Material Properties Evolution Affecting Demisability for Space Debris Mitigation

Chetan Mahawar, Sarath Chandran, Sridhar Panigrahi, V. P. Shaji

Abstract—The ever-growing advancement in space exploration has led to an alarming concern for space debris removal as it restricts further launch operations and adventurous space missions; hence various technologies and methods are explored for re-entry predictions and material selection processes for mitigating space debris. The selection of material and operating conditions is determined with the objective of lightweight structure and ability to demise faster subject to spacecraft survivability during its mission. The various evolving thermal material properties such as emissivity, specific heat capacity, thermal conductivity, radiation intensity, etc. affect demisability of spacecraft. Thus, this paper presents the analysis of evolving thermal material properties of spacecraft, which affect the demisability process and thus estimate demise time using the demisability model by incorporating evolving thermal properties for sensible heating followed by the complete or partial break-up of spacecraft. The demisability analysis thus concludes that the best suitable spacecraft material is based on the least estimated demise time, which fulfills the criteria of design-for-survivability and as well as of design-for-demisability.

Keywords—Demisability, emissivity, lightweight, re-entry, survivability.

I. Introduction

THE diverse application of space technology for Earth observation, weather forecasting, telecommunication service, and remote sensing has led to an accumulation of obsolete satellites which results in systematic congestion of orbital regions of LEO (Low Earth Orbit) and GSO (Geo-synchronous orbit). But due to limitations in the capacity of orbits the Inter-Agency Space Debris Coordination Committee [1] defines the guidelines for the mitigation and decommissioning of non-operational satellites and spacecraft in space like PS3, PS4, CUS, etc., to avoid an accumulation of debris and debris impact on other satellites. The disposal of any object in LEO is classically achieved by natural re-entry in a short time due to atmospheric drag but for satellites in GSO and Medium Earth Orbit (MEO), different strategies and approaches have been proposed in numerous literature [4], [12], [15] to reduce impact hazards in these orbits. But the easiest approach for satellite disposal in GSO orbits is through atmospheric re-entry which occurs for highly inclined orbits using lunisolar perturbations and is defined by entry corridor of crewed vehicles [2].

The re-entry predictions are analyzed using computations of overshoot boundary as described by [3] on basis of the adimensional variable where it is assumed that the entry occurs when the deceleration due to the aerodynamic forces reaches a specific fraction of the gravitational acceleration. This adimensional variable is dependent on material properties, eccentricity, flight path angle, drag coefficient, spacecraft configuration, etc. corresponding to each trajectory for re-entry. The re-entry prediction for GSO trajectories is performed by the Phoenix tool using a representative spacecraft configuration [4]. This tool can also be used for natural re-entry predictions in the case of LEO orbits too.

Disposal via atmospheric re-entry should comply with casualty risk no higher than 10^{-4} for safer re-entry and no damage to the ground population. To comply with this requirement, spacecraft should follow the design-for-demise philosophy and simultaneously design-for-survivability because a satellite designed for demise has to also withstand debris impact for many years during an operational life span in its orbit. The demisability design focuses on faster thermal degradation at the desired pericenter altitude and flight path angle during its re-entry and this occurs with the achieved melting point of the respective material. The rate at which melting point is reached depends upon net heat flux which in turn depends on thermal material properties, mass, irradiation, configuration of the structure, etc., as net heat flux is the deciding factor for spacecraft temperature variation due to sensible heating followed by melting and break-up of spacecraft and its components. This net heat flux is the cause of increasing heat accumulation in the structure and is developed due to environmental drag and thermal radiation. But since the spacecraft has to perform its functional operation for the desired life span, its structural integrity with design for demise has to be accounted for and thus structure needs to be lightweight, and have high specific strength and specific stiffness. The basis of selection of structure material is to withstand thermo-mechanical loads without failure in desired temperature range and solar radiation along with the ability to demise faster. Thus, this paper presents a parametric analysis of evolving thermal material properties which affect demisability time and thus determine the most suitable spacecraft material with constraints on some of the thermal material properties while simultaneously fulfilling design-for-survivability.

II. Methodology

The material selection process which affects the design for demisability and survivability is constrained by the following major design requirements:

Chetan Mahawar, ‘Sci./Eng.C’, Sarath Chandran, ‘Sci./Eng.D’, Sridhar Panigrahi, ‘Division Head’, and V. P. Shaji, ‘Group Head’, are with LPSC Valiamala (ISRO), Trivandrum, India (e-mail: chetan_mahawar@lpsc.gov.in, c_sarath@lpsc.gov.in, sridharpanigrahi@lpsc.gov.in, vp_shaji@lpsc.gov.in).

- Material should be lightweight and withstand thermo-mechanical load during launch and in-orbit operations. Thus, minimum yield strength and elastic modulus should be 5 MPa and 2.5 GPa respectively [5]-[7].
- Material of structure shall not degrade under solar radiation.
- Material shall operate without failure in the temperature range of -20 ℃ to +80 ℃ [8], [9] with a maximum thermal expansion coefficient of 100 µstrain/℃.
- Material should have optimum melting point and thermal degradation rate for faster demisability.

The heat of ablation and variation of thermal material properties on re-entry to the atmosphere decides the elapsed time for thermal degradation because the structure is thermally heated to its melting point due to irradiation, friction drag, etc., which further results in break-up and melting of spacecraft. The heating and melting phenomenon can be mathematically expressed by the heat transfer equation during sensible heating and latent fusion. Thus, thermal material properties such as emissivity and specific heat capacity are governing thermal parameters for demisability of structure which evolves with time due to dependency on spacecraft temperature. Hence, it is of prime importance to study and incorporate the effect of evolving thermal material properties on demisability analysis during re-entry to the atmosphere. This is to decide the most appropriate candidate material for faster demisability considering the constraint of design-for-survivability. Hence, the approach adopted in this paper will be to analyze the variation in major thermal material properties like emissivity and specific heat capacity of materials fulfilling the constraints of design-for-survivability and then incorporate all the variations in demisability model so to estimate total demise time for each selected material according to survivability design. The demisability time for selecting the most suitable material is analyzed by following two methodologies.

A. Demise Time Estimation Assuming Complete Demisability

The demise time is estimated considering complete breakup of spacecraft as determined by Liquid Mass Fraction (LMF) due to latent heating, irrespective of spacecraft location in atmosphere. But since all materials may not achieve melting point on touching the earth's surface therefore the assumption of complete spacecraft break-up will be impractical, but still, the approach holds good for comparing demisability time amongst selected materials and thus determines the most suitable candidate material fulfilling survivability and demisability conditions.

B. Demise Time Estimation Considering Actual Demise

The more realistic methodology to estimate demise time is by considering the actual temperature attained by spacecraft on touching the earth's surface in reference to Belstead's research [13], [15], so as to determine phase change in material and hence correspondingly demisability time. Then the material with the least total demise time among the highest LMF value is selected as the most suitable spacecraft material fulfilling all design conditions.

III. Spacecraft Ablation Analysis

The re-entry of spacecraft into atmosphere on completion of its mission in orbit undergoes severe intensity of irradiation and friction drag leading to an increase in spacecraft temperature followed by thermal degradation and break-up of spacecraft. Thus, ablation analysis of spacecraft is performed using demisability model in a phased manner from the point of atmospheric re-entry.

A. Demisability Model

The demisability model is analyzed in two phases, first where sensible heating of spacecraft takes place and second where spacecraft undergoes thermal degradation due to latent heating. The temperature variation during the sensible heating phase of atmospheric re-entry is described by (1) until the temperature of the object achieves a melting point corresponding to the material. The rate of increase in spacecraft temperature is governed by net heat flux, thermal storage capacity, initial weight, and configuration of structure as expressed by heat exchange equation.

$$\frac{dT_w}{dt} = \frac{A_w}{m_o C_{p,m}(T_w)}\left[\mathrm{q_{av}} - \in (T_w)\sigma \mathrm{T_w^4}\right] \tag{1}$$

where T_w is the instantaneous spacecraft temperature; A_w is the wetted area; m_o is the initial mass of the equivalent object; $C_{p,m}(T_w)$ is varying specific heat capacity of the material; q_{av} is the average heat flux on the object accounting for shape and attitude-dependent factors; $\in (T_w)$ is the varying emissivity of a material; σ is the Stefan-Boltzmann constant.

Once the melting temperature is reached, the object starts melting and loses mass at a rate that is proportional to the net heat flux on structure due to space environment and emitted radiation from object and inversely proportional to the heat of fusion (h_m). The depletion of mass with time beyond a certain level of altitude is expressed by (2) which depends on structural configuration and thermal material properties.

$$\frac{dm}{dt} = -\frac{A_w}{h_m}\left[\mathrm{q_{av}} - \in (T_w)\sigma \mathrm{T_w^4}\right] \tag{2}$$

where m is an instantaneous mass of the spacecraft; h_m is the latent heat of fusion; T_w is constant temperature during melting phase.

The major assumption to highlight in the demisability model is that the body is considered a lumped mass because the conductivity of structure is considered infinite, hence temperature is uniform everywhere in the volume of the object. This approximation holds good for metallic structure but for non-metallic materials, such as composites, an approach considering metallic equivalent properties is used for analysis [10]. The other major assumption is the use of average heat flux rate which is dependent on the object shape and attitude factors [11] and since for the analysis, only spherical-shaped structure is considered therefore its value remains constant irrespective of the material.

B. Spacecraft Re-Entry Equations of Motion

The evolution of trajectory during atmospheric re-entry is analyzed by considering the spacecraft as a point mass with predefined motion parameters. This is required to study the variation in spacecraft temperature with time by knowing the altitude variation with time according to the re-entry equations of motion [12], since temperature variation with altitude is known from an experimental database [13]. The time variation of other thermal properties dependent on spacecraft temperature can be also determined similarly during re-entering flight trajectories. The equations of motion are the result of variation in aerodynamic lift, drag, friction, and gravitational force which cumulatively decide the acceleration of re-entering spacecraft. But for defining equations of motion, a reference frame is decided and for convenience, the reference frame selected is rotating with the atmosphere because a planet's atmosphere rotates with spacecraft hence a planet-fixed reference frame is used in order to express the equations of motion [14] given by (3):

$$v_\infty \cos\gamma = \frac{{v_\infty}^2}{r}(\cos\gamma)^2\, sin\alpha tan\delta - g_\delta sin\alpha + \omega^2 r sin\alpha sin\delta \cos\delta - \, 2\omega\, v_\infty(\sin\gamma\, cos\alpha cos\delta - \cos\gamma\, sin\delta) \quad (3)$$

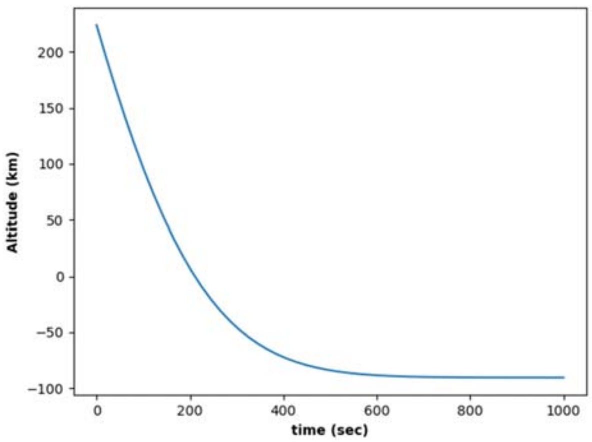

Fig. 1 Altitude variation during atmospheric re-entry

where g_c and g_δ are the component of the gravitational acceleration; ω is the angular rotation velocity of planet; v_∞ is the relative velocity of magnitude; γ is flight path angle; α is azimuth in horizontal plane; δ is longitude.

On solving (3) with flight varying parameters and initial re-entry conditions [15] as shown in Table I, we get the altitude variation with time during descent as shown in Fig. 1.

TABLE I
INITIAL CONDITIONS OF RE-ENTRY

S. No.	Initial conditions	Values
1.	Longitude (deg)	0
2.	Latitude (deg)	0
3.	Altitude (km)	120
4.	Velocity (m/s)	7273
5.	Heading (deg)	42.53
6.	Flight path angle (deg)	-2.612

C. Material Thermal Properties Evolution

The temperature profile variation with altitude corresponding to shape configuration and material type is extracted based on experimental results obtained from Belstead's research [11] presented during first demise workshop for the initial re-entry conditions as summarized in Table I. The variation in temperature as shown in **Error! Reference source not found.** is observed for Al 7075-T6, Ti 6Al4V, and SS 304 corresponding to only spherical shape structure amongst various structural configurations like cylinder, box, and plate because configuration has a negligible effect on temperature profile and accounts as an equivalent shaped body (A_w) corresponding to the actual flight.

The evolution of material properties such as emissivity and specific heat capacity, dependent on spacecraft temperature is incorporated in demisability model based on the material considered for the analysis. The material selection for demisability analysis is amongst the material fulfilling the criteria of design-for-survivability and light-weightedness. As concluded in [16], [17], alloys Al 7075-T6, Ti 6Al4V, SS 304 are the most appropriate material with decreasing priority in the order of their arrangement as per the design for survivability approach. Thus, the present study uses the material properties of above-mentioned materials to determine demisability time and thus find the most suitable material which holds good for design-for-survivability as well as design-for-demisability.

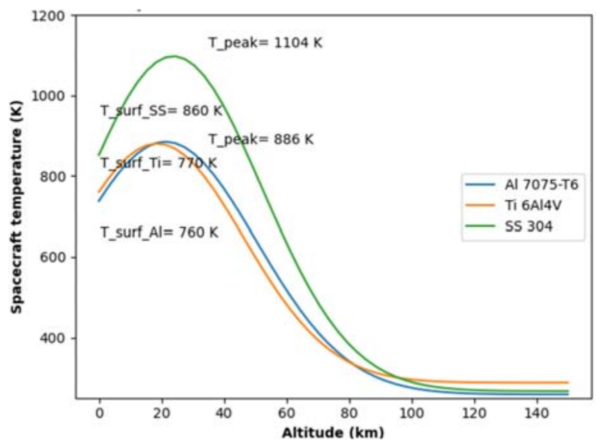

Fig. 2 Spacecraft temperature variation with altitude

The variation in emissivity and specific heat capacity [18] with spacecraft temperature for each candidate material are shown in Figs. 3 and 4, respectively. It can be concluded from the plots that heat storage capacity and emitted radiation power are more for SS 304 and least for Al 7075-T6. Hence, both emissivity and specific heat capacity of Al 7075-T6 will aid to faster demisability in comparison to other materials, since it will lead to higher heat accumulation in structure. The other thermal material properties [19] such as melting point and heat of fusion used in demisability model are listed in Table II.

IV. RESULTS

The demisability analysis is performed by calculating the total demise time for spacecraft thermal degradation from the point of re-entry to atmosphere. The demise time includes the time for sensible heating of spacecraft until it reaches its melting point followed by time to melt and break up partially or completely. The demise time estimation for the thermal degradation process is found by incorporating evolving thermal material properties as discussed above along with the initial parameters as shown in Table III and considering a spherical-shell shaped structure equivalent to a spacecraft body.

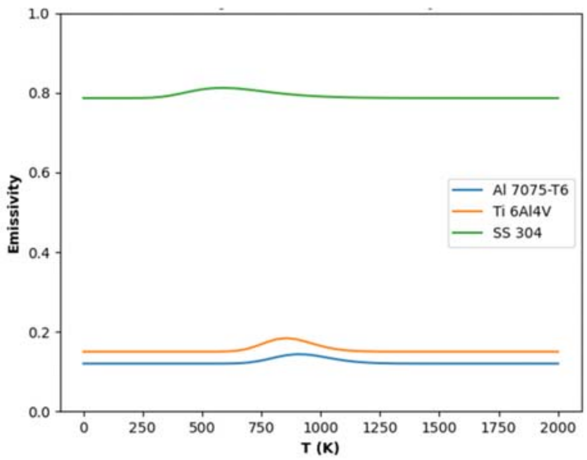

Fig. 3 Emissivity variation with spacecraft temperature

TABLE II
MATERIAL DATABASE

	Al 7075-T6	Ti 6Al4V	SS 304
Melting point (K)	750	1943	1700
Heat of fusion (J/kg)	376788	393559	286098

TABLE III
SPACECRAFT INITIAL PARAMETERS

Parameters	Initial values
Spacecraft mass	30 kg
Spacecraft equivalent dimension	diameter = 1 m
	thickness = 0.03 m
Re-entering temperature	300 K
Average heat flux	794425 W/m^2

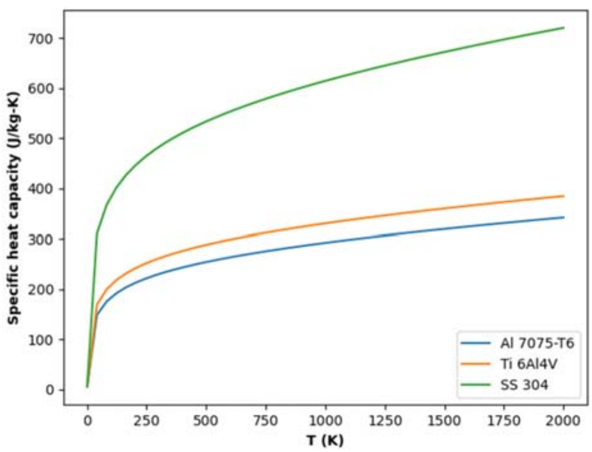

Fig. 2 Specific heat capacity variation with temperature

The results for the computations of demise time using the demisability model equations for the two phases of ablation are shown in Figs. 5 and 6, which are used to select the most suitable candidate material fulfilling design-for-survivability and demisability methodology.

V. DISCUSSION

The material selection criteria for a successful mission of spacecraft are being lightweight, capable enough to withstand thermo-mechanical loads during launch and in-orbit operations, operational within a desirable temperature range, and as well as having faster demisability. The materials used for demisability analysis are amongst the material concluded based on design-for-survivability model, to decide the most suitable candidate material for the spacecraft to have sustainable operation. Thus, in demisability analysis, material is selected based on the least total demise time required for thermal degradation of spacecraft which is determined using the demisability model which considers sensible heating of spacecraft until melting point is reached followed by time to melt and break-up due to latent heating. This demisability model incorporates material properties dependent upon spacecraft temperature which evolves with altitude and eventually with time, hence able to determine demise time with higher accuracy during atmospheric descent. The melting and break-up of spacecraft is expressed by LMF which is the fraction of mass that demises during the re-entry. Therefore, its value lies in range 0 to 1 where value 1 corresponds to complete demise and the value of 0 to complete survival. The time estimation for sensible and latent heating is done separately with reference as t = 0 sec (but in actual will be continuous) and then summed up to get the total demise time for complete thermal degradation process.

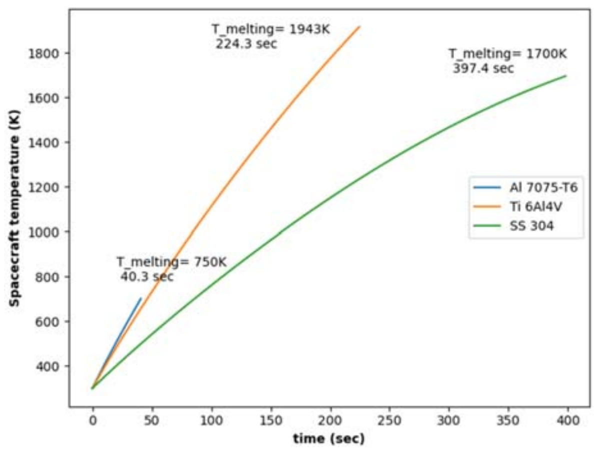

Fig. 5 Spacecraft sensible heating time during re-entry

TABLE IV
DEMISE TIME BASED ON COMPLETE BREAK-UP

Demisable phase	Al 7075-T6	Ti 6Al4V	SS 304
Sensible heating time (sec)	40.3	224.3	397.4
Latent heating time (sec)	149	185.6	200
LMF	1	1	1
Total time (sec)	189.3	409.9	597.4

The demise time for thermal degradation phenomenon is estimated using two approaches. The first approach estimates demise time by considering complete break-up of spacecraft irrespective of spacecraft location in atmosphere, thus assuming each material attains melting point and undergoes phase change phenomenon. Hence the estimated time from Figs. 5 and 6 corresponding to each material is given in Table IV to select the material with least total demise time. But since every material may not undergo phase change process on touching earth's surface, therefore the approach of determining actual demise time may seem impractical, but still, the approach holds good for comparing demisability time of different materials so to decide the most suitable spacecraft material amongst selected list of material.

The other realistic approach to tabulate demise time from Figs. 5 and 6 is based on the temperature attained by respective spacecraft material as melting point on touching earth's surface in reference to Fig. 2 as per Belstead's research database, so to decide whether material will undergo phase change or not. It can then be inferred from Table V that the total demisability time will be the time only for spacecraft sensible heating if particular spacecraft material does not undergo phase change process. Hence the most suitable spacecraft material in this approach will be the one with the least total demise time corresponding to the highest LMF value, as the major purpose is to have complete spacecraft break-up on touching earth's surface.

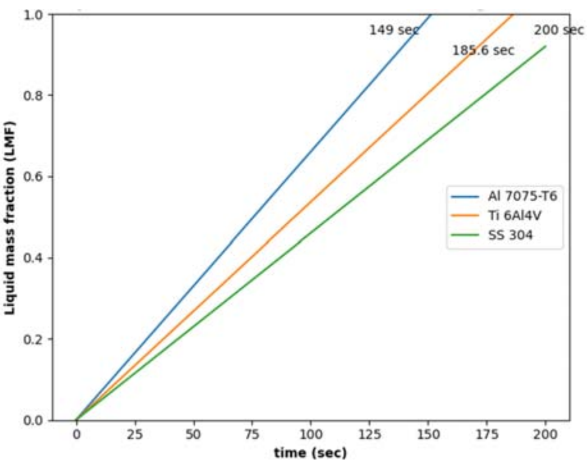

Fig. 6 Spacecraft thermal degradation time during descent

TABLE V
DEMISE TIME BASED ON ACTUAL DEMISABILITY

Demisable phase	Al 7075-T6	Ti 6Al4V	SS 304
Sensible heating time (sec)	40.3	64.5	186.7
Latent heating time (sec)	149	-	-
LMF	1	0	0
Total time (sec)	189.3	64.5	186.7

VI. CONCLUSION

It can be concluded from both methodologies that Al 7075-T6 is the most suitable candidate material as per design-for-survivability and design-for-demisability due to least demisablility time for complete break-up of spacecraft on touching earth's surface. It can also be concluded from the ablation analysis that materials with an emissivity of order .1 to .3 and specific heat capacity of 250 J/(kg-K) to 400 J/(kg-K) can be amongst the desirable spacecraft material as values of thermal properties are closer to that of Al 7075-T6 thermal material properties and are major governing parameters for demisability analysis.

REFERENCES

[1] IADC, "Inter-Agency Space debris Coordination Committee," 2011.

[2] D. R. Chapman, "An analysis of the corridor and guidance requirements for supercircular entry into planetary atmosphere," National Aeronautics and Space Administration, pp. 1-49, 1960.

[3] N. X. Busemann, R. D. Culp and e. a. Vinh, Hypersonic and planetary entry flight mechanics, Ann Arbor: University of Michigan Press, 1980.

[4] T. Mirko and H. G. Lewis, "Re-entry prediction and demisability analysis for atmospheric disposal," Advances in Space research, vol. 68, p. 15, Oct. 2021.

[5] G. I. Barsoum, H. H. Ibrahim and M. A. Fawzy, "Static and Random Vibration Analyses of a University CubeSat Project," Journal of Physics: Conference series, p. 13, 2019.

[6] "European Cooperation for Space Standardization, Structural general requirements," ECSS Stnd., 2008.

[7] A. C. Okolie, S. O. Onuh, Y. T. Olatunbosun and M. S. Abolarin, "Design optimization of Pico-satellite frame for computational analysis and simulation," Am. J. Mech. Ind. Eng. 1, pp. 74-84, 2016.

[8] S. J. Kang and H. U. Oh, "On orbit thermal design and validation of 1 U standardized cubesat of step cube," International Journal of Aerospace engineering, pp. 1-17, 2016.

[9] J. Piattoni, G. P. Candini and G. Pezzi, "Plastic Cubesat: An innovative and low-cost way to perform applied space research and hands-on education," Acta Astronaut, p. 10, 2012.

[10] T. Lips, V. Watermann and G. Koppenwallner, "Comparison of ORSAT and SCARAB re-entry survival results.," in European Conf. on Space Debris, Darmstadt.

[11] X. He and X. Zuo, "Surrogate-based entire trajectory optimization for full space mission from launch to re-entry.," Acta Astronautica, vol. 190, p. 20, 2021.

[12] T. Mirko, H. G. Lewis and C. Camilla, "Spacecraft design optimisation for demise and survivability," Aerospace Science and Technology, p. 20, Oct. 2018.

[13] J. Beck and Holbrough, "Belstead Research (Online)," Jan 2015. (Online), Available: http://www.belstead.com.

[14] A. Tewari, Atmospheric and Space Flight Dynamics: Modeling and Simulation with MATLAB® and Simulink®, Birkhcauser, 2007.

[15] M. Trisolini and H. G. Lewis, "Demisability and survivability sensitivity to design-for-demise techniques.," Acta Astronautica, p. 28, Jan. 2018.

[16] E. A. Slejko, A. Gregorio and V. Lughi, "Material selection for a CubeSat structural bus complying with debris mitigation.," Advances in Space research, p. 9, Dec. 2020.

[17] R. L. Kelley and D. R. Jarkey, "CubeSat material limits for Design for Demise," Space Conferences and Exposition, p. 4, 2015.

[18] J. J. Valencia and P. N. Quested, "Thermophysical properties," ASM Handbook.

[19] MatWeb, "L. L. C. MatWeb," 2022.