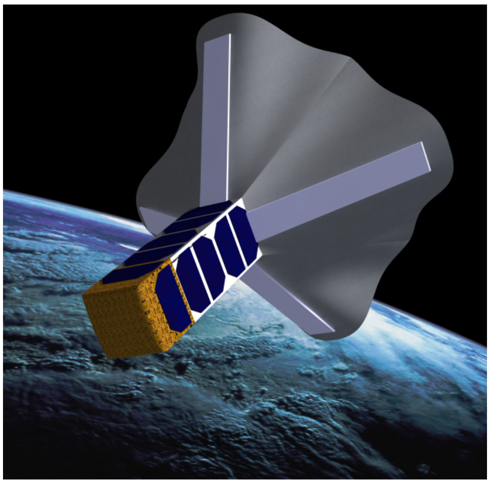

Figure 10: Artistic view of QARMAN with the AeroSDS

6. ACKNOWLEDGMENT

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