DOI: 10.13009/EUCASS2022-7254

Thermite-for-Demise (T4D): Concept Definition for Exothermic Reaction-Aided Spacecraft Demise during Re-entry

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Abstract
Thermites are a particular class of energetic materials that, upon ignition, provide a considerable amount of enthalpy. This extra heat could be used to reach the complete demise of particularly robust components. In this paper, the main two design concepts of this technology are presented, along with guidelines extracted from literature and patents to improve their efficacy. The best sequence of events during the re-entry is then studied with respect to a selected test case through a parametric analysis, using a simple lumped mass re-entry model. The results indicate that for the considered case the best ignition timing corresponds to the temperature peak reached by the spacecraft during the re-entry.

1. Introduction

As shown by latest ESA's Space Environment Report [1], Earth's key orbits become crowded and crowded every year. Our behaviour in space is improving, but we are far from reaching sustainability in the long term. Even without further launches, it is reported that the number of space debris would increase in the next years due to collisions between already orbiting objects. This is particularly true for low-Earth orbits (LEO), that since 2015 have experienced a steep increase in the number of commercial miniaturized satellites, grouped in constellations.

Nowadays, the main countermeasure to this issue is to assure that all the satellites re-enter Earth atmosphere at the end of the operative life. International guidelines [2,3] set the maximum acceptable timeframe to perform the re-entry within 25 years after their operational phase. As natural decay within this timeframe is common for relatively small LEO satellites, an uncontrolled re-entry would be particularly cost-effective. However, such an approach must respect a stringent safety requirement: the casualty risk of a re-entering debris with kinetic energy ≥ 15 J must be lower than 1 in 10000.

In this framework, energetic materials could be used to assure a safe and cost-effective re-entry, avoiding the need of extra fuel to perform a controlled re-entry. It has been reported that such an end-of-life maneuver could cost up to 4 or 5 times the propellant needed on board, implying an important impact on satellite size, mass and cost [4]. Moreover, some common spacecraft components (e.g. ball bearing units, tanks, solar array mechanisms) are particularly robust and it is not easy to make them demisable with standard Design-for-Demise (D4D) techniques. An exothermic reaction during the re-entry could either induce a controlled fragmentation or provide the missing enthalpy for the complete demise of these components.

Thermites, a particular class of energetic materials based on metal / metal oxide couples, are good candidates for such application. They exhibit a highly exothermic reaction upon ignition, a usually high ignition threshold, and a wide versatility. Many parameters can be exploited to tailor their characteristics: formulation, granulometry, fuel to oxidizer ratio, compaction and production methods are the main ones. The work of Fischer and Grubelich [5], that explores just the formulation degree of freedom, is an example of the versatility of thermites. Their embodyment to promote spacecraft demisability is an idea that has been already partially explored by CNES [6], ESA [4] and Semenov Institute of Chemical Physics [7].

In the framework of SPADEXO (Spacecraft demise during re-entry expedited using various exothermic reactions) ESA-TRP project, research on thermites for a novel D4D technique is proceeding. In another publication [8] qualitative considerations on the requirements of such application as been expressed, as well as the presentation of the main thermite couples under investigation and typical values for temperature increase rate during a re-entry. In the

DOI: 10.13009/EUCASS2022-7254

hereby paper the definition of application concepts and the choice of the sequence of events during the re-entry will be addressed. These aspects, as will be discussed, are strictly connected.

Section 2 will present the main two concepts under investigation, derived from the applications in which thermites are employed on ground. The advantages of each concept will be highlighted. In the following sections the study of the best sequence of events during re-entry will be addressed. Section 3 will present the methods used in the analysis, Section 4 the results and the discussion of this study. Lastly, Section 5 will resume the main conclusions of this step of the project, and outline what is currently under investigation in SPADEXO project.

2. Possible concepts description

A recent ESA patent [4] describes in detail various integration concepts that can be considered for the application described in the hereby paper. Also considering thermite applications on ground, it is possible to divide all the embodiments in two main categories: those principally relying on direct contact between the reactive material and the target, and the ones more similar to the thermal torch concept. These two different approaches will be discussed in the next paragraphs, highlighting advantages and disadvantages of each.

2.1 Direct integration

Direct integration concept comprehends all the embodiments in which conduction and radiation from the reacting material are the main heat transfer mechanisms exploited. The main idea is to fill the structural spacecraft voids with thermite, so that the extra enthalpy released can assure the complete ablation of the target do be demised. A fuse can be present or not, depending on the capability of the reactive material to be ignited by the heat generated by the interaction between the satellite and the atmosphere, during the desired re-entry phase.

Similar on-ground applications can help in defining the main losses to be considered for this concept, and a set of strategies to limit them. Crane et al. [9] studied the interaction between a reacting Al-Fe_2O_3 reacting loose powder and a steel substrate. The powder was put in a v-notch and, upon reaction, just the 10% of the theoretical available energy was transferred to the substrate. The reason for such a low heat transfer efficacy were individuated into the losses due to convection and radiation. The gas generated by the mixture upon reaction (around 10%) and the expansion of the air filling the voids among the powder particles can be considered the main drivers for these losses. Also other on-ground applications operating in direct contact with the target, such as [10,11], consider a low gas generation as a key parameter to increase heat transfer efficacy. A confined geometry could be particularly effective to contain these losses, drastically increasing the efficacy of this concept. Moreover, it must be highlighted that convection is the main mode for energy propagation through loose powders [12]. The compaction of the powder could be beneficial to limit the energy lost due to the presence of air. Research on these two strategies is currently on-going at Space Propulsion Laboratory, Politecnico di Milano, and the first results are encouraging. Further publications will deal specifically with these aspects. Moreover, these strategies would be beneficial for a characteristic issue of re-entry thermite applications: the risk to lose chunks of molten material, as was experienced by ESA in past wind tunnel research [13].

2.2 Thermal torch concept

In this embodiment, the thermite reaction products are ejected through a nozzle and impinge on the target, transferring heat and provoking cuts of the desired shape [14]. This solution is used in various on-ground thermal torches [15,16] or military devices for ordnance disposal [17,18]. Globally, convection is the main driver of the heat transfer, but the study of the impingement of the reaction product droplets on the target highlights the effect of other variables, such as their heat capacity, density and kinetic energy [19]. As the kinetic energy is directly connected to the pressure reached before the nozzle, the gas generation of the reacting thermite couple is of paramount importance. Other parameters to be taken into consideration are the nozzle geometry, the distance between the nozzle and the target and the jet duration [20]. In general, a more compact jet, impinging on a smaller area, can induce greater damage [21]. However, it has been reported [22] that, depending on the gas generation properties of the mixture, the highest temperature may be detected not at the exit of the nozzle, but downstream. This is due to the fact that the reaction can proceed also downstream of the nozzle exit. An inclusion of O_2-generating binders or additives could be beneficial to improve the cutting rates.

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2.3 Discussion of the design concepts

The concepts described in the previous paragraphs can be compared to present advantages and disadvantages of each solution.

On one hand, it is evident that direct integration is the simplest concept. As the material is completely confined at least for the first part of the reaction and the subsequent heat transfer phase, no coupling with the external flow must be considered in this first phase. As for the torch-like concept, the possible interaction between the jet and the external conditions dramatically complicates the design. A possible solution to avoid this problem it to position or design the torch such that the nozzle is not subjected to the external flow.

On the other hand, while the direct integration gives little margin to direct the heat flux generated, the torch concept can be designed to focus the extra enthalpy onto a specific, localized target. This could be particularly beneficial for the destruction of structural connections. In this way, a controlled fragmentation could generate smaller debris, later demised by the interaction with the atmosphere. This design would be particularly effective in terms of the thermite mass needed, as there would be the need to heat up just a small portion of the spacecraft. However, this implies that the torch-like device must be kept firmly in position during the re-entry. A direct integration concept, embedded into spacecraft voids would be less sensitive to anchoring issues.

Lastly, it has been emphasized that the torch-like concept relies on a high-pressure generation, that on the contrary must be avoided for the other embodiment. This characteristic increases the risk of uncontrolled fragmentation and leads to the need of a robust casing. On-ground applications usually employ steel and ceramic materials for this component, that are not easily demisable and that should be avoided, if possible, when dealing with D4D.

In SPADEXO framework, the main investigation will be conducted on the direct integration concepts, because of the highlighted simplicity. However, the torch-like concept will be considered for structural connections, aiming at a design that could exclude the interaction between the jet and the surrounding flow. For both the concepts, a key design parameter is the desired sequence of events during the re-entry. This design choice is of paramount importance in determining the effectiveness of the additional enthalpy released. This topic is the subject of the parametric analysis presented in next Sections.

3. Description of the parametric analysis

The right ignition timing is a key parameter to properly exploit the extra enthalpy that can be given by an energetic material stored on board. As the ignition timing is not easy to be predicted with accuracy, also due to the high number of uncertainties that characterize a re-entry, in the hereby paper the wording "sequence of events" is preferred. The re-entry process is characterized by a complex coupling between convective heat up due to interaction with Earth atmosphere for hypersonic velocities, radiation losses and, in the case of this study, the enthalpy released by the reacting thermite. As far as the authors knowledge, this is the first study addressing this topic in detail.

The re-entry model used in this parametric analysis is rather simple and permits to obtain a reasonable set of data in limited time. This model has already been employed in [8] and it was shown to be rather conservative. It can be briefly described with these characteristics:

- Object-oriented code, three degrees of freedom;
- Atmosphere model is NRL-MSISE-00 [23];
- Properties of high-temperature air after the shock wave are represented through simplified curve fits given by [24,25];
- Air dissociation is considered as suggested by [26], with a five-species model;
- A hollow sphere geometry is considered, whose external radius is of 0.5 m, its thickness is 0.03 m and is made by aluminium;
- The re-entering object is considered as random tumbling, and this is represented by a uniform distribution of the heat received by the interaction with the atmosphere;
- Initial conditions for the test case are taken from [27] and reported in Table 1;
- Ablation is represented with a lumped mass model, and temperature is considered uniform on the whole body;
- Material properties are considered temperature independent in first approximation and are described in Table 2.

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Table 1: Initial condition for the test case.

	Value
Longitude	0 deg
Latitude	0 deg
Altitude	120 km
Velocity	7273 m/s
Heading	42.35 deg
Flight path angle	-2.612 deg

Table 2: Material properties considered in the simulations.

	Density	**Melting temperature**	**Heat of ablation**	**Specific heat capacity**	**Emissivity**	**Heat of reaction**
Aluminium	2713 kg/m^3	867 K	386116 J/kg	896 J/kg-K	0.141	-
Thermite (Al+Fe_2O_3)	4175 kg/m^3	-	-	787 J/kg-K	-	3958.2 kJ/kg

The aerothermodynamic model, having a paramount influence on the subsequent results, is now presented more in detail. As commonly done in re-entry models three regions are individuated on the basis of the Knudsen number value.

- For Kn ≥ 10, pure free-molecular regime is considered;
- For Kn ≤ 0.01, continuum regime is considered fully established;
- For intermediate values, a bridging function is emplyed.

In free molecular regime the approach described in [27] is followed:

$$\dot{q}_{fm}^{conv} = F \cdot 11356.6 \cdot \left(\frac{a_t\, \rho_0 V_0^3}{1156}\right) \quad (1)$$

Where $\dot{q}_{fm}^{conv}$ is the convective heat flux for the free-molecular regime, ρ_0 and V_0 are respectively the free-stream density and velocity, F is the suggested shape factor, equal to 0.255 and a_t is the thermal accommodation coefficient, considered constant and equal to 0.9.

For the fully established continuum regime the Fay and Riddell [28] correlation is used:

$$\dot{q}_{cr}^{conv} = \frac{0.763}{Pr_{w,t}}(\rho_{t2}\mu_{t2})^{0.4}(\rho_{w,t}\mu_{w,t})^{0.1}\sqrt{\left(\frac{du_e}{dx}\right)_{t2}}(H_{t2} - h_{w,t})[1 + (Le^{0.52} - 1)]\frac{h_D}{H_{t2}} \quad (2)$$

Where $\dot{q}_{cr}^{conv}$ is the convective heat flux for the continuum regime, $Pr_{w,t}$ is the Prandtl number at the wall, ρ_{t2} and μ_{t2} are respectively the density and the dynamic viscosity at the stagnation point, $\rho_{w,t}$ and $\mu_{w,t}$ are respectively the density and the dynamic viscosity at the wall, the term $(du_e/dx)_{t2}$ is the velocity gradient at the stagnation point, H_{t2} is the stagnation enthalpy, $h_{w,t}$ is the enthalpy at the wall, Le is the Lewis number considered equal to 1.4 and h_D is the

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dissociation energy of the high-temperature air after the shock wave. As highlighted by Fay and Riddell in [28], it is important to note that in this relation the external properties of the flow are much more important in determining the heat transfer rate than the wall values. This is due to the fact that the growth of the boundary layer, which deeply influences the heat transfer, depends mostly upon the external flow properties.

For values of Kn between 0.01 and 10, the following bridging function taken by [29] is chosen:

$$\dot{q}_{tr}^{conv} = \frac{\dot{q}_{cr}^{conv}}{\sqrt{1 + \left(\frac{\dot{q}_{cr}^{conv}}{\dot{q}_{fm}^{conv}}\right)^2}} \quad (3)$$

Where q_{tr}^{conv} is the convective heat flux received by the re-entering spacecraft in transition regime.

As for the radiative heat transfer between the spacecraft and the external environment, a simple relation is employed for all regimes:

$$q^{rad} = -\varepsilon\sigma T_w^4 \quad (4)$$

Where ε is the emissivity of the spacecraft material, σ is the Boltzmann constant and T_w is the wall temperature.

Lastly, the enthalpy released by the thermite during the re-entry is simply modelled as:

$$q^{th} = \frac{m_{th} Q_{react}}{S_{int}} \quad (5)$$

Where m_{th} is the thermite mass, Q_{react} is the specific heat release (in this study, for the couple $Al+Fe_2O_3$ is taken equal to 3958.20 kJ/kg) and S_{int} is the internal surface of the hollow sphere. The mass of thermite considered for the parametric analysis is about 100 kg for a spacecraft mass of about 240 kg. This mass ratio is not realistic, but it was selected to be able to see clearly the influence of the thermite ignition on a lumped-mass object. The oxidizer to fuel ratio is stoichiometric. As it can be seen in Figure 2a, presenting the temperature profile during the re-entry for both the cases without thermite charge and without thermite ignition, the final temperature reached just because of the heat received by the interaction with the atmosphere is about 200 K below the melting temperature of the object, here considered 870 K. The heat released by the thermite charge is needed to reach the melting temperature and start the ablation phase. This extra enthalpy is released, depending on the simulation, according to one of the five profiles presented in Figure 1. As the integral of each profile over time is equal to 1, in all the simulations the same amount of extra heat was released. Profile 1 is step-like, the simplest possible; Profile 2 is normal and has been used to approximate a heat release very concentrated in time; Profiles 3, 4, and 5 are triangular and approximate basic combustion behaviour.

The variation in heat profiles was also coupled with a variation in ignition time, further detailed is presented in Table 3. Four points of interest have been selected for each profile:

- Early ignition, well before the other points considered (later referred to as “Early”);
- Ignition at temperature increase rate peak (later referred to as “TG_{peak}”);
- Ignition at temperature peak (later referred to as “T_{peak}”);
- Late ignition, well after the other points considered (later referred to as “Late”).

The thermite contributes both to the mass of the re-entering object, being stored in it, and to its specific heat. The properties considered for the thermite are presented in Table 2. The thermite is stored inside the re-entering sphere, so its emissivity or view factor have no impact on the simulation. As the thermite does not take part neither the ablation model, only the thickness of the hollow sphere and aluminium heat of ablation are considered.

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Table 3: Thermite heat release profiles characteristics.

Ignition timing	**Start**	**Profile**	**Duration**
Early	90 s (50 s) [a]	1	100 s
TG_{peak}	195 s (155 s) [a]	2[b]	100 s
T_{peak}	260 s (220 s) [a]	3	50 s
Late	280 s (240 s) [a]	4	50 s
		5	50 s

[a] Just for Profile 2, these value are the ones indicated in parenthesis.
[b] For Profile 2, expected value is 50 and standard deviation is 5.

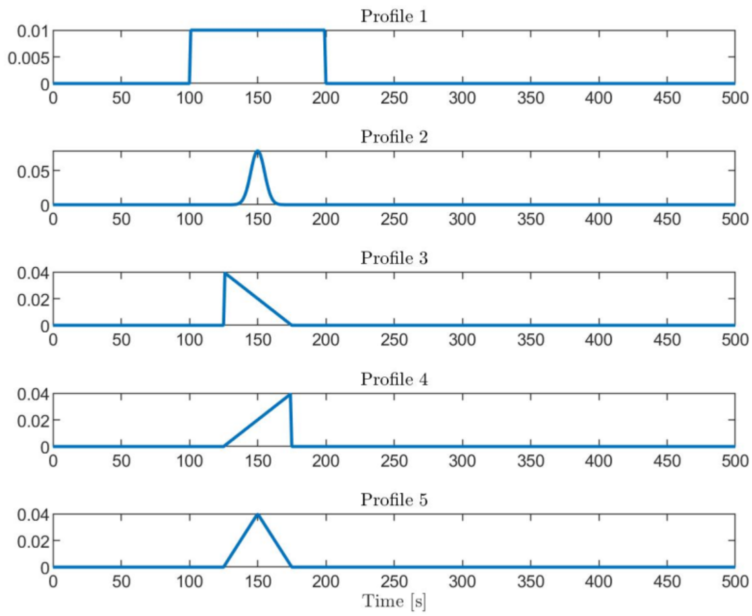

Figure 1: Thermite enthalpy release considered in the parametric analysis. The ignition time is modified at each simulation according to Table 3.

4. Results and discussion of the parametric analysis

Firstly, a comparison with the reference cases, both without thermite and without thermite ignition, can be useful to understand more easily the results of the parametric analysis. In Figure 2a the temperature profile during the re-entry is shown in these two cases: as expected, a failed ignition penalizes the demise, because of the extra mass on-board. Figure 2b presents the heat flux computed for the re-entering sphere without thermite.

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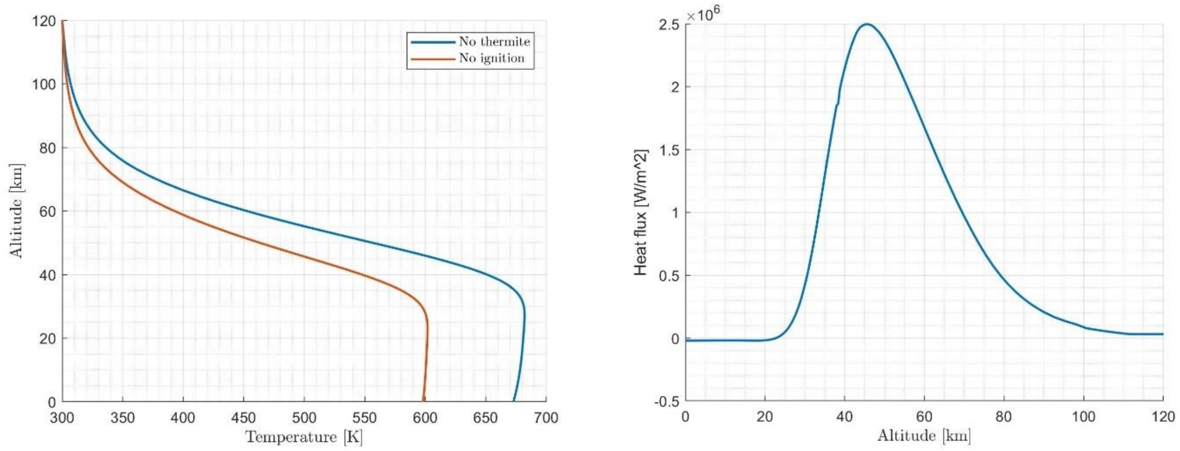

Figure 2: a) On the left, temperature profile during the re-entry for reference cases without thermite (blue) and without ignition (red). b) On the right, heat flux profile during the re-entry for reference case without thermite.

The selected parameter of merit for this analysis is the final wall thickness of the sphere. Figure 5 summarizes this result for each case analysed. A couple of cases have been selected to let the reader understand better how the parametric analysis is performed. For example, Figure 3a shows the temperature profile during the re-entry in the Profile 1 cases. It can be noted that in all the cases the hollow sphere reaches the melting temperature and starts the ablation phase. The Early case is the only one in which the heat release from the thermite is already ended when the ablation phase starts. In Figure 3b, the superposition between the bell-shaped profile shown in Figure 2b and the step-like heat release during the re-entry can be observed. Figures 4a and 4b show the same data for Profile 3. In this case it is particularly easy to appreciate the different ignition timing in Figure 4b, as the Profile 3 is shorter in time with respect to Profile 1.

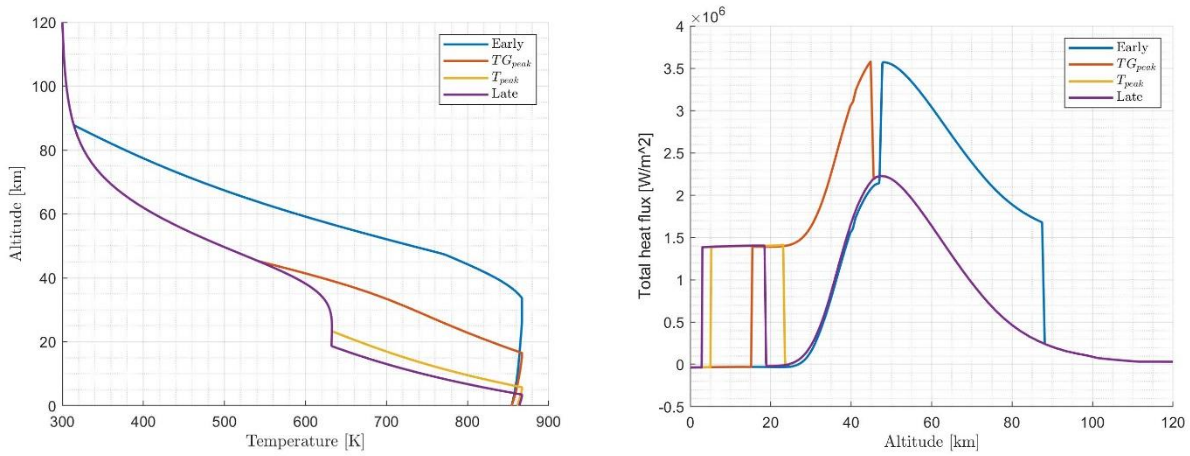

Figure 3: a) On the left, temperature profile during the re-entry for Profile 1, evaluated at different ignition times. b) On the right, heat flux profile during the re-entry for Profile 1, evaluated at different ignition times.

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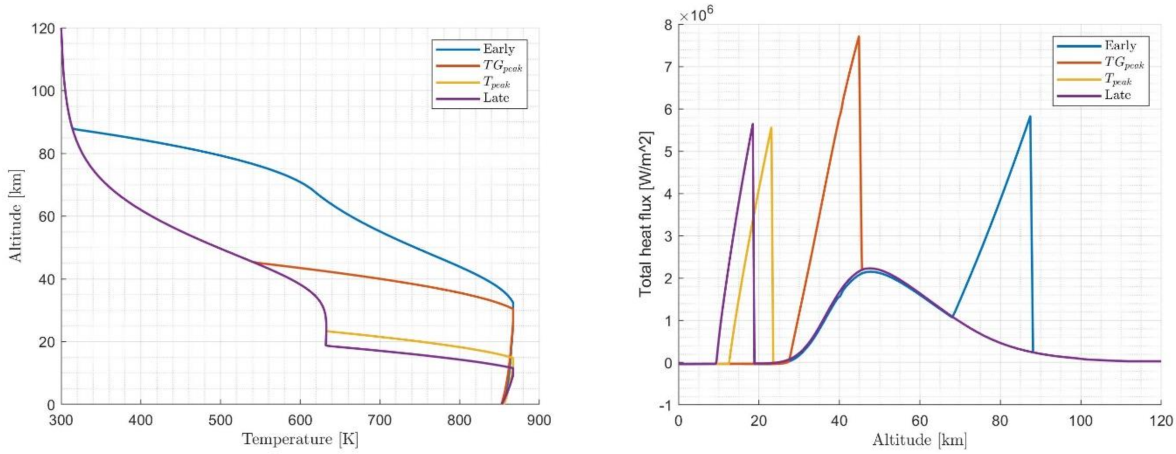

Figure 4: a) On the left, temperature profile during the re-entry for Profile 3, evaluated at different ignition times. b) On the right, heat flux profile during the re-entry for Profile 3, evaluated at different ignition times.

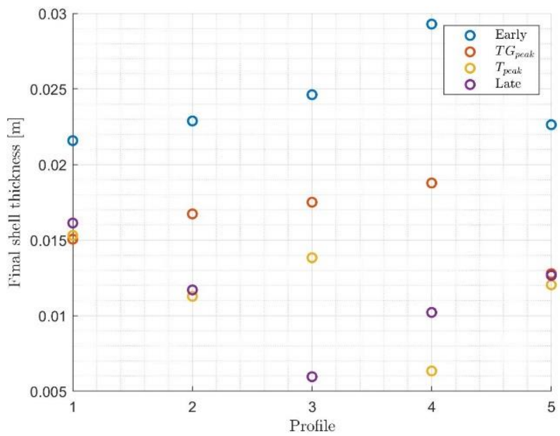

Figure 5: Final shell thickness, at varying Profile applied and ignition time. Starting shell thickness is 0.03 m.

As it can be seen in Figure 5, the most effective cases are either the Late one or the T_{peak} one, for all the five Profiles. This behaviour is more pronounced for the brief heat release Profiles. The reason behind this behaviour can be understood by considering the total enthalpy received by the re-entering object. While the thermite contribution is the same for all the cases, by the definition of the five Profiles, the contribution by the convective and the radiative components varies. A couple of considerations can explain this trend.

A first explanation can be deduced just considering the aerothermodynamics model. Even if, as seen in Section 3, the convective heat flux is more sensitive to the external flux characteristics that to the wall temperature, a higher wall temperature will determine a smaller temperature difference between the wall and the high-temperature air after the shock wave, hence inhibiting the convective heat flux received. At least for this case, the radiative losses are not high enough to make an early ignition beneficial. This deduction can be verified by plotting the difference between the total heat flux experienced by the spacecraft and the heat flux given by the thermite. In Figure 6 this plot is shown for the case of Profile 3. It is evident that the Early curve reaches lower values with respect to the others, possibly because the re-entering sphere temperature is higher when subjected to the heat transfer with the atmosphere.

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A second explanation could be the change of the ballistic coefficient induced by the decrease of mass during the descent. An earlier ignition means an earlier decrease in mass, implying lower velocities and therefore less heat flux to the re-entering body. This consideration can be verified by looking at Figure 7, where the velocity along the re-entry is plotted for Profile 3 at different ignition times. As it can be seen particularly in Figure 7b, the velocity difference can be detected by comparing the velocity profiles for different sequence of events. When the mass starts decreasing, the velocity diminishes with respect to the cases in which the thermite ignition happens later. Looking at Figure 7b, it may seem that the TG_{peak} ignition profile reaches lower velocity values than the Early ignited one. This is true for altitudes below 30 km, but for higher altitudes this behaviour is reversed.

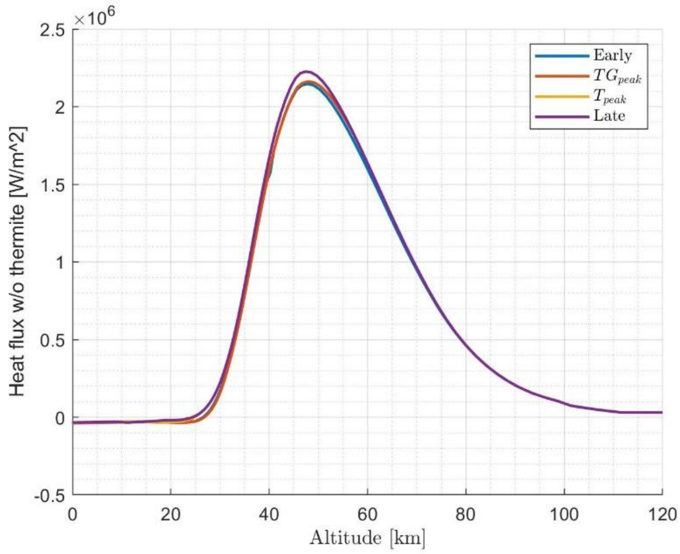

Figure 6: Heat flux profile obtained for Profile 3 at different ignition times, not considering the thermite contribution. T_{peak} and Late profile are superposed during almost all the simulation.

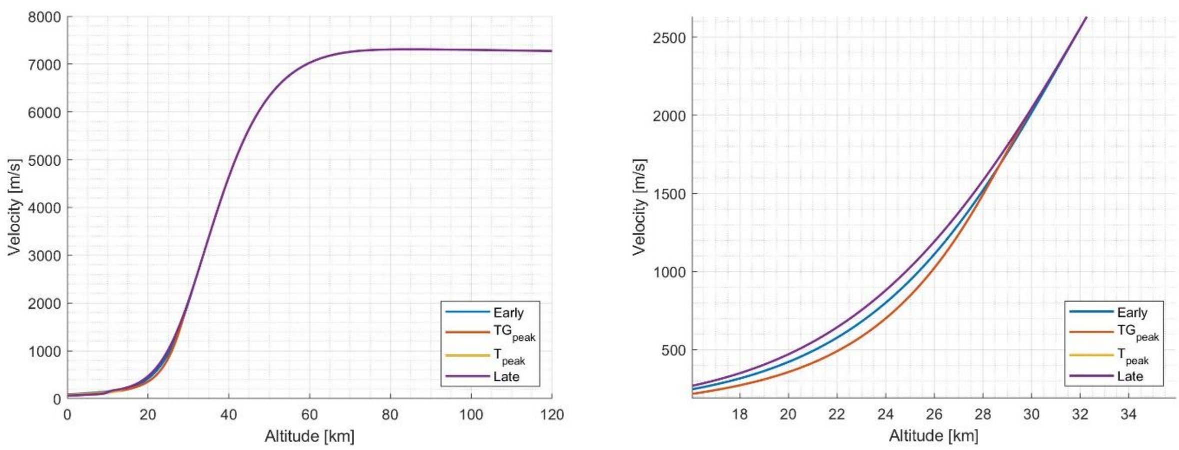

Figure 7: a) On the left, velocity profile during the re-entry for Profile 3 at different ignition times. b) On the right, detail of Figure 7a. T_{peak} and Late velocity profiles are superposed. TG_{peak} velocity profile is lower than Early profile just in the detail considered, during the rest of the simulation is superposed to T_{peak} and Late ones.

5. Conclusions

This paper presents a first analysis of the possible concepts for thermite applications in spacecraft re-entry. Available literature and patents have been reviewed looking for strategies to improve the energetic material effectiveness in providing additional heat to the spacecraft. Two main categories of embodiments have been identified, one relying on

DOI: 10.13009/EUCASS2022-7254

conduction as principal heat transfer mechanism, the other on convection. The advantages and disadvantages of each have been pointed out, highlighting the main challenges in their development.

The necessity of understanding the best ignition timing for such an application has been recognized, and a parametric analysis was performed. A reference re-entry case, considering an aluminium hollow sphere containing thermite, has been subjected to five different heat release profiles, with different ignition times. The results showed that a late ignition, close to the maximum temperature point of the re-entry, allows the best exploitation of the heat generated by the interaction with the atmosphere and is the best sequence of events for the considered case. Two possible explanations were given for this trend. Firstly, a late ignition can assure a long-lasting higher temperature difference between the wall of the re-entering body and the high-temperature gas behind the shock wave, hence promoting the heat transfer. Secondly, an early decrease in mass influences the ballistic coefficient provoking a decrease in the velocity during the re-entry, hindering the heat flux received by the spacecraft.

This analysis will be conducted also for the cases under investigation in SPADEXO framework. The next steps of the project include the realization of thermite charges designed for specific spacecraft components. This solution will be tested in hypersonic wind tunnel to get a deeper understanding of the efficacy of this application.

6. Acknowledgements

The authors acknowledge the financial support of the SPADEXO project by the European Space Agency through the Contract No. 4000136082/21/NL/MG.

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