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MODULAR DESIGN AND MODELING FOR QUALIFICATION OF SMALL SATELLITE DEPLOYERS

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Abstract: *Upcoming space missions depend essentially on the launch system development and its qualification. The paper presents a small satellite deployer benchmark and new trends in the domain. Integrators on the market only offer deploying services, and not much data is available regarding deployers' development. Cost savings and technology improvements may be achieved, but more research is still required. From this perspective the Cube-Sat's deployer design is a relevant example. The authors analyze the qualification requirements and related standards for the assessment of novel deployers applicable to the new generation of small satellites employed in upcoming space missions. The design of a new deployer and model preparation for FEA are described, and a preliminary dynamic verification is provided. Model validation methods are also presented. The gained knowledge from this study will be employed in future industrial projects.*

Key words: *small satellite deployer, modular design, model preparation, FEA, design qualification.*

1. INTRODUCTION

In recent years, the dimensions of small-class satellites have increased so that the standard launch systems developed in the years 2000–2010 no longer meet current requirements. "Classic" size satellites have been growing out of favor because the newly developed components and planned missions require even larger satellites as presented in Fig. 1.

This increased the market demand for different sizes and led to a growth in the need to develop an orbital launch system for these new satellites [2]. Moreover, since satellites are modular, the launch system should also follow the model of modularity to lower costs and increase usability.

Even though on the small satellite markets the deployer is designed and manufactured by major companies in the field, they do not provide access to design information and the corresponding development details. On the other hand, latest micro satellites already exceed current deployer's capabilities. The present research aims to fill this literature gap.

Usually, the process of launching satellites takes place following a typical procedure: the technical details of the satellite to be launched are presented to the integrator, who, depending on the satellite's characteristics, will program the launch process with other satellites or individually from a standard deployer.

The paper addresses the development of a new deployer concept and the qualifying requirements. The content is structured as follows. The next section describes the economic context and the topicality of the subject. Section 3 introduces the new modular concept of

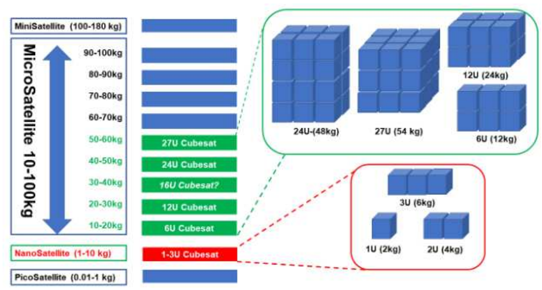

Fig. 1. Small-satellite sizes [1].

the deployer, taking into account the testing requirements. Section 4 compares more preprocessing strategies, while chapter 5 summarizes dynamic and thermal required assessments. Section 6 illustrates a comparative study of dynamic simulations and section 7 draws the conclusions and gives a glimpse of the future work.

2. SMALL SATELLITES AND MARKET DEVELOPMENT

2.1. Development of modular components

Due to the high number of developers for small satellites and satellite components, such as Cube-Sat parts, many modular constituents have emerged on the market. The components are easily integrated with high availability. The modules are accessible as off-the-shelf parts. Developed to a convenient size and fast integration capabilities within any project they represent an ideal solution for a rapid and low-cost mission development.

Some examples of components are presented in Figs. 2–4.

The Poly Picosatellite Orbital Deployer Mk. III Rev and standard have been employed in industry. The

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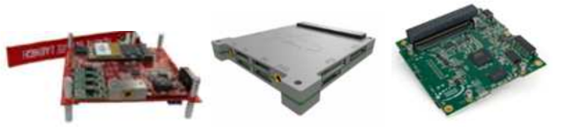
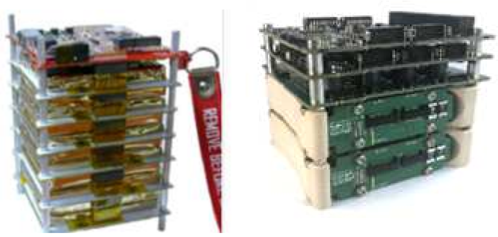


a b c

Fig. 2. Modular components for satellites: *a*– spacecraft system core; *b* – transmission computer; *c* – on board computer.

a b c

Fig. 3. Solar panels for satellites, different sizes: *a* – CubeSat solar panel; *b* – deployable solar array; *c* – CubeSat solar panels.



a b

Fig. 4. Modular energy storage systems: *a* – Battery Matrix; *b* – Modular electrical power system.

deployer has been updated until 2014 by California Polytechnic State University [3]. This document serves as a guideline for new deployers design. However, many companies have established their own models of orbital launchers, such as NanoRacks, ExoLaunch. These models have been employed since 2010 and have been provided only as a service for end-users interested to place new satellites in orbit. Though, as presented in Fig. 5, only 15 developers offer commercial deployers, out of which only 7 provide a maximum 12U size deployers.

	Type	Max Volume	Access Ports	Additional Mass	Extended Body Length	Extended Protrusions	Power/ Data Port	Purge	RF Insulation	Vibration Attenuation	X-Y Constraint	Z Constraint	-Z Additional Volume (Tuna Can)
PSL Astro-Fein	Rail	3U	X								X	X	
PLS-P Astro-Fein	Rail	12U	X									X	X
P-POD Cal Poly	Rail	3U	X					X	X	X		X	X
COSPOD COSATS	Rail	12U	X	X	X	X				X	X	X	X
EXOpod Exolaunch	Rail	12U	X	X		X					X	X	X
ISIPOD ISIS	Rail	3U			X	X					X	X	X
6-POD ISIS	Rail	6U			X	X					X	X	X
QuadPack ISIS	Rail	12U	X	X	X						X	X	X
E-SSOD JAXA	Rail	3U	X									X	
J-SSOD JAXA	Rail	3U, 6U	X	6U	X								
CSD Planetary Systems	Tab	3U, 6U, 12U	X		X		X				X	X	
RailPOD Tyvak	Rail	3U	X									X	X
NLAS Tyvak	Rail	6U	X	X	X	X	X	X	X			X	X
12U Dispenser Tyvak	Rail	12U	X	X	X	X	X	X	X		X	X	X
RAMI UARX	Rail	12U	X	X	X	X	X					X	X

Fig. 5. Deployers available on the market [3].

The space missions announced by NASA and ESA and the expansion of new modular components are increasing the need for larger deployers. Nowadays, even a 12U CubeSat is considered "small."

Currently, to plan and construct a space mission has become more accessible than ever. For example, one can employ the 24U Cube-Sats newly developed structures, occupy 20U with some of the components presented in Figs. 2–4 and organize them such that there is still free left space available so that a 2U camera can also be easily integrated.

3. DEVELOPMENT AND QUALIFICATION OF NEW DEPLOYERS

3.1. New deployer concept

The new modular small-satellite deployer addressed by the study is named M-ODS. It takes into account that the cost/unit to lift a kilogram of payload into orbit is decreasing, but it is still challenging. M-ODS concept attempts to solve the challenging requirement for launching very small satellites into orbit.

As stated before, existing designs are kept secret by the companies. The presented model proposes an open architecture and design and draws up development guidelines for this type of structures.

3.2. Testing requirements

The 2015 general ESA and NASA requirements are operational, but each mission and testing levels may vary in respect to the launch system and are dependent on the vehicle and mission. Therefore, all tests must be performed to comply with a particular launch provider.

If the launch vehicle is unknown at the time of testing, the General Environmental Verification Standard (GEVS, GSFC-STD-7000A) [4] and SMC-S-016 can be employed to drive testing requirements. However, the test levels indicated by these scientific sources do not guarantee to satisfy all the mission requirements.

Typical test requirements are:

- vibration testing – natural frequency analysis, sine vibration, random vibration (Power Spectrum Density – PSD);
- quasi-static testing - static load, shock load;
- thermal-vacuum testing – thermal analysis, thermal cycling, thermal-vacuum bakeout.

3.3. New M-ODS deployer concept

Structural components are main parts of space missions. Essentially, the purpose of the mechanical structure is to provide a simple and robust solution that will survive to the launch conditions and a suitable environment for all systems to function properly. In addition, the mechanical structure that supports all other spacecraft systems, attaches the satellite to the launch vehicle, and accommodates the activation of the separation system. The general objectives of the structural design are to allow simple loading paths, simplified interfaces, and easy integration [5].

For the development of the new modular structure, a basic 3U size was employed. The 3U model was created to define the basic size and to check the interferences, as presented in the Fig. 6.

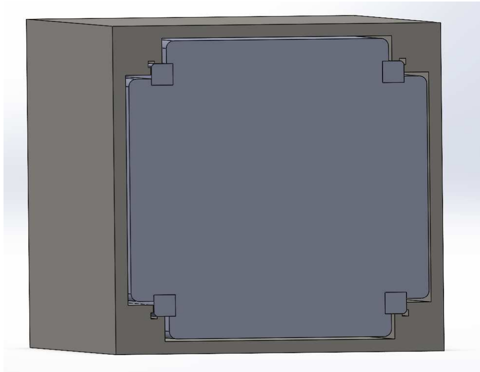


Fig. 6. Phase 1 of the deployer design.

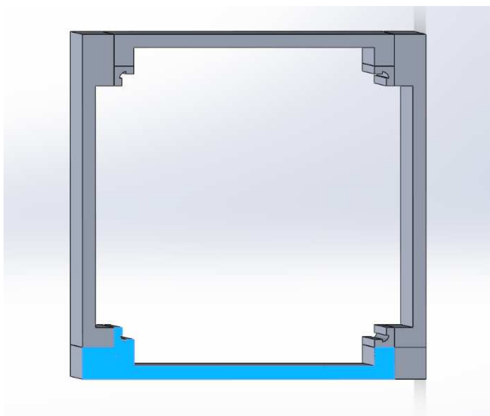


Fig. 7. Model split to improve manufacturability.

In order to be easily manufactured and to have a modular format, the CAD model has been split asymmetrically into the following shapes, as illustrated in Fig. 7.

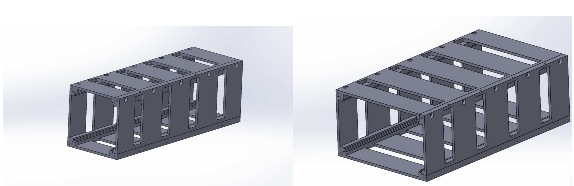
The deployer has to accommodate different size satellites. Therefore, a parametric model was developed with the following sizes, indicated in Table 1.

The full parametric design also includes smaller sizes, such as 1U, 6U, and other sizes (Figs. 9 and 10).

Table 1

Deployer modular design sizes

Dimension	3U	12U	24U	27U
Length [mm]	390	390	504	390
Width [mm]	100	200	200	300
Height [mm]	100	200	200	300
Payload [kg]	6	24	48	54

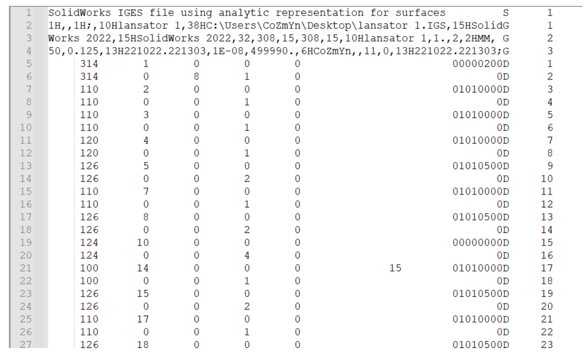


a b

Fig. 9. Parametric model: *a* – 3U model; *b* – 2012 6U model.

a b

Fig. 10. Parametric model: *a* – 12U model; *b* – 27U model.



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SolidWorks IGES file using analytic representation for surfaces       S      1
1H,,1H;,10Hlansator 1,38HC:\Users\CoZmYn\Desktop\lansator 1.IGS,15HSolidG      1
Works 2022,15HSolidWorks 2022,32,308,15,308,15,10Hlansator 1,1.,2,2HMM, G      2
50,0.125,13H221022.221303,1E-08,499990.,6HCoZmYn,,11,0,13H221022.221303;G      3
     314       1       0       0       0                        00000200D      1
     314       0       8       1       0                              0D      2
     110       2       0       0       0                        01010000D      3
     110       0       0       1       0                              0D      4
     110       3       0       0       0                        01010000D      5
     110       0       0       1       0                              0D      6
     120       4       0       0       0                        01010000D      7
     120       0       0       1       0                              0D      8
     126       5       0       0       0                        01010500D      9
     126       0       0       2       0                              0D     10
     110       7       0       0       0                        01010000D     11
     110       0       0       1       0                              0D     12
     126       8       0       0       0                        01010500D     13
     126       0       0       2       0                              0D     14
     124      10       0       0       0                        00000000D     15
     124       0       0       4       0                              0D     16
     100      14       0       0       0              15        01010000D     17
     100       0       0       1       0                              0D     18
     126      15       0       0       0                        01010500D     19
     126       0       0       2       0                              0D     20
     110      17       0       0       0                        01010000D     21
     110       0       0       1       0                              0D     22
     126      18       0       0       0                        01010500D     23
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Fig. 11. Details of IGES file viewed as text.

3.4. Model preparation for FEA analysis

Although the geometry looks simple, data transfer from the CAD system to the CAE software can be a problem if topology analysis and appropriate corrections are not made. The Initial Graphics Exchange Specification (IGES) is a neutral provider of text files that allow the digital exchange of information between computer-aided graphics software. It is based on an ASCII text format. Using IGES, a CAD user can exchange models in the form of circuit diagrams, wireframes, undefined surfaces, or solid models. Applications supported by IGES include traditional engineering drawing files, models for analysis as well as other manufacturing functions.

IGES products are composed of geometric and non-geometric entities; the geometric entities are lines, points, circles, etc. Non-geometric entities are annotations or structural entities that give the model the orientation of the faces and other essential details to assemble the original geometry (Fig. 11).

3.4.1. Topology clean-up. The preprocessing software automatically analyses the imported geometry and indicates common problems, such as: split edges, duplicate edges, and inaccurate faces. These types of import errors can be easily fixed (Fig. 12).

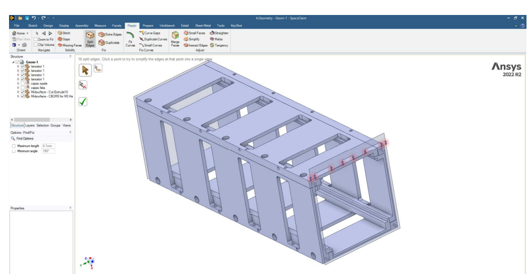


Fig. 12. Geometry problems after import.

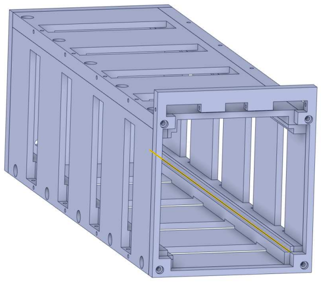


Fig. 13. Geometry defeaturing.

3.4.2 Geometry simplification. After all the geometry checks, the next step is to generate the computational model. In order to obtain a good mesh with no distortion the small radiuses and holes have to be deleted, as illustrated in Fig. 13.

4. MESH ANALYSIS

The use of Direct modeling design techniques helps the designer to create a more compact and lighter generation of launchers. Reducing the weight of the launch systems, and increasing their capabilities, leads to an increase in the mission capacity of the active payload. Before generating the mesh, it is necessary to ensure that the CAD model is prepared for the analysis. For this purpose, different design analysis tools are employed to identify the areas where the computational model can be improved. Finally, the model is composed of only the required features, so that the meshing process is simplified [6]. Using three different modeling methods, an analysis of the mesh is presented as follows.

4.1. Hexahedral mesh

The results of hexahedral meshing are presented in Fig. 14. This model is composed of 117.602 elements and 543.544 nodes.

The mesh has been analyzed using the skewness criterion. Most elements do not present any skewness on a scale from 0 to 1, 0 being an un-deformed element and 1 representing a deformed element. Approximately 45,000 elements, representing close to 40% of total elements, count a small deformation of 0.05.

4.2 Tetrahedral mesh

The results of tetrahedral mesh are presented in Fig. 15. This mesh is composed of the 109.313 elements and 206.706 nodes.

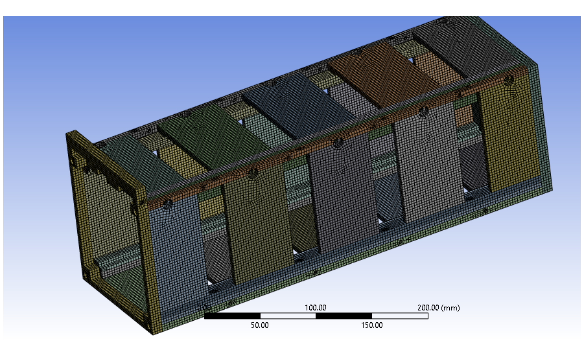


Fig. 14. Hexahedral mesh.

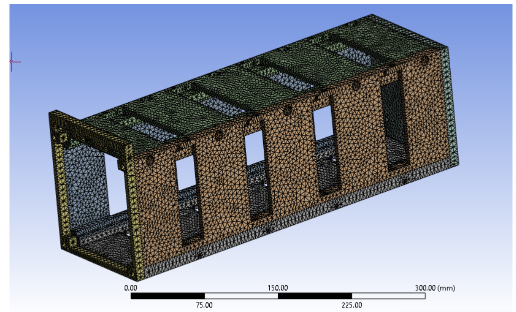


Fig. 15. Tetrahedral mesh.

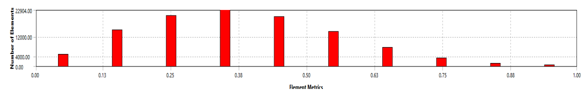


Fig. 16. Element quality analysis.

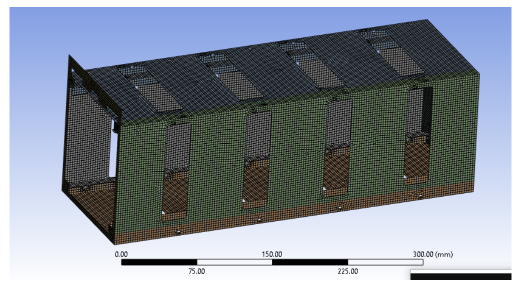


Fig. 17. Shell mesh.

Element distribution is illustrated in Fig. 16.

Elements quality, as presented in Fig. 16, show that the discrete model is composed of finite elements below 0.5 skewness factor, that count 84.000 elements, representing 75% of the model. A remaining number of elements show a skewness factor above 0.5.

4.3 Shell mesh

The shell mesh is presented in Fig 17. This model is composed of 39.526 elements and 37.267 nodes.

84% of elements present a skewness value of 0.04.

5. MODEL QUALIFICATION

As mentioned in section 3.2, for the qualification of the model, the following tests must be accomplished:

- quasi-static tests: 13g (g = 9.8 m/s^2), applied on each axis: X, Y, Z
- vibration tests comprise a modal analysis. The first natural frequency must be higher than 90 Hz.
- sine vibration: sweep rate, 2 oct/min, applied on each axis: X, Y, Z with the profile described in Table 2.
- random vibration: RMS acceleration of 8.03g, is applied on each axis: X, Y, Z with the profile described in Table 3.

Table 2

Test profile for sine vibration testing [4]

Frequency [Hz]	Amplitude [g]
5–100	2.5
100–125	1.25

Table 3

Test profile for random vibration testing [4]

Frequency [Hz]	Amplitude [g^2/Hz]
20	0.01125
130	0.05625
800	0.05625
2000	0.015

Table 4

Safety factors defined by ESA [6]

ECSS-E-ST-32-10_0110029

Load type	Requirements				Comments
	Vehicle		KQ	KA	
Global flight loads	Satellite		1,25 [a]	1	
	Launch vehicle		$1{,}25_{\text{corrected}}$ [b]	1 or J_P [c]	Typical value to be considered for dimensioning are J_P=1,05 to 1,1
	Man-rated S/C	Launch loads	1,4	1,2	
		On orbit loads	1,5		
Internal pressure	in conformance with ECSS–E-ST-32-02 [i]				Applicable for satellite and launch vehicles
Dynamic local loads [d]	Satellite		1,25 [a, e]	1	
	Launch vehicle		1,25 [e]	N/A	
Hoisting loads [f]	Satellite		2	N/A	
Hoisting loads [g] (fail safe)	Satellite		1	N/A	
Storage and transportation loads	Satellite -local transportation and storage loads -other transportation loads		2 1,4	N/A	
Thermal loads [h]	Satellite		1	1	
	Launch vehicle		1	1	

- thermal-vacuum test. The thermal cycle has the following characteristics: min temperature: −20 ±2 °C, max temperature: 50 ±2 °C, temperature variation rate: ≥1 °C/min, dwell time: one hour at extreme temperatures, cycles: 4;
- thermal Vacuum Bake Out: max temperature: 50 ±2 °C, temperature variation rate: ≥ 1 °C/min, vacuum: 10^{-5} mBar, duration: 3 hours after thermal stabilization.

5.1. Results evaluation

According to ESA, standard ECSS-E-ST-32-10C the following safety factors are defined in Table 4 [6].

For satellites and deployers, the recommended safety factor is 1.25. According to NASA standard -STD-7000B, the recommended safety factor for metallic structures is also 1.25, which is in accordance with ESA ECSS-E-ST-32-10C [7 and 8].

6. DYNAMIC ASSESSMENT

6.1. Modal analysis

The primary purpose is to determine the natural frequencies and modal shapes of the deployer. The results of these tests aim to check the resonances. The most important eigenvalue of the deployer is the first natural frequency and the corresponding vibration mode shape.

A comparison between the different mesh types was performed to determine the best computational model for all the dynamic qualification tests, taking into account two reasons: all the dynamic tests are based on the modal model and the computation time can be dramatically decreased by an appropriate mesh.

For the hexahedral mesh (Fig. 18), it was difficult to achieve a conforming mesh model, computational times were the highest, no simplifications were performed, and realistic condition constraints were defined. In the zones were cilyndrical shapes appear the mesh becomes distorted.

Tetrahedral mesh seems to offer the trade-off between the mesh generation methods because it was easy to obtain a qualitative good mesh (Fig. 19).

One of the main advantages of the shell mesh (Fig. 19) is the fast computational time, but there are some drawbacks, such as:

- the structural model was difficult to obtain, and many manual editing operations were required;
- limitation in conforming to the real problem in the 2D domain.

Even in the case of the shell mesh, elements can become distorted, as presented in Fig. 20.

Although the mode shapes are similar for all the three modeling techniques, the natural frequencies values differ within a margin of −12% up to 17% (Fig. 21). While the shell model is considered the conservative one, low differences can be observed between the two 3D models. The 3D models provide higher dynamic stiffness.

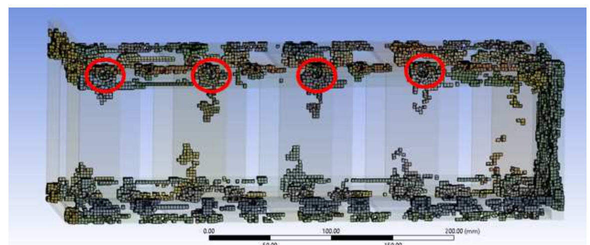


Fig. 18. Hexahedral mesh.

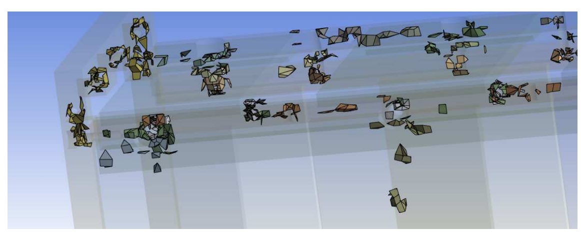
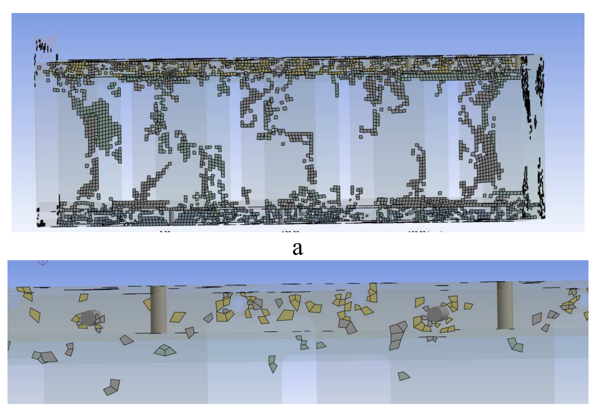


Fig. 19. Tetrahedral mesh.



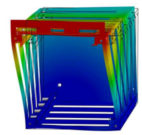
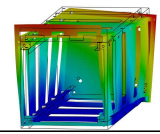
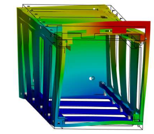
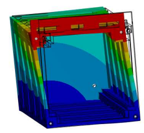
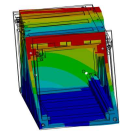
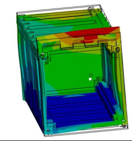
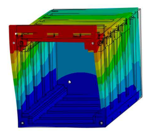
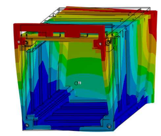
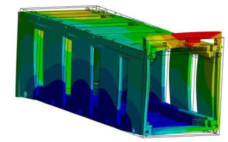
a

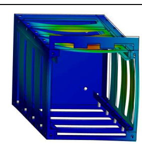
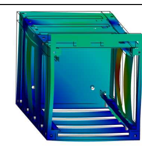
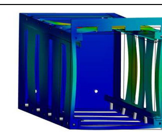
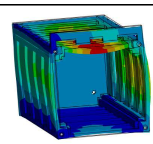
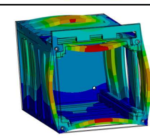
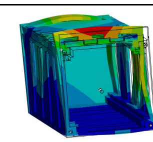
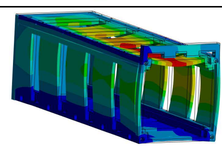
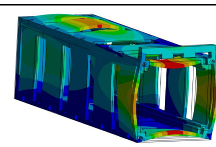
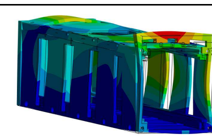
b

Fig. 20. Deployer shell mesh: *a* – shell mesh; *b* – detail.

Table 5

Mode shapes for the three studied models

	Mode 1	Mode 2	Mode 3
Shell			
Tetrahedral			
Hexahedral			

	Mode 4	Mode 5	Mode 6
Shell			
Tetrahedral			
Hexahedral			

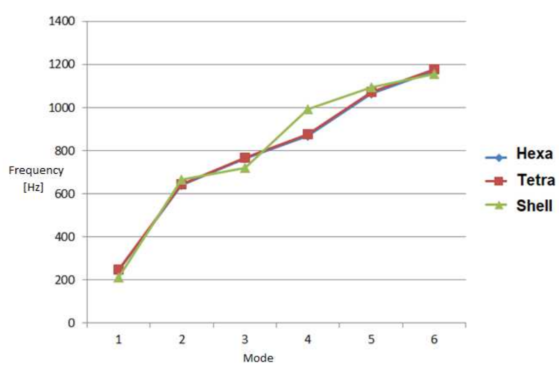


Fig. 21. Comparison between the resonances of the three computational models.

As expected, all the models proved that the first resonant frequency is higher than 90 Hz.

6.2. Quasistatic analysis

The quasi-static analysis has been accomplished for the three models with 13g accelerations applied on all axes. The results are summarized in Table 6. The maximum displacement is 0.101 mm and the maximum von Mises stress is 19.42 MPa for the shell model (Figs. 22 and 23).

Table 6

Displacement and stress comparison

Model	Displacements [mm]	Maximum equivalent stress [MPa]
Hexa	0.072	31.27
Tetra	0.071	24.21
Shell	0.101	19.44

In the case of the quasistatic analysis the shell model provides 30% higher values in respect to the displacements, but minimum stress concentrators. This is due to the more accurate modeling of the restrained areas around the fixed supports.

6.3. Sine vibration

The harmonic analysis has been completed for a frequency domain of 30–2400 Hz to encompass the first 10 resonances. The frequency response for the three models is summarized in Table 7.

The harmonic response proved that the highest amplitude occurs at the dominant natural frequency, as illustrated in Table 7. Details of the maximum amplitudes are given in Table 8.

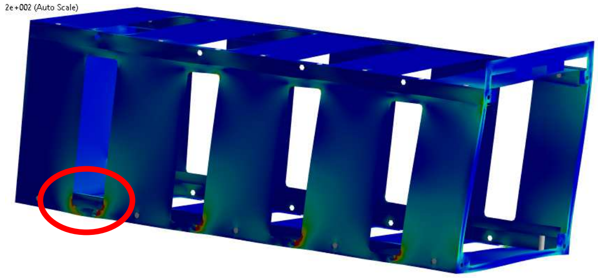


Fig. 22. Maximum equivalent von Mises Stress.

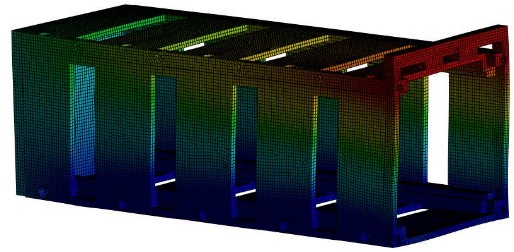


Fig. 23. Maximum deformation.

Table 7

Frequency response analysis

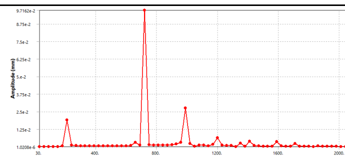
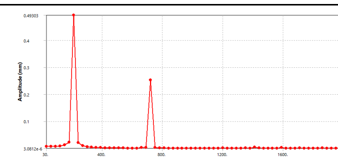
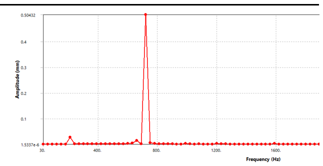
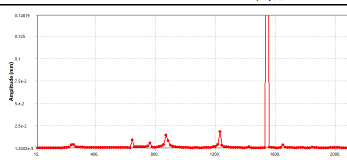
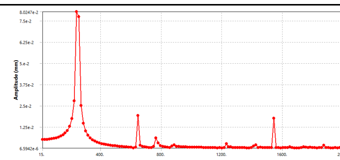
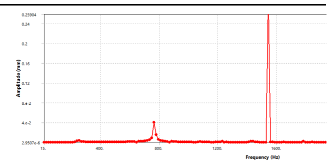
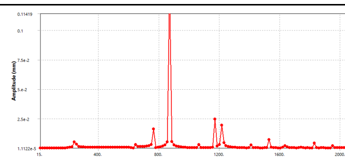
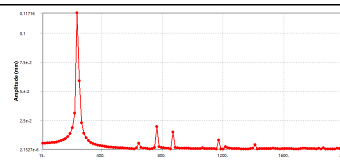
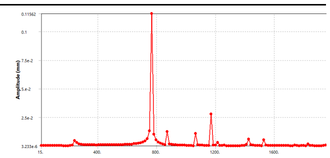
	X axis	*Y* axis	*Z* axis
Shell			
Tetra			
Hexa			

Table 8

Comparison of the maximum amplitudes.

Model	Frequency [Hz]	Amplitude *X* [mm]	Amplitude *Y* [mm]	Amplitude *Z* [mm]
Shell	210	0.019	0.493	0.027
	720	0.097	0.252	0.504
	990	0.027	0.001	0.002
Tetra	240	0.003	0.080	0.002
	645	0.008	0.019	0.001
	870	0.014	0.001	0.001
Hexa	240	0.005	0.117	0.004
	645	0.003	0.004	0.002
	870	0.135	0.014	0.012

As expected, the shell model is more flexible than the 3D ones, recording frequencies about 15% lower, but higher amplitudes in all directions. The 3D models are nearly similar, both in terms of frequency values and the amplitude levels.

Dynamic simulation results proved that the design of M-ODS deployer is in accordance with qualification standard requirements no matter the modeling technique is addressed.

7. CONCLUSIONS

Model preparation is a crucial step in product development. A tuned computational model with the required accuracy and small computation times can bring significant advantages for the designer.

Simulation results proved that significant differences may occur between various numerical models, therefore the simulation results must be checked by means of laboratory experiments.

There is little data available for orbital deployer's projects. The M-ODS concept has been developed as an open project, to allow the design and validation process for other actors on the market or future industrial applications. The present modular design is not destined only for Cube-Sat class satellite types, but can also accommodate uncommon satellite sizes and variants.

Although the general FEM theory recommends the use of reduced order models, experiments have to certify the most appropriate numerical model, especially in small satellite industry where all the standard requirements have to be precisely met.

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