Mission Capability Assessment of 3D Printing Cubesats

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Abstract: With the successful development of integrated technologies, many spacecraft subsystems have been continuously miniaturized, and CubeSats have gradually become the main executors of space science exploration missions. It is a new challenge to combine the CubeSat design with 3D printing. Compared with traditional manufacturing through machining, 3D printing technology has several advantages including short period of manufacturing, high accuracy in manufacturing small parts and low cost. The research task of this paper is a LEO (Low Earth Orbit) CubeSat mission, with a maximum acceleration of 5 g during the launch process, the internal operating temperature range of the CubeSat is from 0 to 40 °C, external temperature from -80 to 100 °C. The environmental factors were fully considered in the CubeSat design process, the impact load received during the CubeSat launch process and the working environment after reaching orbit were simulated and analyzed by ANSYS after the design parameters were obtained to verify the feasibility of the design.

Key words: CubeSat, 3D printing, ANSYS

1. Introduction
This paper introduces the design and manufacture of CubeSat platform based on 3D printing technology. The great potential in space exploration of CubeSat can be seen by the recent explosion of micro-satellite missions. Such as the biological science micro-satellite Biosentinel [1], an 3U experimental CubeSat STU-2 [2], and night vision observation satellite Athenoxat-1 [3]. Short research cycle and low cost are the main advantages of the CubeSat, however, the CubeSat structure usually made by traditional mechanical processing methods. Such processing methods are costly and require a long cycle, which is against the CubeSat’s design intention. With the development of manufacturing technology, 3D printing technology gradually replaced many traditional processing methods. A 3D printing CubeSat platform realized within the activities of the Space Robotics Laboratory and V-Lab of the II Faculty of Engineering of Bologna University in collaboration with the Aerospace System Laboratory of the University of Rome “La Sapienza” [4]. Using rapid prototyping technology for manufacturing has several advantages including high fast implementation, accuracy in manufacturing small parts and low cost. In addition, this technology is very useful for the manufacture of small satellites, because the accuracy of the details can be achieved, which is quite difficult and expensive by using traditional technology.

The main content of this research is a satellite in the shape of a cube with side length of 10 cm. Each cube cell is called a unit, weighs about 1kg, and the CubeSat is named based on the number of units (2U, 3U, ...). The CubeSat is made by 3D printing technology, using PLA as the material. The research is based on the LEO mission, with a maximum acceleration of 5 g during the launch process, the internal operating temperature range of the CubeSat is from 0 to 40 °C, external temperature from
-80 to 100 °C. These factors should be fully considered during the CubeSat design process, using ANSYS to do the environmental simulation and analysis to verify the feasibility of the design.

2. Methodology

2.1 Structure

The framework structure for a single CubeSat provides enough internal workspace for the hardware required to run the CubeSat. Although there are various CubeSat structure designs, several consistent design guidelines can be found by comparing these CubeSats.

The CubeSat frame is usually designed as a cube with a side length of 100 mm, square columns (8.5 mm side and 113.5 mm height), placed at four parallel corners. The CubeSat is usually made of aluminum, because it has characteristics such as ease of machining, high-strength-to-weight ratio, lightweight and low cost [5]. In this paper, the 3D printing technology is combined with the structural framework design, and the applicability of the technology in the final product manufacturing has been discussed. PLA (polylactic acid) is selected as the material. This new material has the characteristics of lightweight and high-strength-to-weight ratio, and has the advantages of rapid realization, high accuracy and low cost of 3D printing technology [6].

Compared with traditional metal processing technology, the design scheme combining PLA and 3D printing technology is much more advanced. Traditional manufacturing techniques typically remove materials, while 3D printing typically adds them layer by layer. The technique is also known as FDM (Fused Deposition Modeling), in which the material is fused into a very thin wire and deposited in layers. The adoption of this technology changes the concept of primary and secondary structure in the traditional design process, because the whole structure can be produced at the same time, which not only reduces the number of parts, reduces the need for screws and adhesion, but also improves the stability of the overall structure. One disadvantage of this technique is that the material is not isotropic. On the other hand, the direction of material deposition needs to be optimized to be consistent with the priority direction of maximum load. After the completion of production, the products also need to be processed to achieve the desired effect, such as surface grinding [7].

2.2 Power subsystem

CubeSat runs on two main sources of power: secondary batteries and solar panels. The architecture of the power subsystem is very simple. The solar panel is connected directly to the battery through a blocking diode. The power subsystem of the CubeSat can be customized according to the operational requirements of each subsystem, including propulsion, control and data handling (CDH), attitude determination and control system (ADCS), payload, etc. [8]. The input power required by the subsystem can be determined by the next equation:

\[ P_{req} = P_{in} / \eta_{pwr}, \]

where \( P_{req} \) is the input power of the subsystem, \( \eta_{pwr} \) is the utilization efficiency. This equation gives the required power for all subsystems, the sum of these is the total power \( P_{tot} \) required by the CubeSat. This total power then needs to be distributed to each electrical component by the distribution unit (PDU), which is powered by a power processing unit (PPU) connected to a high-voltage electronics assembly (HVEA). The power required by PDU can be determined by equation:

\[ P_{PDU} = P_{tot} / \eta_{PDU}, \]

where \( P_{PDU} \) is the total power required by PDU and \( \eta_{PDU} \) is its efficiency. The power required by PPU and HEVA can also be calculated according to the respective efficiency of components in a similar way. It should be noted that the correct power budget should be achieved in a certain order (PDU, PPU, and then HEVA). The mass of wiring and related constraints usually accounts for 10-25% of the whole
power subsystem; wiring efficiency (about 95%) is also traditionally used to estimate the power required for a PPU. It is also common to add a power margin of 10-20% to the power budget \[9\]. The resulting value is the amount of power the solar array needs to provide.

![Solar Cell Efficiency](image)

Figure 1. Solar Cell Efficiency \[5\].

Suitable solar cell arrays must be used to provide the required power for the CubeSat. To guarantee the power supply during the CubeSat’s mission cycle, the state of the solar array needs to be considered not only at the beginning-of-life (BOL), but also at the end-of-life (EOL) phase. The degradation coefficient \(D\) is introduced here, which can be defined by equation:

\[
D = (1 - D_a)^L, \tag{3}
\]

where \(D_a\) is the approximate degradation rate of the solar array (%), and \(L\) is the duration of the mission (year). Figure 1 shows the typical photovoltaic solar cells, they have an efficiency range of 12-25% at the beginning-of-life (BOL), and there are a few new commercial photovoltaic solar cells designs that can achieve up to 30% efficiency \[5\]. Therefore, the efficiency of these solar cells at the end-of-life (EOL) can be calculated by equation:

\[
\eta_{EOL} = \eta_{BOL}D, \tag{4}
\]

where \(\eta_{BOL}\) and \(\eta_{EOL}\) respectively represent the efficiency of the solar cell array at the beginning-of-life (BOL) and at the end-of-life (EOL). Other parameters of the solar cell array, such as power density \(P\) and specific power \(\rho\), they can be calculated by equation (4), and only need to be replaced \(\eta_{EOL}\) with \(P_{EOL}\) or \(\rho_{EOL}\), and at the same time replaced \(\eta_{BOL}\) with \(P_{BOL}\) or \(\rho_{BOL}\). From these parameters, we can estimate the mass \(m_{SA}\) and expansion area \(A_{SA}\) of the solar cell array:

\[
m_{SA} = \frac{P_{tot}}{\rho_{EOL}}, \tag{5}
\]

\[
A_{SA} = \frac{P_{tot}}{I_{s.min}\eta_{EOL}\cos(\theta_s)}, \tag{6}
\]

where \(\theta_s\) is the incidence angle of sunlight, \(I_{s.min}\) is the local minimum solar irradiation intensity. If the solar array is a mobile deployment system, with gimbal and extra support bracket, etc., additional quality budget needs to be considered. The power available at the beginning-of-life (BOL) and at the end-of-life (EOL) of a solar array can be calculated by next equation:

\[
P_{SA} = A_{SA}\eta I_{s.min}\cos(\theta_s) \tag{7}
\]
For CubeSat, it is vital to keep the system running when it is not exposed to sunlight, and this part of power is provided by the battery. The mass of the battery can be estimated by equation:

\[ m_b = \frac{P_{b,\text{req}} t_b}{\rho_{b} \eta_{b} d}, \]  

where \( P_{b,\text{req}} \) is the power the battery needs to provide, \( t_b \) is the time for the battery to provide power (hours), \( \rho_{b} \) is the specific power of the battery, \( \eta_{b} \) is the utilization efficiency of the power provided by the battery, and \( d \) represents the discharge depth of the battery. The charging rate of the battery is considered to ensure that the CubeSat can fully charge the battery during each orbit day. The charging time of the battery can be calculated by next equation:

\[ t_c = \frac{P_{b,\text{req}} t_b}{V_{b} \eta_{b} d C_r}, \]  

where \( V_{b} \) is the battery output voltage and \( C_r \) is the charge rate of the batteries for a given charger.

2.3 Other Subsystems

Although not the primary focus of this research, a complete CubeSat design requires thermal subsystem, the communication system, ADCS, and CDH subsystems. Due to the high surface area-to-volume ratio, the thermal system design requirements of the CubeSat should be the highest of all subsystems during the design process. The normal operating temperature range of internal components is from 0 to 40 °C, and the solar cell array can run between -80 and 100 °C \([9]\). Specific temperature depends on the different system of survival, but usually only extending ±10 °C from the working temperature range. The primary function of the communication system is to provide the signal connections to ground stations on the earth. The CubeSat consists of the satellite’s onboard antennas, radios and data circuit boards. In addition, to ensure that the solar arrays point accurately to the sun during orbit, a three-axis stability system is required, the power requirements and mass of the electronic package must also be considered. The CDH subsystem acts as the CubeSat’s brain, receiving telemetry information and controlling the CubeSat’s actions when no human operator is present. Finally, a comprehensive set of autonomous navigation software is also a necessary condition for the successful operation of spacecraft.

3. Parametric results for CubeSat

3.1 CubeSat mission Overview

The CubeSat under development will be able to test the effectiveness of structures manufactured using rapid prototyping technology in an orbital environment and will be able to perform performance tests on the camera payload to evaluate its reliability and endurance. This camera is a commercial camera, using COTS optical devices, which can adapt to different observation tasks by changing the system’s field of view and resolution.

3.2 Structure

Depending on the design specification and the material selected, the CubeSat space frame usually has a mass of no more than 200 g. To meet the related design requirements, such as the installation of related subsystems, additional braces and brackets may be required, the quality varies from 5-50 g. The mounting of the brackets completes the frame structure and provides CubeSat hardware and payload mounting points \([10]\).

The outer panel is usually made of metal or fiberglass for the circuit board, with a mass of about 50 g per 10x10 cm of coverage area. These panels are mounted on the CubeSat frame surface to shield the internal hardware and provide mounting points for the expandable solar array as fixpoints.

Except the design considerations mentioned above, during the structural design process, 15-25 %
mass allowance should be reserved for components that may need to be installed.

3.3 Power subsystem

The mass of the gimbal and mounting hardware for the four solar arrays is approximately 40% of the total mass of the solar array, resulting in a mass of about 60 g for each array and a power demand of about 2 W. The CubeSat uses Spectrolab UTJ (ultra-triple junction solar cells), it can provide the power density of more than 380 W/m², 450 W/kg of specific power and efficiency of about 28% at BOL [11]. A battery performance degradation of 7.8%, which included the assembly, design, and the storage-time losses, was assumed to happen before launch with an annual performance degradation of 0.5% during the task [11]. The view factor to the sun was approximated as 0.9, and assuming that the solar cell packing fraction is 0.9 [12], the average solar incidence angle was assumed as 30 deg (a conservative estimate), these values can identify the solar array area of 0.078 m², the total quality of solar array is about 124 g. In addition, CubeSat’s PPU has a mass of about 50 g with an efficiency of 92%, while PDU has a mass of about 35 g with an efficiency of 85%. These values are very conservative estimates for the power subsystem. The mass of HVEA is approximately 45 g, and the efficiency is 97%. These values refer to the mass quoted by the manufacturer of spacecraft electronics [5]. Table 1 shows the mass equipment list (MEL) of the CubeSat.

Table 1. MEL for the 1U CubeSat

| Subsystem                  | Component                                               | Mass, kg | No. |
|---------------------------|---------------------------------------------------------|----------|-----|
| Structure                 | Frame                                                   | 0.200    | 1   |
|                           | Brackets                                                | 0.005    | 8   |
|                           | Braces                                                  | 0.020    | 4   |
|                           | 10×10cm exterior panel                                  | 0.050    | 6   |
|                           | Solar array gimbal and mounting system                  | 0.024    | 4   |
| Power                     | Solar arrays                                            | 0.035    | 4   |
|                           | Battery                                                 | 0.178    | 1   |
|                           | Battery charger                                          | 0.050    | 1   |
|                           | PPU                                                     | 0.050    | 1   |
|                           | PDU                                                     | 0.035    | 1   |
|                           | HVEA                                                    | 0.050    | 1   |
| CDH                       | CDH board                                               | 0.050    | 1   |
|                           | EPS board                                               | 0.100    | 1   |
| Bus Dry Mass              | Payload (assumed)                                       | 0.350    | 1   |
| LEO mission wet mass      | Payload (assumed)                                       | 1.719    |     |

4. Simulation

Due to the unique nature of the space environment, CubeSat must perform a series of simulation tests before it can operate in the orbit. The CubeSat structure was analyzed by means of finite element analysis software ANSYS. In addition, random vibration analysis was performed to ensure that the CubeSat structure can withstand the impact load during the launch process. The feasibility of the CubeSat structure is verified by these results.

4.1 Static Structural
The CubeSat structure is validated by the numerical experiment. During launch process, CubeSat will be fixed inside the P-Pod, and the corresponding structural constraints should be added to the numerical model. In addition, the maximum acceleration impact during the launch process should also be considered. Static Structural module of ANSYS is used for calculation and analysis, the results show that the maximum stress of CubeSat Structure is 8.06 MPa, lower than the PLA yield strength of 40 MPa. Figure 2 is the Von mises stress diagram of the CubeSat structure.

Figure 2. The Von mises stress diagram of the CubeSat structure.

4.2 Thermal stress
Temperature difference is a major characteristic of the space environment, CubeSat running around LEO will go through a temperature change of 100 °C. The simulation model of steady state temperature through numerical analysis software, set the conditions for internal temperature range from 0 to 40 °C, external temperature range from -80 to 100 °C. CubeSat structure should be able to resist a sudden change of temperature (circulating temperature between 20 to 60 °C), so the thermal shock test is also needed. According to the experimental results, the maximum value of Thermal strain is only 0.009, lower than 0.01, which is acceptable. Figure 3 is the thermal strain diagram of the structure.

The effects of radiation on PLA material can be neglected [13], the PLA material can withstand years of total dose of radiation to $10^9$ rad (SI), and the absorbed radiation dose of CubeSat run a year is far lower than this value in LEO (less than $10^7$ rad, height 2000 kilometers [14]). As for the direct sunlight, it should be considered that the structure of the PLA material is almost completely covered by the solar panels.
4.3 Random Vibration

To make the CubeSat structure conform to the emission conditions, random vibration simulation experiments are required. During the experiment, typical vibration characteristics of launch environment were simulated. NASA GEV qualification and acceptance were used as reference, and the specific parameters are shown in Table 2. The specific contents of the experiment include “Harmonic Response” and “Random Vibration”. Two identical harmonic response were performed before and after the random vibration test to assess the degree of structural degradation that may result from the launch load.

Table 2. NASA GEV qualification and acceptance [5].

| Frequency (Hz) | Qualification | Acceptance |
|---------------|---------------|------------|
|               | ASD (g²/HZ)   | ASD (g²/HZ)|
| 20            | .026          | .013       |
| 20-80         | +6 dB/oct     | +6 dB/oct  |
| 80-500        | .16           | .08        |
| 500-2000      | -6 dB/oct     | -6 dB/oct  |
| 2000          | .026          | .013       |
| Overall       | 14.1 Grms     | 10.0 Grms  |

Figure 4 shows the response curve of the two Harmonic response tests. This experiment helps us to evaluate the natural frequency of the structure, and the peak value indicates that the tested point (bottom panel) has reached the resonant frequency. As shown in this figure, the trend and peak points of the two response curves are close to each other, indicating that the structure has no structural degradation after the random vibration test, and the overall CubeSat structure conforms to the stiffness specifications of launch.
Figure 4. Pre/Post Random Vibration test comparison between the curves of Harmonic Response.

5. Conclusion
As the primary performer of today’s space exploration missions, the CubeSat design considers orbit, payload, thermal balance, subsystem layout, and mission requirements. In this research, a CubeSat design for performing LEO tasks was proposed, including power budget, mass distribution, and ground testing, and the CubeSat structure for manufacturing was combined with 3D printing technology. The feasibility of the new rapid prototyping technology in CubeSat manufacturing was evaluated by simulation tests. The results show that the CubeSat can withstand the launch loads without structural damage and can meet the launch stiffness specification.

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