# Aerothermodynamic and Feasibility Study of a Deployable Aerobraking Re-Entry Capsule

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**Abstract:** A new small recoverable re-entry capsule with deployable heat shield is analyzed. The possible utilization of the capsule is for safe Earth return of science payloads or data from low Earth orbit at an inexpensive cost, taking advantage of its deployable structure to perform an aerobraking re-entry mission, with relatively low heat and mechanical loads. The system concept for the heat shield is based on umbrella-like frameworks and existing ceramic fabrics. An aerothermodynamic analysis is developed to show that the peak heat flux, for a capsule with a ballistic coefficient lower than 10 kg/m<sup>2</sup>, is in the range 250-350 kW/m<sup>2</sup> and the corresponding surface temperatures are sustainable by off-the-shelf ceramic materials.

The article summarizes the main concept and the numerical predictions concerning the re-entry trajectories, the aero-thermal loads, the possibility to control the re-entry trajectory varying the capsule cross-sectional surface. Attention is focused on the effect of the material surface catalyticity on the surface heat flux, in presence of reacting gas mixtures formed behind the strong shock wave in front of the capsule.

**Keywords:** Atmospheric re-entry, hypersonic flows, aerodynamic de-orbit, materials surface catalyticity.

# 1 Introduction

In recent years there has been a wider interest for small space platforms (microsatellites and nano-satellites), mainly used in traditional space fields, such as remote sensing for Earth's environmental protection, prevention of natural disasters and homeland security. Other applications include scientific experiments or technology, space exploration, observation of the universe, astrophysics, biology or physical sciences in microgravity.

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Reducing size, mass and power implies a significant reduction of the cost of the mission; on the other hand, as a consequence of the limited size and power, these systems require more sophisticated solutions to achieve ambitious scientific and technological goals.

A particularly interesting aspect related to the proliferation of micro and nanosatellites is the parallel investigation of small recoverable platforms from low orbit. The development of relatively small capsules able to safely return small payloads to Earth, can provide "physical" recovery of experimental samples or scientific data, as complementary capability to the one offered real-time by links to ground stations. Among the most critical aspects related to recoverable space capsules one must mention the aerodynamic heating, which requires use of complex thermal protection systems, as well as the development of appropriate guidance and navigation systems, able to ensure that the capsule follows the proper atmospheric re-entry flight corridor.

Much effort is currently dedicated in the aerospace community to study inflatable or deployable systems for atmospheric re-entry. This kind of capsules, in fact, can be easily accommodated in launch vehicles in folded configuration and, when deployed, exhibits a low ballistic parameter (i.e. the ratio between the capsule mass and its surface, times the drag coefficient) by increasing its cross sectional area, its drag coefficient, or both. The lower ballistic parameter allows to obtain larger decelerations in the upper part of the Earth atmosphere, offering as advantage the reduction of the aerothermal and mechanical peak loads and, consequently, a much higher reliability in the re-entry phase. Examples of inflatable systems recently proposed and already tested are the Inflatable Re-entry and Descent Technology (IRDT) [Wilde et al. (2001)] and the Inflatable Re-entry Vehicle Experiment (IRVE) [Lindell et al. (2006)]. Different concepts include re-entry systems based on mechanically deployable heat shields. In 1990 a deployable capsule was developed using an umbrella-like heat-shield, made of silicon fabrics and called parashield [Akin (1990)]; a similar concept, Bremsat, was studied in 1996 at the University of Bremen [Wiegand and Königsmann (1996)].

Opening the parashield, the cylindrical container (in which the payload and the subsystems necessary to accomplish the mission can be stowed) is completely shadowed by the umbrella cap. Furthermore, the sphere-cone configuration is intrinsically stable at all flight regimes.

The presence of thermal protection systems, based on flexible and deployable structures, able to achieve a very low ballistic coefficient for the satellite, offers several additional advantages, ensuring Earth return within an acceptable timeframe, appreciable reduction of the levels of deceleration and of thermal and mechanical loads, structural simplification, controlled de-orbit by means of aerodynamic deceleration (aerobraking), avoiding the use of traditional propulsion systems. On the other hand, the use of such an innovative method for de-orbiting requires sophisticated control algorithms.

The objective of the present work is to analyze a new concept of deployable recoverable re-entry system for safe return of nanosatellite payloads or data from low Earth orbit at low cost. In particular, the attention is focused on the analysis of the re-entry phase of the nanosatellite re-entry system with deployable umbrella-like heat shield.

The system could be launched in orbit with the heat shield in folded configuration and used to execute on-orbit experiments (e.g. earth observation, microgravity, technology, life science, etc.) for periods of relatively short duration (up to 1 month); after the scientific mission, the heat shield can be deployed to increase the sectional area and therefore the aerodynamic drag, to execute a controlled de-orbit and re-entry in reasonable time, so that payload and experimentation data can be safely recovered after a small duration atmospheric re-entry (up to 1 hour).

The work is intended as a preliminary feasibility analysis of the concept, concerning the most critical aspects from several points of view, like re-entry trajectories, aerothermal and mechanical loads on the materials, landing point dispersion, guidance, control and telecommunication issues (plasma frequency and blackout aspects).

The paper is organized as follows. After the description of the capsule concept in paragraph 2, the numerical models implemented to study the capsule orbital decay and the re-entry trajectory, along with the numerical model implemented for CFD calculations, will be illustrated in paragraph 3. In the subsequent paragraphs 4 and 5 results of calculations concerning the evaluation of the re-entry trajectory, the aerothermal and mechanical loads are presented. Paragraph 6 deals with the analysis of the plasma sheet due to the air ionization in the shock layer around the re-entry capsule, which is particularly important for its relation to the radio communications blackout problem. A discussion is dedicated to the effects of the material surface catalytic activity. In particular, computational results give an indication of the effect of the contribution to the surface heat flux related to the chemical energy released by the gas when atomic species recombine at the body surface. The landing point dispersion and the corresponding analysis to control the re-entry trajectory in its first leg, properly modulating the capsule cross sectional surface area, is carried out in paragraph 7. Final comments and conclusions are presented in paragraph 8.

### 2 System configuration

The system launch mass is intended to be only few tens of kilograms so that the entire platform can be launched as a secondary payload of a launch vehicle or with a smaller air-launched rocket. At an altitude of approximately 300 km, the onboard instruments can be operated for approximately one month. After completing the experiments, the nanosatellite performs an aerodynamic de-orbit maneuver, via ground command, taking advantage of the deployable structure.

Due to the high reduction of the ballistic coefficient  $(m/C_D \cdot S)$  and the consequent reduction of the aerothermal and mechanical loads, a simple concept for the heat shield can be considered, using mechanical framework (umbrella-like) and existing ceramic fabric. Therefore, after aerocapture and separation from the satellite bus, the capsule safely re-enters through the atmosphere and, after landing, the payload is delivered for post flight inspection and experimentation. The small mass/surface area ratio results in terminal velocities of the order of 10 m/sec, requiring only terminal decelerators (e.g. small parachutes) or shock absorbers to mitigate the landing impact.

The re-entry phase, in particular, is intended to be controlled by adjusting the parashield bevel and, therefore, its surface, in such a way to cope with the differences between the re-entry trajectory detected by on board instrumentation (e.g. GPS, beacons, accelerometers) and the nominal one.

The concept for the re-entry capsule (see Figure 1) mainly consists of a central cylindrical structure containing all the subsystems necessary to perform the onorbit mission and the re-entry phase, umbrella-like frameworks, existing ceramic fabrics (e.g. Nextel) for the conical umbrella and off-the-shelf ceramic materials (e.g. silica, allumina or zirconia) for the nose.

The main necessary subsystems may include parachute assembly, beacon, main computer and electronics, sensors, AOCS, IMU and GPS instruments, batteries and deployment mechanisms.

Typical values regarding mass and reference surface area of the satellite modules are reported in Table 1, while Figure 1 shows possible vehicle configurations in the different phases of the mission.

Component	Mass [kg]	Reference surface [m <sup>2</sup> ]
Re-entry capsule	5	1
Satellite bus	15	0.0625

Table 1: Mass and reference surface of the main vehicle modules



Figure 1: Capsule concept in launch (a), de-orbit (b) and re-entry (c) configuration

#### 3 Models

#### 3.1 Dynamic models

The satellite orbital decay from an initial altitude of 300 km has been evaluated comparing results obtained implementing two different models: the direct solution of the dynamic equations of motion [Anderson (1989)] and the reduction in orbital period due to atmospheric drag [Tobiska et al. (1987)].

In both cases the density has been assumed to vary with respect to the satellite altitude (above 175 km) according to the exponential law 3.1. In Equation 3.1, *SH* is a scale height, strongly influenced by solar activity, i.e. by the solar radio flux, and  $\rho_0$  is the reference density at an altitude of 175 km.

$$\rho = \rho_0 e^{-\frac{h - 175}{SH}} \tag{3.1}$$

Equations 3.2 characterize the dynamic of the satellite and have been numerically solved taking advantage of the Euler's method (being *t* the time, *V* the velocity, *h* the altitude,  $\gamma$  the flight path angle,  $\beta$  the ballistic parameter, *g* the gravity acceleration, *r* the curvature radius of the trajectory,  $\psi$  the azimuth angle,  $\lambda$  the latitude and  $\Lambda$  the longitude).

$$\begin{cases} \frac{dV}{dt} = -\frac{\rho V^2}{2} \cdot \frac{C_D S}{m} - g \sin \gamma = -\frac{\rho V^2}{2\beta} - g \sin \gamma \\ V \frac{d\gamma}{dt} = \left(\frac{V^2}{r} - g\right) \cos \gamma \\ V \frac{d\psi}{dt} = -\frac{V^2}{r} \cos \gamma \cos \psi \tan \lambda \\ \frac{dh}{dt} = V \sin \gamma \\ \frac{d\lambda}{dt} = \frac{V}{r} \cos \gamma \sin \psi \\ \frac{d\lambda}{dt} = \frac{V}{r} \cdot \frac{\cos \gamma \cos \psi}{\cos \lambda} \end{cases}$$
(3.2)

The second model, based on Equations 3.3, on the contrary, calculates the reduc-

tion of orbital period *P* due to the atmospheric drag and the consequent reduction of altitude (where  $\rho$  is the air density and  $\mu_{\oplus}$  the Earth's standard gravitational parameter).

$$\begin{cases} \frac{dP}{dt} = -3\pi \frac{\rho r}{\beta} \\ r = \sqrt[3]{\frac{P^2 \mu_{\oplus}}{4\pi^2}} \end{cases}$$
(3.3)

In order to cross-check and validate the above models, the satellite lifetime (i.e. the time necessary to decay from an orbit 300 km altitude to 180 km altitude, considering the umbrella-like heat shield in folded configuration) has been calculated for different solar activity and reported in Table 2. In particular, calculations have been carried out considering a reference surface of 0.0625 m<sup>2</sup>, for the Minimum Solar Activity (MiSA), the Average Solar Activity (ASA) and the Maximum Solar Activity (MaSA).

Table 2: Satellite lifetime evaluation

	Satellite lifetime [days]		
	MiSA	ASA	MaSA
Direct equation solution	74.8	40.8	27.5
Period reduction model	77.8	39.3	26.8

Table 2 shows that results obtained using the "Period reduction model", despite their higher level of approximation, do not differ more than 5% from the "Direct equation solution". The satellite lifetime strongly depends on the solar activity but, considering the folded configuration, is in any case in the range 1-2 months.

# 3.2 Computational Fluid Dynamic (CFD) model

In order to evaluate the electrons number density and correspondingly the characteristic plasma frequency (which is related to the radio communication blackout [Savino et al. (2010)]), CFD simulations for the flow field around the capsule have been carried out. The numerical model is based on the solution of the Navier-Stokes equations for a chemically reacting mixture composed of eleven species (N<sub>2</sub>, N, O<sub>2</sub>, O, NO, N<sub>2</sub><sup>+</sup>, N<sup>+</sup>, O<sub>2</sub><sup>+</sup>, O<sup>+</sup>, NO<sup>+</sup>, e<sup>-</sup>) in chemical non equilibrium. In particular, the parameters regarding the kinetic ionization reactions have been set according to Evans' model, which has been validated in previous works also cross checking experimental results (see, for example, [Savino et al. (2010)]).

The flow equations have been solved in FLUENT adopting for convective numerical fluxes its AUSM+ scheme, customized with appropriate functions to implement the previously mentioned thermo-chemical models. The grid convergence analysis and the justification of the model for the rarefied regime under study is presented in Section 6.

# 4 Re-entry trajectories

As discussed in Sections 1 and 2, after completing the experiments, in order to allow fast payload recovery, the nanosatellite has to perform a controlled aerodynamic de-orbit maneuver, taking advantage of the deployable structure. Increasing the cross section, the ballistic parameter of the satellite decreases, so that the deorbit time to the atmospheric entry interface is strongly reduced.

Table 3 summarizes the parameters which characterize the re-entry phase between 270 and 100 km. The drag coefficient has been assumed equal to 2 in this trajectory leg (because of the high rarefaction level), according to the results obtained from a Direct Monte Carlo aerodynamic simulation carried out at 250 km altitude.

	Mass [kg]	Reference Surface [m <sup>2</sup> ]	Drag Coefficient	$\beta$ [kg/m <sup>2</sup> ]
Satellite	20	1	2	10

Table 3: Satellite ballistic parameter

Figure 2 shows the comparison of the results obtained by two different models discussed in the previous section. In both cases calculations have been carried out for an Average Solar Activity. The results agreement appears satisfactory.

At an altitude of 100 km, the capsule and the satellite bus detach; as shown in Table 4 the two modules are characterized by very different values of the ballistic parameter (mainly due to the different cross section).

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	Mass [kg]	Reference Surface [m <sup>2</sup> ]	Drag Coefficient	$\beta$ [kg/m <sup>2</sup> ]
Capsule	5	1	1	5
Satellite bus	15	0.0625	1	240

Table 4: Capsule and satellite bus ballistic parameter

Figures from 3 to 5 show the final trajectories legs, obtained integrating the dynamic equations of motion, both for the capsule and the satellite bus, considering again an Average Solar Activity and a re-enter from an equatorial orbit. The drag coefficient has been assumed to be equal to 1, according to CFD simulations (see Section 6).



Figure 2: Satellite orbital decay (considering the umbrella-like heat shield deployed) for an average solar activity

The analysis of the Figure 4 shows that the relatively low ballistic parameter allows the capsule to re-enter in a longer time, gradually dissipating its initial kinetic energy, because most of the deceleration occurs at higher altitudes (i.e. in the most rarefied part of the atmosphere). On the other hand, the satellite bus re-enter the atmosphere with a steep trajectory and is destroyed by the relatively high aerothermal and mechanical loads (see next Section). Figure 6 finally shows the longitude variation along the re-entry trajectory, when the initial longitude (at the entry interface, corresponding to an altitude of 100 km) is 36°E.

The initial latitude allows to monitor the final leg of the re-entry trajectory from the Malindi Ground Station, as shown by Figure 7 and Figure 8. These figures also show that the ground track for the satellite bus is much longer than the one corresponding to the capsule, in spite of its less duration.

#### 5 Aerothermal and mechanical loads

The stagnation point heat flux, along the re-entry trajectory, has been estimated taking advantage of Equation 5.1, also known as Tauber's engineering formula [Tauber (1989)] (being  $R_c$  the nose curvature radius, in this case equal to 20 cm), and plotted



Figure 3: Flight-path angle variation along the re-entry trajectories



Figure 4: Altitude Vs time along the re-entry trajectories



Figure 5: Velocity variation along the re-entry trajectories



Figure 6: Longitude variation along the re-entry trajectories



Figure 7: Capsule ground track along the re-entry trajectory



Figure 8: Satellite bus ground track along the re-entry trajectory

in Figure 9.

$$q = 1.83 \cdot 10^{-4} \sqrt{\frac{\rho_{\infty}}{R_c}} \cdot V_{\infty}^3 \tag{5.1}$$

Figure 9 clearly shows that the maximum stagnation point heat flux, for the capsule geometry and mass parameters summarized in Table 1, is in the order of  $250 \text{ kW/m}^2$  (which can be tolerated by off-the-shelf materials able to withstand temperatures in the order of 1550 K), while for the satellite bus it is in the order of 1800 kW/m<sup>2</sup>, probably leading to its destruction within the atmosphere (due to the absence of an adequate thermal protection system). Increasing the ballistic coefficient, the peak heat flux altitude decreases and the corresponding convective heat transfer rate increases. Computations show that, for a ballistic coefficient in the order of 10 kg/m<sup>2</sup>, the maximum heat flux is about 350 kW/m<sup>2</sup>.

Figure 10 shows that the maximum deceleration, according to the theory, changes very slightly with the ballistic parameter.

Finally, Figure 11 shows that the dynamic pressure in the re-entry phase for the



Figure 9: Stagnation point heat flux variation along the re-entry trajectories



Figure 10: Acceleration variation along the re-entry trajectories



Figure 11: Dynamic pressure variation along the re-entry trajectories

capsule is much lower (less than 1 kPa) than for the massive satellite bus.

#### 6 Fluid dynamic simulations

#### 6.1 Grid convergence and model validation

As discussed in Section 3, CFD simulations have been carried out taking advantage of the FLUENT software [Fluent Inc. (2006)], for the re-entry conditions summarized in Table 5.

h [km]	Mach number	V [km/s]	H [MJ/kg]	p [Pa]	T [K]
84.0	21.1	5.86	17.4	0.525	191

Table 5: Maximum stagnation point heat flux conditions

A grid convergence analysis has been performed in order to ensure the solution independence from the mesh fineness. In Table 6 the values of three parameters (i.e. drag coefficient, stagnation point pressure and stagnation point heat flux), calculated using a coarse and a fine mesh, composed of about 1500 and 7000 quadrilateral cells, respectively, have been compared. Table 6 and Figure 12 show that the main results are not significantly different.

	e	5	
	Coarse mesh	Fine mesh	Error
	$(\approx 1500 \text{ cells})$	$(\approx 7000 \text{ cells})$	
Drag Coefficient	1.259	1.261	0.159%
Stagnation point pressure [Pa]	138.0	139.0	0.725%
Average surface heat flux [kW/m <sup>2</sup> ]	74.93	76.52	2.12%

Table 6: Grid convergence analysis



Figure 12: Contour of static pressure for the coarse mesh (left) and comparison between coarse and fine meshes in the stagnation region (right)

It is well known that at relatively high altitudes (in the order of 80 km) air rarefaction may become not negligible. In order to obtain confidence on the validity of the Navier-Stokes equations the so called rarefaction parameter [Bird (1970)] has been evaluated according to equation 6.1, where  $\mu$  is the fluid dynamic viscosity, pthe pressure and D the capsule diameter.

$$\xi = \frac{\mu V}{D p} \tag{6.1}$$

For the simulated conditions corresponding to Table 5,  $\xi$  resulted to be less than  $10^{-3}$  in the entropy layer around the capsule, so the flow can be considered continuous and the Navier-Stokes equations applied. Rarefaction effects have been also taken into account considering the drag coefficient variation with the Knudsen number (i.e. the rarefaction level), according to Equation 6.2, which provides a bridging relation between the continuum (c) and the free molecular flow (f) regimes.

$$C_D = C_{D,c} + (C_{D,f} - C_{D,c}) \frac{Kn}{Kn + 0.1}$$
(6.2)

Figure 13 shows the variation of the drag coefficient in the range 60-280 km. The results from Equation 6.2 are compared to numerical results obtained with CFD (at 60 and 90 km of altitude) and Direct Simulation Monte Carlo (at 250 km). Finally, Figure 14 shows the trajectory computations based on different assumptions on the drag coefficient. The solid line correspond to a piecewise constant drag coefficient ( $C_D=2$  in the range 300-100 km,  $C_D=1$  below 100 km), while the dashed line corresponds to the continuous drag coefficient shown in Figure 13. The results are not significantly different, in terms of landing point shift.



Figure 13: Comparison between the drag coefficients obtained from CFD (60 and 90 km) and DSMC method (250 km) and its variation, as a function of the Knudsen number, according to the bridging function

### 6.2 Chemical and ionization effects

For the conditions shown in Table 5, the maximum concentrations of the dissociated species are reported in Table 7. Due to the relatively low density at these altitudes, the chemical dissociation reactions are frozen, so that the maximum molar fraction of atomic oxygen is less than 2%; therefore, for the considered flight conditions, phenomena related to chemical non equilibrium and surface catalytic activity are negligible.

The computed electron densities are summarized in Table 8: in particular, the number of electrons per cubic centimeter, labeled in the present work as N, based on



Figure 14: Comparison between the re-entry trajectory obtained assuming a piecewise constant drag coefficient and the one relative to a variable drag coefficient (according to the bridging law 6.2)

Table 7: Maximum molar fractions for dissociated species around the stagnation point

	0	Ν	NO
Maximum molar fraction	1.70e-2	1.37e-3	1.50e-3

Equation 6.3, allowed to calculate the plasma frequency  $f_P$ .

$$f_P = \frac{56380}{2\pi} \sqrt{N}$$
(6.3)

Table 8: Maximum electron molar and volume fraction and for the plasma frequency in conditions listed in Table 5

Electrons	Electron volume	Plasma
molar fraction	fraction [cm <sup>-3</sup> ]	frequency [GHz]
2.07e-3	3.3e12	16.3

Although, as expected, the maximum electrons molar fraction is much lower than 1%, this value leads to a maximum plasma frequency of about 16.3 GHz, occurring

in the stagnation point region. Although the plasma frequency decreases along the capsule up to 5 GHz, moving far away from the stagnation point to the rear of the capsule, the present results suggest that a blackout problem arises during the atmospheric re-entry of the capsule due to the thick plasma established in the viscous shock layer. The study of the wake region has not been addressed in the present work and it should be considered to investigate if the antennas could be properly placed at the capsule base, i.e. in the afterbody region, not directly exposed to the air stream.

Nonetheless CFD simulations carried out at different free stream conditions, summarized in Table 9 and corresponding to the altitude of 75 km, reached about 3 minutes after the atmospheric entry interface, led to a maximum plasma frequency of 5.77 GHz, as reported in Table 10, and to a reduction along the capsule surface up to 1.44 GHz. Therefore, radio-communications blackout would have a relatively small duration (maximum three minutes of the total re-entry time, i.e. 30 minutes according to Figure 4), thanks to the very steep reduction of the total enthalpy, as depicted in Figure 15.

Table 9: Free stream conditions after 3 minutes after re-entry start

h [km]	М	V [km/s]	H [MJ/kg]	p [Pa]	T [K]
75	14.2	4.10	8.61	2.33	208

Table 10: Maximum values for the electron molar and volume fraction and for the plasma frequency in conditions listed in Table 9

Electrons	Electron volume	Plasma
molar fraction	fraction [cm <sup>-3</sup> ]	frequency [GHz]
5.15e-5	4.14e11	5.77

### 6.3 Effect of the material surface catalytic efficiency

The real distribution of the heat fluxes on the capsule surface is strongly affected by the catalytic recombination efficiency of the dissociated molecular species. In general, this property is evaluated in plasma wind tunnels for test conditions relevant to the flight mission program. In the present case no data are available for the materials that can be employed for the development of the rigid front shield and of the ceramic fabrics utilized for the deployable system. The catalytic efficiency of the materials defines the finite catalytic reaction rates of gases in chemical nonequilibrium on the material surface.



Figure 15: Comparison between total enthalpy and altitude variation along the capsule re-entry trajectory

During hypersonic flow through Earth's atmosphere, strong shock wave form in front of the re-entering capsule. Molecular oxygen and nitrogen molecules, with dissociation energies of about 500 kJ/mol and 950 kJ/mol, respectively, can dissociated in these extreme environment. If the resulting atomic species diffuse to the capsule surface and recombine there, this dissociation energy becomes available and some fraction may go directly to the surface as heat. The molecular species may recombine with different rates (depending on the catalytic efficiency of the materials).

Many studies on oxygen and nitrogen recombination have been carried out for different materials. The materials that can be in principle utilized for the proposed concept are Silica foams or Silica-Allumina based ceramic fibers, that exhibit a moderate catalytic efficiency to oxygen and nitrogen recombination.

In order to evaluate the effect of the partial catalytic efficiency at the typical conditions investigated during the present work, CFD simulations have been carried out using the nonequilibrium kinetic model presented in [Serpico et al. (1998); Monteverde et al. (2010); Monteverde and Savino (2012); Sciti et al. (2012)].

The heat fluxes distribution around the capsule, computed under the two extreme conditions of fully catalytic and non-catalytic surface, show that the maximum heat flux, for the fully catalytic surface can be also twice the corresponding value in the case of a non catalytic wall. For instance, for an altitude of 75 km and a value of

the Mach number equal to 20, corresponding to the conditions of maximum heat flux for a re-entry capsule with ballistic coefficient of 10 kg/m<sup>2</sup>, the stagnation point heat flux can be 160 kW/m<sup>2</sup> for the ideal case of zero catalytic recombination efficiency and 350 kW/m<sup>2</sup> in the opposite case of fully catalytic surface.

The corresponding radiative surface equilibrium temperatures, assuming a realistic value of 0.8 for the surface emissivity, are 1370 K and 1654 K, respectively.

The significant influence of the surface catalytic activity under reentry conditions suggests to dedicate future studies to the experimental characterization of the materials, selected for the thermal protection system of the capsule, through extensive plasma wind tunnel campaigns.

# 7 Control of the target re-entry point

The first leg of the re-entry trajectory shown in Section 4 has been computed considering an Average Solar Activity and a constant Drag Coefficient  $C_D=2$  (whose value has been calculated by DSMC simulations at an altitude of 250 km). In case Solar Activity or Drag Coefficient were different from the nominal ones, the possibility to control the trajectory, varying the capsule cross section area, has been afterwards considered: for a bevel angle variation from 39° to 45° the capsule crosssectional area, considering the conical shape of the umbrella-like heat shield, has a corresponding variation from 0.8 to 1.33 m<sup>2</sup>.

A suitable mechanism, able to provide the angle variation previously described, is currently under investigation, but different kinematic solutions are analyzed. In particular, as shown in Figure 1, the deployable heat shield is composed of a flexible woven fabrics connected to a mechanical frame composed by several ribs. They can be simply deployed using torsion springs and, once the umbrella is open, they can be adjusted with a kinematic mechanism driven by a single electromechanical actuator placed along the axis of the capsule, capable of translating worm screws through appropriate pulleys.

In order to show the importance of such a kind of control, the ground track error has been calculated and depicted in Figure 16 and in Figure 17 in terms of shift for the entry interface point, as a function of the solar radio flux and drag coefficients errors, respectively. It is evident that, even small uncertainties in the above mentioned parameters strongly shifts the entry interface point, resulting in a big dispersion of the landing point.

A simple algorithm for the capsule control in the first leg of the re-entry has been therefore provided. At each time step, in particular, a surface increment (or decrement) has been applied to the reference surface, in accordance with Equation 7.1.

$$\Delta S = k_1 \cdot (H - H_{nom}) + k_2 \cdot (\lambda - \lambda_{nom})$$
(7.1)



Figure 16: Entry interface point shift for different values of the solar radio flux error (with respect to the average conditions)



Figure 17: Entry interface point shift for different values of the drag coefficient error (with respect to the average conditions)

In the previous equation the surface correction, necessary to deal with the type of error analyzed, has been assumed to be proportional to the difference between the altitude detected by GPS system and the nominal one, along with the difference between the longitude detected by GPS and the nominal one (the constant gains have been assumed equal to  $10^{-1}$ ).

Figure 18 shows the surface correction computed in presence of a solar radio flux 2% higher than the average one and for different errors in the reference density  $\rho_0$  evaluation.

Finally, from Figure 19 it is possible to evaluate the effectiveness of the control algorithm for a solar radio flux 2% higher than the average one and a reference density 10% smaller than the nominal one. The controlled trajectory is in fact



Figure 18: Surface control necessary to cope with the nominal trajectory in presence of a solar radio flux the 2% higher than the average one and for different values of the nominal density at 175 km

compared to the nominal and to the uncontrolled ones, showing that it is very close to the former.

Analogous calculation have been carried out for the uncertainty in the drag coefficient, as shown in Figure 20 and Figure 21.

### 8 Conclusions

Calculations presented in the previous sections show that the presented capsule concept, thanks to its very low ballistic coefficient, is able to safely re-enter under relatively low aerothermal and mechanical loads, thus allowing a simple concept for the heat shield, based on inexpensive ceramic materials and fabrics. The final velocity at sea level, in the order of 10 m/s, allows to employ only terminal decelerators (e.g. small parachutes) or shock absorbers to mitigate the landing impact.

In addition, due to the relatively fast reduction of the total enthalpy throughout at very high altitudes, a relatively small duration of plasma blackout phenomena occurs, allowing GPS-based navigation for almost all the mission.

The possibility to control the re-entry trajectory in its first leg (using GPS systems) has also been analyzed: a preliminary and simple control algorithm has been



Figure 19: Comparison between a controlled trajectory and an uncontrolled one, in presence of a solar radio flux 2% higher than the average one and of a reference density 10% smaller than the nominal one

proved to be efficient to compensate deviations from the nominal trajectory due to uncertainties in density and in the drag coefficient.

The effects of the surface catalytic efficiency on the heat flux have been discussed showing the need for plasma wind tunnel characterization of the materials surface behavior.

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Figure 20: Surface control necessary to cope with the nominal trajectory in presence of different values of the drag coefficient error



Figure 21: Comparison between a controlled trajectory and an uncontrolled one, considering a drag coefficient the 5% smaller than the nominal one

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