Right-sizing Small Satellites

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ABSTRACT

Spacecraft standardization has been a topic of great debate within the space community. This paper intends to be a provocative thought piece asking one fundamental question: "is there a 'right size' for small satellites?" In order to answer this question, we propose three top-down design factors for the space systems engineering process: spacecraft utility, mission utility, and optimum cost. Spacecraft utility quantitatively measures the capability of a spacecraft, derived from its volume and power properties. Mission utility then measures the aggregate value of a constellation. Optimum cost, which is a function of spacecraft mass and quantity, can be determined by assessing the break-even point. Data from the small satellite community, including USAF Academy FalconSAT and Surrey Satellite Technology Ltd. (SSTL) missions, is presented in support of this discussion, constrained to systems with a mass less than 200 kg. These design factors inform the mission developer in determining the appropriate system architecture. Using these design factors, a notional standardized spacecraft configuration is presented, with a mass of 30 kg and 50 cm cubed volume that optimizes spacecraft utility, mission utility, and cost.

1. INTRODUCTION

Spacecraft standardization has been a topic of great debate within the space community. Proposals such as CubeSat[1], U.S. DoD's Space Test Program's Standard Interface Vehicle[2], and Plug-and-Play Satellite[3] are examples of proposed spacecraft standards, with varying degrees of adoption. Many have attempted to constrain the design space of a new mission to one of these standards to containerize the system to target lower launch costs; leverage a common spacecraft bus design; and modular designs, respectively. However, this bottom-up space systems engineering approach can result in a less than optimal design for most missions.

This paper engages the debate on spacecraft standards by starting with a fundamental question: "is there a 'right size' for small satellites?" To answer this question, we propose three new top-down figures of merit for evaluating a small spacecraft design: *spacecraft utility*, *mission utility*, and *optimum cost*. These concerns span government, academia, and commercial interests. Understanding the answer to the 'right size' question, and how these figures of merit can be applied, will help the community take pause and grasp to better assess these issues before pursuing any further standardization or, perhaps, to develop a better approach to evolve current or develop future standards.

Spacecraft utility is a proposed quantitative figure of merit, where maximum spacecraft utility is a normalized result derived from total spacecraft volume, payload volume, and power. These factors greatly influence the potential capability of the hosted payload, and hence the capability of the spacecraft as a whole.

Mission utility describes the aggregate value of a constellation of small satellites. For example, it is obvious that five cooperative remote sensing satellites provide more utility than one. However, it has not been obvious if very small satellites, such as satellite-on-a-chip, can become valuable in a large constellation.

Optimum cost is then determined by the system and mission level configurations, looking mainly at the break-even point. When determining cost, launch vehicle integration (LVI) techniques and costs are considered. Containerization is an enabler up to a certain spacecraft volume.

Data from the small satellite community, including USAF Academy FalconSAT and Surrey Satellite Technology Ltd. (SSTL) missions, is presented in support of this discussion. With a focus on determining an optimum spacecraft configuration with an individual spacecraft mass less than 200 kg, a proposed design is presented.

2. BACKGROUND

This paper specifically focuses on the concerns of small satellites with a mass less than 200 kg. Specifically, three topics outlined in the introduction are explored further in this background section:

- Lowering launch costs through containerization
- Rapid development with standardized bus designs
- Modular bus and payload design

Lowering Launch Costs Through Containerization

CubeSats were originally envisioned to enable budget-constrained entities, such as academic organizations, to build and launch satellites for very low cost[1]. One of the key enablers of the CubeSat success story is the adoption of a containerized approach to LVI, namely the P-POD[4]. Containerizing small satellites was inspired by the modern sea-faring shipping container, which had first use in 1956, but was inspired by British cargo movement systems created in 1792[5].

The P-POD, first used in 2003, was not the first envisaged satellite container and deployment system. The Orbiting Picosatellite Orbiting Launcher (OPAL), first used in 2000, paved the way for the CubeSat community[6]. OPAL was inspired by NASA's various deployment systems for the Space Shuttle, such as the Payload Ejection System (PES)[7].

The motivation for containerization of spacecraft is to enforce a mass properties standard that allows for rapid LVI, as long as the user complies with the standard. Containerization also reduces launch program risk, making rideshares quite common at present and soon may be ubiquitous.

The original objective of the CubeSat concept was to give low-budget research programs affordable access to space. Unit costs for CubeSat launches have ranged from $40,000 in the mid-2000s for a 1U system (1 kg, 10x10x10 cm)[8] to nearly $85,000 in present day costs through providers such as Nanoracks[9]. Nanoracks deploys small spacecraft from the International Space Station and also offers a 3U deployment cost of $220,000.

This approach has made it possible to execute a space mission for under $200,000. This low-cost method has been attractive to academia, government, and industry circles. However, to leverage this low-cost access, mission sponsors were constrained to the container standards, which then forced spacecraft designers miniaturize payloads to fit 1U-3U CubeSats. However, this approach is debatable in the broader space systems engineering context.

Rapid Development with Standardized Bus Designs

The DoD's Space Test Program proposed the Standardized Interface Vehicle (STP-SIV) in the mid-2000s[2]. It is a standard bus configuration that conforms to the EELV Secondary Payload Adapter standard (ESPA). The ESPA small spacecraft standard specifies a 181 kg maximum mass (400 lbs) and volume of 60.9 x 71.7 x 96.5 cm (24 x 28 x 38 in). There are also other requirements, such as center of gravity and moment of inertia. The SIV has a standard set of spacecraft subsystems to perform a variety of missions.

One of the key features of the SIV is standardized payload accommodation. The mass is limited to 60 kg and a volume of 45.0 x 40.4 x 66.8 cm. The power is limited to 100 W orbital average power (OAP). The payload volume is approximately 29% of the ESPA standard volume. A commercialized version of the SIV is offered by Ball Aerospace: the BCP-100.[10]

In the late 2000s, the National Reconnaissance Office (NRO) inaugurated a standardized CubeSat bus development program called Colony. Pumpkin, Inc. was the bus contractor for the Colony I bus (known as MISC-2), which was used on the Naval Research Laboratory's QbX mission. The Colony I bus offered 1.5U of payload volume (~40% total bus volume when considering the sidewalls, etc.) and deployable solar arrays offered a maximum of 8 W OAP[11]. The Colony I program demonstrated in 2010 that a standardized CubeSat bus could be successful on orbit.

The NRO went on to develop Colony II, where Boeing was selected as the bus contractor. The first Colony II buses were used on the SENSE mission sponsored by the Space and Missile Systems Center. The 3U Colony II bus provides 1.5U payload volume with an improved 10 W OAP[12]. The publicized cost goal per bus is $250,000 each[13]. The SENSE mission pair was launched in November 2013 and were still functional at the time of publication (August 2014).

Modular Bus and Payload Design

The Plug-and-Play satellite concept was launched by the Air Force Research Laboratory (AFRL) Space Vehicles Directorate. The goal is to facilitate satellite assembly in a few days by leveraging modular components defined by open standards and interfaces, which can automatically configure once connected[14]. A prototype spacecraft called PnPSat-1 was developed as a proof of principle. The spacecraft volume was 51 x 51 x 61 cm with a mass of 181 kg, which is on the same order as an ESPA-class vehicle. Although the satellite has not yet flown, the approach proposes some common sense ways of simplifying spacecraft integration.

3. SPACECRAFT UTILITY

Spacecraft utility is a proposed quantitative figure of merit based on the overall volume, payload volume, and OAP of the spacecraft. The volume of a spacecraft directly influences the available payload volume and surface area for solar power collection. Obviously, as a spacecraft grows in size, it can provide more capability to the hosted payload. This discussion proposes a framework by which the utility of one spacecraft configuration can be compared to another.

Spacecraft volume and power generation were found to be the key limiters as spacecraft size diminishes, albeit obvious[15]. However, spacecraft miniaturization continues to thrive, as evidenced by the PocketQube movement and recent workshops. PocketQube is an emerging spacecraft standard where each spacecraft can be as small as one-eighth U (1/8U), measuring 5 x 5 x 5 cm with a mass of 125 grams. $50Sat is an example of a successful PocketQube[16]. A similar concept was proposed by the University of Surrey a few years earlier, as a part of a comprehensive study of spacecraft miniaturization technologies[17].

To further explore the concept of utility, a notional list of spacecraft missions is shown, each one arguably requiring successively larger spacecraft:

- Simple demonstration beacon
- In-situ space weather/radiation monitoring
- Low-resolution visible imagery
- LEO communication systems (Iridium, etc.)
- Global Navigation Satellite Systems (GNSS)
- Infrared imagery
- High-resolution visible imagery
- GEO communication systems/relays

But this definition of spacecraft utility does not address the value of multiple satellites to perform a mission. The next section discusses this aspect, using two different mission examples that require a constellation.

What this paper is first attempting to tackle from a spacecraft utility standpoint is how to determine the "right size" for a particular set of mission requirements. A concerning trend in the small satellite industry is our fixation on a particular small satellite standard, namely the 3U CubeSat. While the 3U CubeSat is an excellent choice when the payload can readily "fit" without modification, the mission costs can skyrocket when a payload is purposefully miniaturized to fit a 3U CubeSat. The primary author has personally witnessed several payload development programs with this aim. The result, in every case, was program failure. These efforts to force-fit high functioning payloads within a 1.5U payload space proved to be cost-prohibitive.

The Theoretically Perfect Satellite

To quantitatively determine the utility of a spacecraft, the theoretically perfect satellite is proposed as follows: the payload consumes 100% of the spacecraft volume and power (i.e. bus volume and power negligible), power available is infinite, and total volume is infinite. Although mass is considered later in this paper, it would be zero. Therefore, spacecraft utility (*ScU*) is an asymptotic value that approaches unity defined by the following mathematical model:

$$ScU = \eta\left(\frac{P}{P+100}\right)\left(\frac{V}{V+1}\right) \tag{1}$$

where η = lumped payload volume and power efficiency (fraction of spacecraft dedicated to the payload, where 1 = ideal); P = OAP in Watts (∞ = ideal); and V = spacecraft volume in m^3 (∞ = ideal). The weighting factors have been initially chosen where 100 Watts carries the same weight at 1 m^3.

Table 1 below summaries a wide range of *ScU* values for spacecraft, both conceptual and actual, ranging from a few grams to nearly 200 kg. The results are straightforward, demonstrating that larger volumes with more power approach an *ScU* value limited by η, which is expected. Therefore, spacecraft designers should focus on reducing spacecraft bus "overhead" volume, while maximizing OAP for an overall volume that is adequate for the hosted payload.

Table 1: Spacecraft Utility Examples [18,11,12,19,20,21,2]

Mission	Bus Cost ($K)	Mass (kg)	η	OAP (W)	Volume (cm^3)	*ScU*
SpaceChip	2.7	0.01	0.01	0.001	2×2×0.3	1.2×10^{-13}
MCMSat	24	0.170	0.1	0.88	10×10×1	8.4×10^{-8}
PCBSat	13	0.25	0.05	0.88	10×10×2.5	1.2×10^{-7}
$50Sat	0.25	0.22	0.3	0.55	5×5×7.5	3.1×10^{-7}
1U CS	75	1	0.1	1.6	10×10×10	1.6×10^{-6}
Colony I	250	3	0.4	8	10×10×30	8.9×10^{-5}
Colony II	250	3	0.4	10	10×10×30	0.0001
FS-2	1,500	19.5	0.2	10	32×32×32	0.0006
FS-3	2,100	54.3	0.21	18.9	45×45×63	0.004
DMC	-	88	0.5	30	64×64×68	0.025
FS-5	2,400	137.7	0.51	38	61×72×97	0.043
DMC-2	15,000	96	0.5	50	63×66×84	0.043
SIV	-	181	0.35	100	61×72×97	0.05
FS-6	2,600	164.3	0.48	102	61×72×97	0.07

4. MISSION UTILITY

Mission utility in general for small satellites has been explored previously, but not yet quantified in relation to the quantified *ScU* function just proposed. For some missions, only one spacecraft is required to perform the mission, such as the Hubble Space Telescope (HST). However, it is obvious that three HST's would return more science data than one, but it would exceed the requirements of the overall stated mission. Below is a sample list of missions that require multiple satellites to perform the mission, requiring successively larger constellations.

- Communication relays
- Earth observation
- Earth science (terrestrial)
- Global Navigation Satellite Systems (GNSS)
- Global satellite telephone/data
- Upper-atmospheric space weather monitoring

Mission utility is less straightforward than spacecraft utility, as it is more ambiguous. The main driver of mission utility is the required number of spacecraft to perform a mission. To understand the concept, let us explore the *ScU* of the first Disaster Monitoring Constellation.[20]

Disaster Monitoring Constellation Example

The first DMC had a lofty mission requirement: "To monitor widespread disasters worldwide and distribute data to relief organizations on demand." During mission concept development, one of the key requirements translated from the user needs was that the DMC was to have a 24-hour revisit capability of every location on the globe in order to detect natural disasters. The requirements flowdown produced these top-level system architecture requirements:

- Five small satellites distributed evenly in a 686 km sun-synchronous circular orbit
- Visible 3-band sensor, swath width of 600 km
- Ground sample distance of 32 m
- Four distributed ground stations world wide

Further flowing down requirements to each vehicle, the actual satellite configuration was as follows:

- 88 kg bus mass
- 64 × 64 × 68 cm bus volume
- $\eta = 0.50$
- OAP of 30 W

This results in an *ScU* of 0.025. This value was also shown in Table 1.

Space Weather Forecasting Example

Giving another example, a study proposing a mission to help improve the forecasting of space weather concluded that a constellation of small satellites would be required.[22] The study suggested a mission architecture as follows:

- Ten small satellites evenly spaced in a 90 degree circular orbit, 350-500 km altitude
- Electrostatic analyzer payload
- Single ground station

Again, flowing down the requirements of each vehicle, the actual proposed satellite configuration is as follows:

- 1 kg 1U CubeSat
- 10 × 10 × 10 cm volume
- 10 × 10 × 1 cm payload volume ($\eta = 0.1$)
- OAP of 1.6 W

This results in an *ScU* of 1.6×10^{-6}, as shown in Table 1.

Defining Mission Utility

Observing the two examples one draws the conclusion that five small satellites each with an *ScU* of 0.025 are adequate to satisfy the DMC mission, while ten CubeSats each with an *ScU* of 1.6×10^{-6} are adequate for improving space weather forecasting models. An expression of *MU* is not readily apparent using these two cases.

One approach may be to model the problem as if it were a parallel reliability problem. This would suggest that *MU* can be expressed in terms of *ScU* and the quantity of satellites in the constellation:

$$MU = 1 - \left(1 - ScU\right)^{n} \qquad (2)$$

where n is the number of spacecraft in the constellation and *MU* and *ScU* are as previously defined. Using this approach, *MU*, like *ScU*, approaches unity as the mission configuration gains more utility. Using the prior examples, the DMC mission has an *MU* of 0.12, while the space weather mission has an *MU* of 1.6×10^{-5}. These numbers initially make sense, but should not be directly compared across missions.

Instead, this expression can be used as a tool to compare various architecture proposals for the same mission basic mission requirements. Mission utility more easily facilitates trades between quantity, size, and bus efficiency of spacecraft for a particular mission. Furthermore, an *MU* scale would have to be developed and vetted by the community.

5. OPTIMUM COST

Predicting and determining the optimum cost for a space mission is a challenging undertaking. Many have tried to estimate the true cost of a program, but typically underestimate the cost significantly, resulting in classic program management challenges and in some cases program failure and cancellation.

We propose that optimum cost can be thought of as a quantitative value that considers the following parameters:

- Bus cost (drives *ScU*)
- LVI costs (drives *MU*)
- Potential revenue

Optimum cost, as proposed here, suggests that if investments are made with the aim of increasing *ScU*, then *MU* will increase, coupled with any increase (or decrease) in the number of satellites employed. Increasing *ScU* and *MU* will in general accelerate recovering mission costs by more quickly generating revenue in commercial applications, as the system should be more capable.

Individual Spacecraft Costs

As shown in Table 1, individual spacecraft costs for small satellites can range from tens of thousands dollars for academically-built systems, 100-200 thousand dollars for commercially procured CubeSat buses, up to single and possibly double digit millions of dollars.

Spacecraft utility is again a direct function of the bus design. A larger investment here, primarily in bus function impact reduction, will result in a larger *ScU*.

LVI Costs and Containerization

One of the greatest innovations in the small satellite industry was the introduction of containerized deployment systems. With several different approaches attempted by academia and NASA, the space community has generally adopted the P-POD deployment system for CubeSats. However, this approach is not necessarily the most cost effective solution on a per kilogram basis.

The basic P-POD has a mass of 3 kg and has the ability to deploy up to 4.5 kg of CubeSat mass, in a variety of configurations. The LVI overhead in this case is 40%. Original CubeSat program data advertised a cost of $40K per 1U (1 kg) CubeSat. Costs have now risen to $85K for a 1U deployment and $220k for a 3U CubeSat deployment through Nanoracks. The Naval Post Graduate school's CubeSat launcher (NPSCuL-Lite), can accommodate up to eight 3U CubeSats. The mass of the deployment system itself is approximately 41.3 kg, while deploying 36 kg of CubeSats. This results in an LVI overhead of 53.4%[23].

However, when one looks at the ESPA LVI system, for a 181 kg satellite, the LVI mass overhead is only 13%. This is found when you consider 1/6th of the ESPA ring is the overhead (104 kg total ESPA mass, plus the adapter and separation system mass of approximately 10 kg for each ESPA-class vehicle)[2]. (Ariane's Structure for Auxiliary Payloads is a similar concept to ESPA with much more flight heritage, and arguably a better approach given that secondary payloads are loaded axially, vs. cantilever loading on ESPA. But is appears that ASAP is not as mass efficient as ESPA)[7].

What this simple comparison illustrates is that there is a very high mass cost for LVI of CubeSat systems. This is due to the containerization system. Total launched mass is what is typically used to calculate the rideshare costs. However, one must recall that containerization does partially offset some of the other aspects of LVI, such as assuring no interference of any kind from the rideshare.

Determining Optimum Cost

What this short discussion suggests is that small satellite designers must look at the whole system architecture when trying to determine the optimum cost. Firstly, the individual spacecraft bus costs will drive *ScU*. Particularly, the more investment into increasing the payload volume fraction and the OAP will improve the individual spacecraft utility.

ScU then drives *MU*, whether or not more than one spacecraft is required to accomplish the mission. The individual spacecraft cost, the quantity of spacecraft, and the LVI costs are used to determine the overall space segments costs. Other costs, such as the ground architecture, mission management, etc. are not considered.

For academic and/or science missions, cost estimation efforts stop here. The program manager in this setting is motivated to constrain total costs to the available program budget, which is considered "optimum."

For commercially-driven missions, the optimum cost is defined differently and is more complex. In this case, an assumed revenue generation forecast is used to help determine the optimum cost. Different cost scenarios can be considered depending on the profit goal, balancing the spectrum between the fastest return on investment and overall lifetime returns.

6. OBJECTIVE DESIGN

Figure 2 below plots the payload fraction results from Table 1, but includes the power and volume contributions of the Equation 1 as well, where:

$$P_f = \left(\frac{P}{P+100}\right) \quad (3)$$

$$V_f = \left(\frac{V}{V+1}\right) \quad (4)$$

Note first that the initial balance of 100 Watts to 1 m^3 appears to be a good choice. Also note that the payload volume fraction appears to be the most dominant feature.

An objective design would look at all three parameters and include cost as a consideration. Figure 3 illustrates the *ScU* (solid line) of all the designs considered, with results on a logarithmic scale on the left side. The most recent FalconSAT and DMC missions appear to have the highest *ScU*. Then, *ScU* per cost is plotted as individual data points, as not all costs are available at this time.

A proposed objective design would be a highly bus efficient design, with dimensions of 50 × 50 × 50 cm. the payload fraction would approach 70%, with an OAP of 100 W. This configuration and capability is now possible with technologies that SSTL has developed.

With a target cost of $1M, it would also be the most cost effective solution in this comparison. With a target mass of 30 kg in a non-containerized configuration, the LVI costs could be kept to a minimum in single and multi-spacecraft constellations.

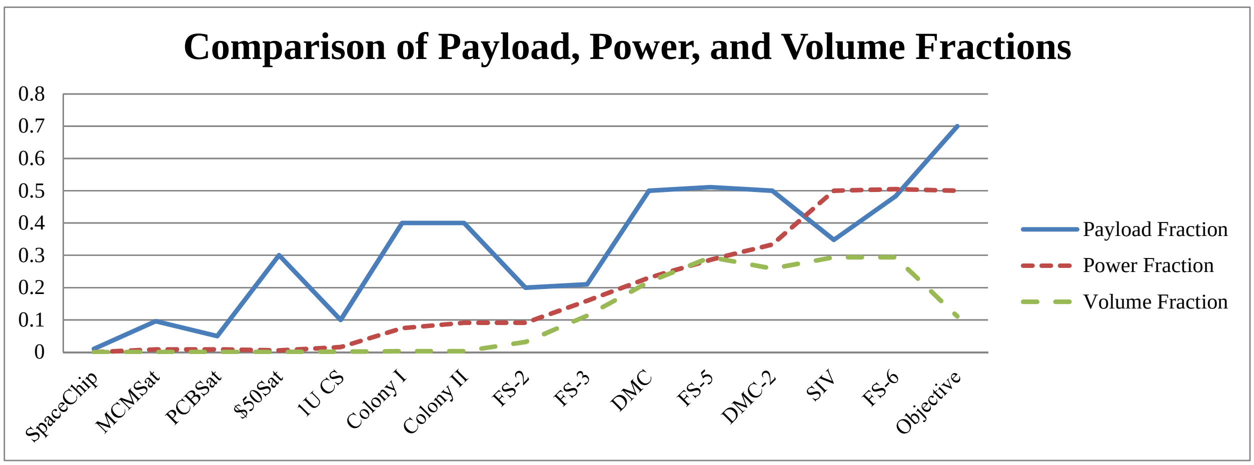

Figure 2: Spacecraft Utility Component Analysis

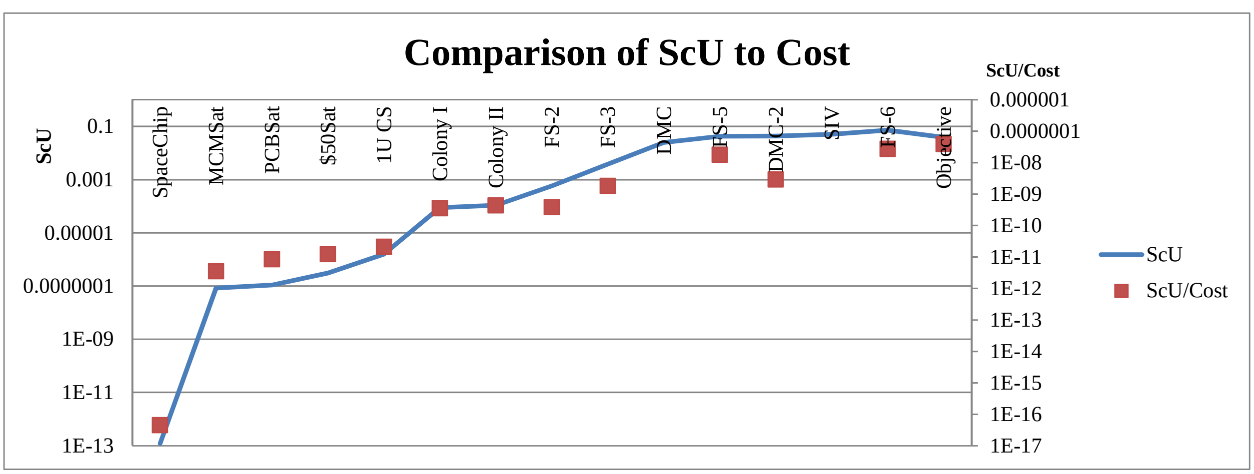

Figure 3: Spacecraft Utility Cost Analysis

7. SUMMARY AND CONCLUSIONS

In summary, this is the first step in taking a fresh look at the optimal size of a standardized small satellite, with the goal of inspiring more research on the topic. Many have argued that CubeSats are too small to be meaningful while others have criticized that the SIV is too inflexible. The combination of spacecraft utility (*ScU*), mission utility (*MU*), and optimum cost helps identify a reasonable spacecraft design point to achieve high mission utility while making fiscal sense. A new standardized, non-containerized configuration is presented, with a mass of 30 kg and 50 cm cubed volume that optimizes construction cost, launch cost, performance capability (hence utility) and value (revenue). Based on community feedback on this approach, the fidelity of the models will be improved with the ultimate goal of "right-sizing" a standardized small satellite form factor and deployment approach.

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