www.adeepakpublishing.com

Arcos, Y., Gurfil, P., and Koenig, M. (2021): JoSS, Vol. 10, No. 3, pp. 1075–1095
(Peer-reviewed article available at www.jossonline.com)

www. JoSSonline.com

Optimized Design of a Thermal Vacuum Testbed for Nanosatellite Verification Tests

Yoel Arcos and Pini Gurfil
Faculty of Aerospace Engineering Technion-Israel Institute of Technology
Haifa, Israel

Michael Koenig
Department of Mechanical Engineering, Braude College of Engineering
Karmiel, Israel

Abstract

Before a satellite is launched into orbit, specific verification exercises should be conducted, including the thermal vacuum (TV) test, among the critical environmental tests. However, many low-budget nanosatellite projects, most of which are led by universities, opt to forego this high-cost testing. Others may perform the testing using superfluous infrastructure, with a process unsuitable for nanosatellites. This article provides guidelines for designing a designated TV chamber for nanosatellites, so that a low-cost test infrastructure can be constructed for the performance of qualification and acceptance tests of nanosatellites. It is further shown that the temperature regime of an orbiting nanosatellite is relatively small, and is easy to emulate in a test facility. For their small size and limited temperature range, nanosatellites can be tested at a system level in a conduction-based facility, reducing the required infrastructure cost significantly in comparison to a radiation-based facility. The results show that a TV chamber measuring 0.6m in diameter and 0.5m in length, based on a conduction cold plate, with a temperature range of −25°C to +55°C, is suited for the verification testing of a 1U-6U nanosatellite and/or one of its subsystems. Due to its restricted temperature range, several low-cost commercial coolants, such as propylene glycol water or ethylene glycol water, may be used in the TV chamber thermal control system.

1. Introduction

With their low-cost, short development timeframe, and ever-increasing capabilities, nanosatellites and CubeSats, which have already served as an important research tool at the university level, are becoming increasingly feasible ventures for industrial companies and space agencies all over the world.

However, before launching any satellite into orbit, it is necessary to validate its design and verify that it does not fail after the launch. This is achieved through a series of qualification and acceptance tests, which include environmental tests in which the satellite is exposed to simulated space environment conditions.

One environmental verification test is the thermal vacuum (TV) test. The TV test is a procedure in which

Corresponding Author: Yoel Arcos – yoelarcos28@gmail.com
Publication History: Submitted – 05/15/20; Revision Accepted – 09/08/21; Published – 10/29/21

a satellite, or a satellite sub-system, is exposed to conditions similar to those it will experience during orbit while its performance under these conditions is tested. During a TV test, the satellite functionality is tested, as well as workmanship and survivability. The TV test is known to be very expensive. It can last for several weeks, as the test procedure dictates several cycles and dwell times. Moreover, the high cost of the test is also related to the expensive test infrastructure and the cost of operation.

As university-run CubeSat projects are often independent, are not bound to any space industry standards, and have limited schedule and budgets, and as TV tests are expensive procedures, these tests are often skipped altogether. Where testing does occur, it is performed using facilities with capabilities beyond what is needed.

Regardless of their size, nanosatellites share some similar features that affect their thermal responses and temperature ranges. Unlike larger satellites, nanosatellites are usually built using commercial components that have very limited operating temperature ranges. This fact alone suggests that nanosatellites may have limited temperature ranges at the system-level, and it may be possible to perform a complete TV test using low-capability facilities designed exclusively for the testing of nanosatellites. This may reduce the cost of the facility in turn, and the TV test for nanosatellites as a whole.

This paper first examines the typical thermal conditions of a nanosatellite during orbit. With nanosatellite conditions established, it then provides recommendations for designing a suitable TV chamber (TVC) optimized for the verification testing of nanosatellites.

This work contributes general guidelines for designing and constructing a low-cost, dedicated TV test facility. With such a low-cost facility, it may be possible to reduce the total cost of TV testing for nanosatellites, which may then increase the chance of manufacturers and universities performing TV tests on newly designed nanosatellites. Ultimately, it is hoped that this may contribute to a decrease in the failure rate of nanosatellite missions after launch.

2. Background

2.1. CubeSats and Nanosatellites

"CubeSat" is a term used to define a class of satellites designed according to a standard size and form factor. The standard CubeSat size, called "1U," measures 10 × 10 × 10 cm, and is extendable to larger sizes from 1U to 6U, and even 12U and more. It was developed by California Polytechnic State University at San Luis Obispo and Stanford University in 1999 (Deepak and Twiggs, 2012) to provide a modular platform for education and space exploration. With increasing collaboration of government agencies as they have developed, CubeSats have advanced from their use in educational programs to industry-level use. They now provide cost-effective platforms for new technology demonstrations, space and science investigations, and advanced mission concepts using constellations, swarms, and disaggregated systems.

The term nanosatellite is used for a class of satellites whose mass is between 1-10 kg. Nanosatellites may be launched individually, or using a multiple-satellite launch, with several nanosatellites working together in a satellite swarm. In 2015, the term "lean satellites" was proposed by the IAA study group (Cho and Graziani, 2015) to define "a satellite that utilizes non-traditional risk-taking development approaches to achieve low-cost and fast-delivery with a small number of team members" (Cho, 2016). In this work, the term nanosatellites will be used to describe small satellites with a mass of 10 kg or less.

Nanosatellites are increasingly capable of performing commercial missions that required much larger satellites in the past. For example, a 6U CubeSat constellation of 35 8-kg nanosatellites has been proposed to replace a constellation of 5 156-kg satellites with similar mission costs, but with significantly increased performance, including coverage of the entire Earth, with images taken every 3.5 hours, instead of every 24 hours (Tsitas and Kingston, 2012).

Since 2014, several proposals for nanosatellite missions in deep space have been presented, such as MarCO, NASA's 6U nanosatellite mission to Mars (Klesh and Krajewski, 2015), and INSPIRE, NASA's first nanosatellite interplanetary mission (Klesh et al., 2013), while new concepts of propulsion have also been presented (Baker et al., 2005).

One of the reasons CubeSats are cost-effective is the use of commercial off-the-shelf (COTS) components to construct satellites with various capabilities. However, integrating COTS components can impose a risk, since such components are not always tested for spaceflight. Complications and failures in operational performance due to inaccurate thermal and electronic testing may even cause mission failure. Until 2012, CubeSats had a 50% failure rate due to a variety of errors that can be classified as functional integration failures, where the satellite system was not operated in a flight-like conditions prior to launch (Maldonado et al., 2015; Swartwout, 2013).

Orbital decay limits the lifetime of most CubeSat missions to periods of several days to several months, due to their release altitudes as secondary payloads (Qiao et al., 2013). While orbiting at low Earth orbit (LEO), CubeSats have an average lifetime of about 200 days (Swartwout, 2013). In spite of their short lifetime, the advantage of low development cost is enough to make them cost-effective. However, a large number of CubeSats do not reach their planned lifetime. About 20% of commercial CubeSat missions fail before reaching their orbit lifetime (Swartwout, 2013), and only 48% of nanosatellites succeeded in mission after successful launch (Bouwmeester and Guo, 2010).

Whereas large satellite manufacturers are dedicated to different testing standards, for nanosatellites, the situation is different. Due to the high cost of TV infrastructure, manufacturers may choose to skip this testing procedure or only execute it partially, thus posing a risk to miss major design faults that otherwise will not be exposed. Cho and Hirokazu (2012) found that thermal vacuum tests were often waived for nano-class CubeSats due to several reasons: a) scheduling constraints; b) unavailable test facility; or c) the test was deemed unnecessary. However, for larger-scale satellites of 50 kg and greater, all the interviewed developers performed the recommended TV test. This shows that unlike traditional large-scale satellites, in the CubeSat field, environmental testing is wrongly considered as not critical, and in turn may contribute to the large percentage of early mortality among CubeSats and nanosatellites.

CubeSats are a great method for testing new technologies, as well as for performing various missions at a significantly lower cost than ever before. However, the CubeSat's main advantage of low cost may also lead to low reliability, and, in turn, high percentage of failures, due to usage of untested subsystems, and/or skipping system-level testing because of a tight schedule or low budget.

2.2. Nanosatellite Thermal Features

The thermal features of nanosatellites, which are crucial for designing an optimized TVC, are the satellite's heat dissipation and temperature range. The satellite's temperature range is strongly dependent on its internal heat dissipation, orbit, mission, and life-time, and may vary between two structurally-identical satellites on different missions.

An orbiting satellite exchanges heat with the environment solely through radiation, while heat flows from different internal components and to the outer surfaces of the satellite via both conduction and radiation. In addition, the thermal conditions experienced by the satellite are changing constantly, since the satellite's orientation is dynamic, and night and day cycles change frequently. For these reasons, satellite thermal behavior is strongly influenced by the mass and thermo-optical properties (i.e., emissivity and absorptivity) of the satellite. The satellite's mass and specific heat dictate the temperature amplitude of the satellite throughout its orbit, while the thermo-optical properties of the external faces of the satellite affect the minimum and maximum temperatures of the satellite.

Nanosatellites with a mass of 3kg or less may experience a temperature gradient of about 50°C on their external walls throughout an orbit of 90 minutes (Jacques, 2009; Corpino et al., 2015). The temperature minimum and maximum of the sub-systems and the external walls are strongly dependent on the satellite thermal design and may vary greatly with size and shape. However, as nanosatellites are designed using COTS components, they are usually based on similar sub-systems with specific temperature requirements. For example, batteries are known to be one of the most thermally challenging components in the satellite, since they have a narrow operation temperature range; the common lithium-ion polymer (LiPo) battery operates at a very narrow temperature range of 0°C to +45°C (Corpino et al., 2015), although other batteries have a larger allowed margin. Other sub-systems require different temperature ranges, but all of the electronic components operate optimally within this range. This alone dictates the overall temperature amplitude on the satellite radiators. A range of about 0°C to +50°C on the radiators is common, and covers the majority of satellites, with rare occasions of thermal designs deviating from this range; otherwise, the electronic components will fail due to either overheating or freezing.

Some satellites are designed with MLI (multi-layer insulation) coating at specific locations, with radiators in other areas to compensate the energy balance with the environment. Radiators are usually very reflective, if expected to be exposed to external solar or IR radiation, or relatively black if not expected to be directed to any external radiative heat source. MLI is used to cover specific areas or sub-systems, acting as a radiation shield, and experiences a large temperature gradient since it does not conduct heat to the satellite itself, and has a low thermal mass. While maintaining a relatively small gradient on the protected sub-systems, externally fixed MLI covers may experience a wide temperature amplitude of −70°C to +60°C (Diaz-Aguado et al., 2006). The use of MLI in CubeSat design is very rare, as this method is more relevant for larger satellites. This is crucial, as the meaning of this is that the temperature gradient between the internal sub-systems and external walls cannot be extremely large, and the external walls temperature is somewhat similar to that of the internal sub-systems, as the external walls act as thermal radiators.

The 6U CubeSat structure is a generic, modular structure with multiple mounting configurations of stack PCBs (Printed Circuit Boards) and other modules and payloads, and is suited for nano-class satellites. The structure/chassis itself is usually made from Al-6061 Aluminum alloy, whose mass is about 1100 grams.[1] The internal component and sub-systems are dependent on the mission profile and design of each satellite and may vary significantly. The outer walls of the CubeSat are designed to shield the satellite's electronic sub-systems from high-energy particles and cosmic rays, and may also act as radiators. The walls, as well as the bottom side of the solar panels, are usually made of anodized Al-6061, due to its low weight and low-cost properties (Jacques, 2009; Jayaram and Gonzales, 2011). Anodized Al-6061 is quite reflective, with relatively low nominal emissivity value of 0.14 and absorptivity of 0.44 (Kauder, 2005).

The average heat generated inside the satellite is between 15W and 40W for microsatellites weighing 10kg or more (Baturkin, 2005), and about 10W or lower for lighter nanosatellites of mass of less than 5kg (Jacques, 2009; Corpino et al., 2015; Chandrashekar, 2016). The heat dissipation may peak for a short period of time to higher levels during payload operation, but still is limited to the capabilities of the power source sub-system of the satellite.

The energy consumed by the different sub-systems on-board the satellite is extracted from the solar panels (or solar arrays), and is stored in the battery sub-system for peak operation, and for crucial activities during eclipse. The solar panels, consisting of several solar cells, are pointed towards the Sun as much as possible to maximize the limited efficiency of the arrays.

The most common solar cells for CubeSat applications are triple-junction, made with GaIn/GaAs/Ge layers that are constructed on a germanium substrate, with a maximal efficiency of 28% to 30%[2] (Nishioka et al., 2006). For assessing the total heat dissipation of a complete Cube-Sat, it is safe to assume a maximal solar cell efficiency of 30%.

2.3. Spacecraft Thermal Design Verification

Thermal design verification consists of two fundamentally different activities: hardware verification and thermal analysis verification. The former is required to validate the sustainability of the designed spacecraft in realistic and worst-case conditions, and the latter is necessary to confirm the accuracy of the thermal model and calculations, including assumptions and inaccuracies (Fortescue et al., 2003).

[1] See www.isispace.nl.

[2] See www.azurspace.com.

2.3.1. Hardware Verification

Prior to launch, it is essential to verify that the spacecraft hardware will operate as expected during launch and in space (launch vibration tests, vacuum tests, temperature exposure tests) and will continue to operate correctly throughout the spacecraft's life. This is achieved by exposing qualification samples or units to conditions more severe than will be encountered in flight, to verify that the design is suitably robust (Qualification tests). The flight hardware itself will then be tested to limits that also exceed expected flight conditions, but are less severe than the qualification values (Acceptance tests) (ECSS, 2004). An additional optional Proto-Qual test is sometimes used, as a compromise in which qualification testing is done on flight hardware. This test reduces schedule time and cost of building dedicated hardware for qualification (also known as Flight-Proof testing).

Temperature screening/exposure tests are designed to expose the hardware to different hot and cold conditions, as it will experience during flight with an additional margin. Temperature screening tests include a bake-out test, a thermal cycle (TC) test and a thermal vacuum (TV) test.

The thermal vacuum test is a TC test performed in vacuum, the main objective of which is to demonstrate the ability of the equipment to perform in worst-case conditions during flight, including an adequate margin. A TV test is effective for demonstrating performance and survival/turn-on capabilities (ECSS, 2002).

The thermal vacuum test primarily includes system-level functional performance tests between and at temperature extremes. Emphasis is on component and sub-system interaction and interfaces, and on end-to-end system performance.

During TV testing, functional performance tests are performed at different temperature extremes. The functional performance test is designated to confirm the reliable operation of the spacecraft in the different modes of operation and under temperature conditions more extreme than predicted during flight. The test duration is sufficient to demonstrate the system survivability under the orbital conditions over the predicted life time of the mission. This test is performed in a vacuum on a completely integrated prototype or flight model for qualification and acceptance of the space hardware. The extreme temperature conditions are achieved by varying the test facility's heat sink temperature (Nuss, 1987).

There are several common standards used in the space industry to define the complete verification procedure. Each standard requires different testing conditions, with some standards more conservative than others. MIL-STD-1540, NASA's GSFC-7000, and ESA's ECSS-10-03 are the most common standards, covering both large-scale and small-scale satellites. Each standard specifies different parameter values (such as temperature range, number of cycles, and vacuum level) for qualification and acceptance test procedures.

In addition, there are currently several international standards seeking to specify CubeSat verification procedures (such as ISO1770, ISO19683, and CubeSat Design Specification). Naturally, these standards are less conservative when compared to the three classical space industry standards, with some of them not even requiring a TV test procedure for satellite-level qualification. A complete comparison between all the space industry standards and their approach regarding TV testing is out of scope for the current paper. However, for the purpose of characterizing a standardized low-cost TV facility, the temperature range requirement specified in ECSS-10-03 was used as a baseline, as it is quite conservative, but not the most severe requirement compared to the other standards.

2.3.2. Thermal Model Verification

A Thermal Mathematical Model (TMM) may not be totally accurate, since it is based on several assumptions and estimates of material properties. Thus, it is important to verify the accuracy of the thermal model and, where inaccuracies are found, to amend the TMM accordingly. This is done by performing a thermal balance (TB) test that makes use of scaled down or full-scale models of spacecraft or parts of spacecraft (e.g., the payload module may be tested on its own if the spacecraft service module to which it is to be attached is already a well-established design) (Fortescue et al., 2003).

During TV testing, the temperatures derived from external thermal conditions are imposed on the spacecraft. However, during TB testing, the external thermal conditions themselves are imposed on the spacecraft. Similarly to a TV test, a TB test requires vacuum conditions and a heat sink to simulate the cold radiative environment of space. The difference between the TV and the TB test facility is that a TB test facility also requires an external heat source (or sources) to simulate the external heat inputs.

3. Benchmark Nanosatellite Thermal Behavior

To estimate the temperature regime of an orbiting nanosatellite, a TMM of a sample nanosatellite was used as reference. As discussed above, the components and sub-systems of different CubeSats are common, with typical allowed operating temperature range. Because of this, the components on board are restricted to a limited temperature range, whether it is a 1U or 6U CubeSat. As heat is conducted from the internal components outwards to the external surfaces, a temperature gradient is generated. The gradient is dependent on surface area, conductivity and distance. In principle, the larger the satellite, the larger the temperature gradient, assuming similar thermal-mechanical design. Since the thermal design of a 6U CubeSat is an assembly of 6 1U CubeSats, a 6U CubeSat will generate a larger temperature gradient between its internal components and external walls.

Adelis Satellite Mission for Swarming and Geolocation (SAMSON) is a nanosatellite mission initiated and led by the Asher Space Research Institute (ASRI) of the Technion-Israel Institute of Technology. Adelis-SAMSON includes three inter-communicating 6U CubeSats, with two main objectives: (1) demonstrate long-term autonomous cluster flight of multiple satellites; and (2) geolocate a cooperative radiating electromagnetic source on Earth. The satellite's life expectancy during the mission is at least one year (Gurfil and Herscovitz, 2012). All three satellites have identical payloads, propulsion systems, components, and mechanical structures. Each satellite contains an electric power system with deployable solar panels, a communication system, an on-board data handling system, an altitude control system, and a cold-gas propulsion system for orbit and cluster-keeping. All three satellites are to be launched with the same inclination and semi-major axis, into a near-circular orbit. In orbit, the satellites will maintain a cluster with inter-satellite relative distance ranging from 1 km to 250 km. The orbit is planned to be a near-circular LEO, with inclination of above 35° and with minimum perigee and maximum apogee altitudes ranging from 500 to 800 km, respectively. Adelis-SAMSON has two possible configurations: a fully deployed configuration; and a stowed configuration. Adelis-SAMSON will be in a stowed configuration during launch and will stay stowed while ejected from the POD (Push-Out Deployer) into trajectory. In the stowed configuration, none of the subsystems/components are deployed. After entering orbit, Adelis-SAMSON will gradually deploy its components until fully deployed configuration is achieved. The components deployed are two arrays of deployable antennae and solar panels. Both configurations are shown in Figure 1.

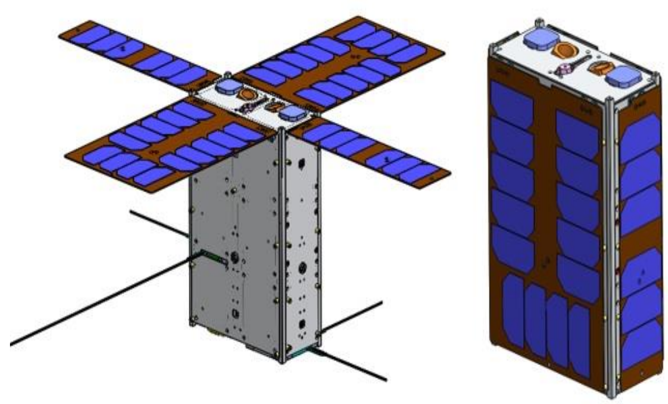

Figure 1: Adelis-SAMSON's two states of deployment.

From a thermal design point of view, Adelis-SAMSON is a typical 6U nanosatellite:

- It is constructed of a typical 6U structure, including external surfaces;
- It has a near-circular orbit, with nominal altitude of 540km (LEO);
- Most of its sub-systems are COTS;

- It has a standard solar panel deployment (2×3U and 2 × 6U);
- No usage of MLI; and
- Passive thermal control system.

These parameters make Adelis-SAMSON an appropriate example of a 6U nanosatellite, and can be used for understanding the thermal behavior of small satellites.

3.1. SAMSON TMM Description

To assess the temperature range of Adelis-SAMSON during its various operational profiles, a numerical TMM was built using Siemens NX Version 12 Space System Thermal. This tool allows a convenient calculation of view factors of the different boundary conditions during orbit, and temperature distribution over time of each node in the finite element grid.

The satellite is divided into five subsystems:

- U1 containing reaction wheels (RW), RW board, GPS + clock board, sun sensor and a GPS antenna;
- U2 containing CPU board, PCU board, PDU board;
- U3 and U4 are reserved for a cold-gas propulsion system, using krypton;
- U5 containing foldable antenna array, AX100, transmitter board and NanMind board; and
- U6 containing the payload and a battery package.

The S-Band antenna is externally connected to the bottom cover plate of the nanosatellite. The solar panels are made from a thin PCB layer that is attached to an aluminum bottom cover.

A finite element grid was constructed for each subsystem (or board-level components), the chassis, the external walls and solar panels, as shown in Figure 2. Material properties (including thermo-optical properties) were defined accordingly, and thermal contacts were specified at given locations. The TMM was verified using several numerical methods. Then, an orbit was defined based on the main characteristics of Adelis-SAMSON's orbit: eccentricity, nominal altitude, inclination, and orientation. It was assumed that Adelis-SAMSON's solar panels will be pointed towards the sun during nominal operation, even inside Earth's umbra (eclipse). During geolocation, the satellite will perform a maneuver to point its antenna towards the nadir for about 15 minutes, and then will maneuver back to its nominal orientation with solar panels towards the sun. These orbit characteristics were defined in order to calculate the transient boundary conditions and view factors during orbit.

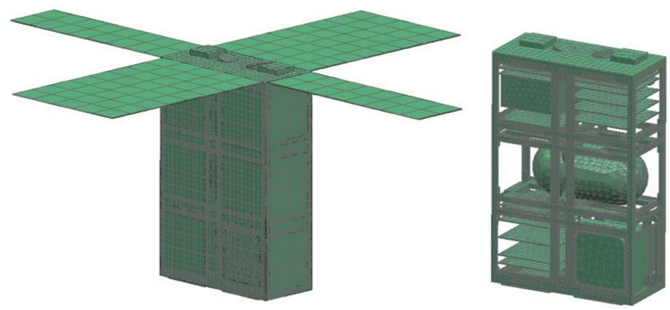

Figure 2: Finite element grid of Adelis-SAMSON.

Enclosure radiation view factors were calculated using hemicube method for every element in the TMM. The radiation view factors calculated included view factor between each element and the Sun, each element and the Earth, each element and the ambient, and between each element and all the surrounding elements.

Several assumptions were made to maintain a realistic complexity level of the TMM. The main assumptions made are as follows:

- All sub-systems are connected to the main chassis using bolts and aluminum frame. Boards are stacked using spacers made of aluminum. Thermal contact between bolt-attached areas is assumed to be 1,000 W/(m^2K) or less, as it is a typical value in deep vacuum (Nishino et al., 1995);
- Since the solar panels are mounted on the main body using a deployable mechanism, the thermal conductance through this mechanism was assumed low. Thus, thermal contact between the solar panels and the main body was neglected in the TMM;

- External antennas (arrays of deployable antennas) were neglected completely in the TMM;
- Short-time heat dissipation (of less than 10 seconds) of any component is neglected in the TMM;
- Earth is regarded to as a black body at an average temperature of −18°C;
- The solar heat flux is constant at 1377 $\frac{W}{m^2}$;
- Albedo constant is 0.3;
- Geolocation occurs once every four cycles, and lasts 15 minutes;
- Geolocation may occur during either day or night. However, since for the purpose of thermal analysis geolocation occurs during daytime at noon, making it a conservatively extreme hot case; and
- During the geolocation procedure, the satellite maneuver is such that its bottom antenna is faced towards the nadir.

Power consumption of each sub-system was defined in the TMM according to Adelis-SAMSON's evaluated electrical power usage. The Safe-Mode profile, characterized by low power consumption (total of 10 to 15W), is used as a nominal cold scenario. The nominal hot scenario is based on a combination of nominal a profile (total of ~ 15W) with a geolocation profile occurring once every four cycles, and may last about 15 minutes (total of 30W).

The results of both the safe-mode scenario and the geolocation scenario are summarized in Table 1. The minimal and maximal temperatures of various surfaces are compared. The solar panels experience the largest temperature gradient during orbit and reaches the lowest and highest temperatures measured int the satellite. The solar panels temperatures remain similar in both scenarios, as the activation of the internal subsystem of the CubeSat does not affect the solar panels. All of the external walls of the satellite experience higher temperatures during the geolocation scenario, as they act as the radiators of the satellite. The lowest temperature inside the satellite reaches about 23°C during the safe-mode scenario, while the highest temperature inside reaches 94°C on the CPU board, located in Unit 2. However, the board allowed temperature requirement is measured at its ambient; in this case the chassis stack frame holding the boards in parallel, measuring about 72°C.

Table 1. Temperature Distribution Along External Faces of Adelis-SAMSON

	Safe Mode, °C		Geolocation, °C	
	min	max	min	max
Solar Panels	16.3	50.8	16.3	50.8
Walls	26.2	37.5	36.3	50.3
Top Wall	25.0	37.2	35.2	48.5
Bottom antenna	23.3	39.2	32.9	50.0

The temperature of the external surfaces of the satellite, as calculated in the TMM for the nominal scenarios of Adelis-SAMSON are shown in Figure 3, under Nominal, SM (Nominal, Safe-Mode scenario) and Nominal, GL (Nominal, Geo-Location scenario).

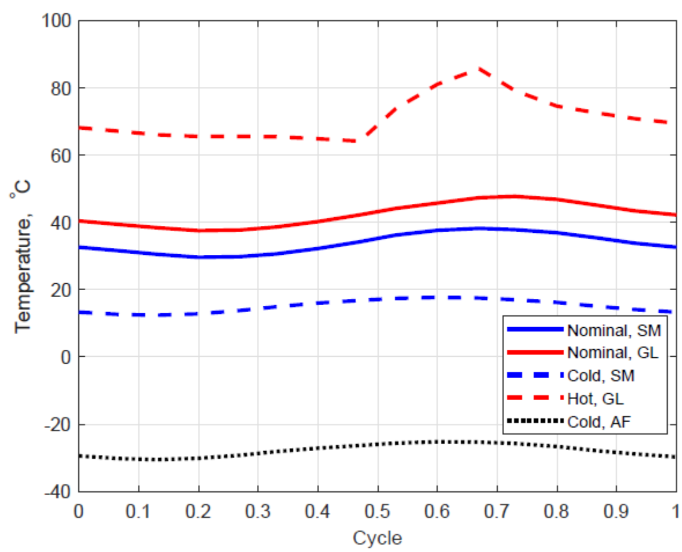

Figure 3: Adelis-SAMSON's external peripheral walls minimal and maximal temperatures.

3.2. Examination of Boundary Thermal Scenarios

The above Adelis-SAMSON TMM should be regarded as a nominal-case scenario, and may be representative of a typical 6U nanosatellite temperature range. In this section, the same TMM will be used, with different conditions, to evaluate high and low limits of a 6U nanosatellite temperature range.

3.2.1. Extreme Hot Scenario

During the Earth's elliptical orbit around the Sun, the solar heat flux magnitude varies. The periapsis of Earth's orbit around the Sun occurs on the third day of the year (January 3), at a distance of 0.983AU, resulting in an average heat flux of $1412.7 \frac{W}{m^2}$. Albedo radiation peaks on the same date as the solar heat flux, since albedo radiation is directly related to direct solar radiation. Earth IR radiation stays relatively constant throughout the year, and does not peak as a function of Earth's true anomaly.

As discussed earlier, a satellite's thermal boundary conditions are influenced by its altitude, among other parameters. Earth's view factor, and the albedo view factor increase with lower altitudes, resulting in larger heat fluxes on the satellite. A low limit LEO altitude of 400 km is taken as reference. Lower altitudes result in very short lifetime and are not common.

As a satellite orbits the Earth, throughout most of the orbit its solar panels are directed towards the Sun. By doing so, the satellite walls are not exposed to the Sun, since they are hidden behind the solar panels. Direct solar radiation on those walls for an extended period of time may result in over-heating of electronic sub-systems, or other failures. For this reason, the solar panels act both as an energy supply sub-system, and as a heat shield for the satellite's main body. However, while this is true for most of the orbit period, it is not true for all of it. During orbit, a satellite will perform a maneuver and change its orientation. While doing so, the satellite's walls are exposed to direct sunlight for a short period of time, without the solar panels protecting them. To assess hot scenario temperatures, we can assume a maneuver for about 15 minutes once every four orbits. This scenario represents a payload mission above a fixed point on Earth, similar to Adelis-SAMSON.

Some nanosatellite missions may operate at a Sun-synchronous orbit (SSO). SSO is achieved when the satellite's orbit angular precession, $\dot{\Omega}$, is equal to that of the Sun's, thus the angle between the orbit's normal and the Sun is approximately constant. This means that the satellite is crossing the equator at constant times of the day. When the local time at ascending node (LTAN) is either 06:00 or 18:00, a special case of SSO is achieved: a dawn/dusk orbit, during which the orbit's normal points at the Sun, resulting in a satellite that never eclipses, and is constantly exposed to solar radiation throughout its orbit. In a dawn/dusk orbit, the inclination is about 66.5°, as Earth's obliquity is about 23.5° relative to the ecliptic plane. An extreme hot scenario will contain a dawn/dusk SSO, with LTAN of either 06:00 or 18:00.

Naturally, the extreme hot scenario will include the satellite's highest heat dissipation profile. For such a profile, Adelis-SAMSON's nominal plus geolocation profile will be used, in accordance with maneuvering once every four orbits. Thus, the internal heat dissipation profile will contain three cycles of nominal heat dissipation profile, and one extreme heat dissipation profile.

Finally, the thermo-optical properties of the satellite are significant factors in its thermal behavior. These properties may change over time, as solar radiation affects the coating quality. Moreover, a certain deviation from values found in the literature may be common due to manufacturing quality and abrasion. For an extreme hot scenario, high absorptivity and low emissivity values will be used, with a deviation of 10% from nominal values. Table 2 shows the different parameters and their impact on the thermal behavior of

Table 2. Extreme Hot Scenario Parameters

Parameters	Description	Value
Solar heat flux	Max. solar heat flux (At Periapsis-January 3rd)	$1412.7 \frac{W}{m^2}$
Satellite orientation	Cruise + maneuver every 7th cycle	15 min Maneuver
Nominal altitude	Satellite‘s altitude above the Earth	400 km
LTAN	SSO with orbit‘s normal points towards the Sun	06:00 or 18:00
Internal heat dissipation	Adelis-SAMSON's cruise + geolocation profile	15 to 30 W
Mass	Total satellite mass	8 to 10 kg
Absorptivity	External walls absorptivity	Nominal + 10%
Emissivity	External walls emissivity	Nominal - 10%

the satellite during the extreme hot scenario. The temperature of the external surfaces of the satellite, as calculated in the TMM for the extreme hot scenario are shown in Figure 3, under Hot, GL (Hot Geo-Location scenario).

From the results of the extreme hot scenario, it is clear that this scenario is too severe; no satellite containing electronic sub-systems is expected to function properly in such conditions. In this scenario, the satellite external walls cycle between 62°C and 85°C, which will result in over-heating of almost any component on board, since internal sub-systems and components will always experience higher temperatures than the external walls.

3.2.2. Extreme Cold Scenario

Earth's apoapsis occurs on the 184th day of the year (July 3), at which Earth's distance from the Sun is 1.016AU, and the solar heat flux reaches a minimal value of $1322.5\frac{W}{m^2}$. The extreme cold scenario occurs on this date, where both solar and albedo radiation decrease to a minimum.

As described in the previous section, during cruise (i.e., most of the cycle), the solar panels are directed towards the Sun, thus shading the walls of the satellite's main body.

A maximal typical LEO altitude of 1,000 km is used as a high limit.

LTAN of 00:00 or 12:00 is suitable for the extreme cold scenario, since the satellite spends about 30% of its orbit in eclipse, allowing the satellite to cool down to low temperatures. With these LTAN values, inclination angle does not impact the thermal behavior of the satellite, so either value can be taken for the analysis. For this reason, Adelis-SAMSON's inclination angle of 35° will be used.

A functioning satellite has some heat dissipation even during minimal operation. Adelis-SAMSON's safe mode will be used as a power profile during the cold scenario. Table 3 shows the different parameters and their impact on the thermal behavior of the satellite during the extreme cold scenario. The temperature of the external surfaces of the satellite, as calculated in the TMM for the extreme cold scenario are shown in Figure 3, under Cold, SM (Cold, Safe-Mode scenario).

3.2.3. Activation Failure Scenario

During ejection of the satellite from the launcher into orbit, there is a chance of deployment failure, which may lead to delay in the start-up of the satellite. Some time may pass until the satellite is completely deployed and turned on (during which the satellite can cool down to low temperatures), since no internal heat dissipation occurs during this time. To assess the conditions possible for a satellite cold start-up, the TMM was used to simulate a case in which the satellite is ejected from the launcher, and is not turned on for a total of 15 cycles. In this simulation, the satellite's conditions are similar to the conditions in the extreme-cold scenario, so that the minimum possible temperature will be achieved.

This scenario is practical for evaluating the final cold temperatures to which a 6U satellite will stabilize

Table 3: Extreme Cold Scenario Parameters

Parameters	Description	Value
Solar heat flux	Min. solar heat flux (At Apoapsis- July 3rd)	$1322.5\,\frac{W}{m^2}$
Satellite orientation	Cruise	-
Nominal altitude	Satellite's altitude above the Earth	1,000 km
LTAN	Sun at orbit's normal	00:00 or 12:00
Internal heat dissipation	Adelis-SAMSON's 'Safe Mode'	10 to 15 W
Mass	Total satellite mass	8 to 10 kg
Absorptivity	External walls Absorptivity	Nominal - 10%
Emissivity	External walls emissivity	Nominal + 10%

after a long period of time without activation. However, it is important to mention that a satellite that failed to start up after more than two cycles will most likely never start up, as it has experienced a critical failure.

Assuming an initial temperature of 20°C at ejection, where R is the Earth's radius, h is the orbit altitude, and β from the launcher, we can learn from Figure 4 that the satellite reaches a quasi-stabilization after about ten cycles, and an amplitude of 10°C is maintained between −33°C and −20°C. The satellite's temperature inside the launcher is unknown, and is dependent on a large number of variables, such as ambient temperature during launch, number of payloads ejected before the given satellite and more. However, the initial temperature does not affect the final temperatures, since stabilization is reached.

Due to a combination of the fact that the ambient temperature inside the launcher is kept at about 20°C, and the fact that satellites start up after no more than

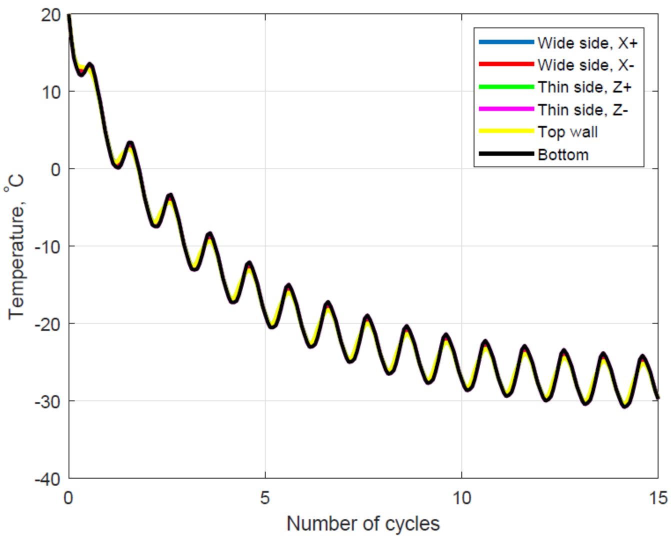

Figure 4: Temperature distribution - Activation Failure Scenario.

two cycles, the lowest temperature a 6U nanosatellite will reach during an activation-failure scenario is about −5°C. If a satellite starts up after an extraordinarily uncharacteristic eight cycles, it will reach a minimal temperature of about −25°C.

The temperature of the external surfaces of the satellite, as calculated in the TMM for the activation failure scenario are shown in Figure 3, under the Cold, AF (Cold, Activation-Failure scenario).

It is worth mentioning the effect of beta (β) angle on the thermal behavior of nanosatellites. Beta angle of a single satellite may vary throughout the year, changing its eclipse duration to sun exposure ratio. The maximal eclipse fraction a satellite can experience in LEO is ~0.35, meaning 65% of the satellite period it will be exposed to direct solar radiation. The minimal eclipse fraction can reach to 0, and last for a few days over a year. The eclipse fraction can be calculated using the following equation (Gilmore and Bello, 1994):

$$f_{\mathrm{E}} = \begin{cases} \dfrac{1}{180^\circ} \cos^{-1}\left[\dfrac{\sqrt{h^2+2Rh}}{(R+h)\cos\beta}\right], & if\ |\beta| < \beta^* \\ 0, & |\beta| \geq \beta^* \end{cases}$$

where R is the Earth's radius, h is the orbit altitude, β is the orbit beta angle, and β^* is the beta angle at which eclipse begins. The maximal eclipse fraction will be the coldest scenario the satellite will experience, in terms of external conditions. The minimal eclipse fraction will be the hottest scenario of the satellite. The TMM discussed above was executed under conditions of maximal eclipse fraction, thus it is already representative of a cold case scenario. Evaluating the extreme hot case conditions can be performed using the two methods discussed herein: either based on a dawn-dusk orbit, or based on an extreme beta angle. As extreme beta angle conditions may last several days in a year, in contrast with dawn-dusk conditions, which may remain constant, in this paper, the extreme hot case was evaluated based on a dawn-dusk orbit. The hot case scenario discussed above is already too severe for a real satellite to endure, thus it is not necessary to run it with a minimal eclipse fraction. Moreover, while designing a TV chamber, heating the tested item is a much simpler challenge than cooling it down, and usually the performance of the facility will be measured by its ability to reach the minimal required temperature. Thus, eclipse fraction will not change the conclusions presented in this article.

3.3. Summary - Nanosatellite Typical Thermal Behavior

From the Adelis-SAMSON benchmark TMM results (summarized in Figure 3), one can learn that the external faces of a 6U typical nanosatellite can reach a maximum temperature of about 50°C during payload operation, and a minimum temperature of about 25°C during safe-mode. In either power profile, the temperature amplitude of the external wall is about 10°C to 15°C. Though experiencing severe thermal conditions, the nanosatellite maintains a relatively benign temperature distribution throughout its orbit.

The solar panels, being thermally isolated and having low mass, may experience a larger temperature amplitude. For a 6U's typical solar panel structure, the panels may reach a maximum temperature of about 45°C and a minimum temperature of about 13°C. In addition to the Adelis-SAMSON nominal orbit, three more scenarios were examined (extreme hot, extreme cold, and activation failure), where the satellite is turned off.

We conclude that the extreme hot scenario is too severe, resulting in an over-heating of almost every sub-system and component aboard the satellite, meaning this scenario is unrealistic and may never occur for a functioning nanosatellite. However, this scenario may be useful as an upper limit condition for the TV testing procedure of a nanosatellite.

The extreme cold scenario is also useful, as it shows that in very cold conditions, the satellite's external walls will reach a minimum temperature of about 1°C, where its internal faces and sub-systems will maintain higher temperatures throughout the satellite life time.

4. TVC System Design

After concluding the representative thermal features of nanosatellites, the design of the optimized TVC can commence. First, we list the key design parameters that are important to address while designing the TVC. Then, we schematically describe the system design of the required chamber, after which we perform a thermal analysis on the satellite-TVC using the nanosatellite TMM. This analysis is also used for comparison of different design methods. Finally, we present the analytic flow analysis required for choosing the appropriate cooling system.

4.1. Parameters for TVC Design

• Chamber Size

The size of the chamber is the most basic feature, and is mainly derived from the size of the tested item. As we plan to mainly use this chamber for testing of common nanosatellites or nanosatellite sub-systems, the 6U CubeSat standard design will be used as a reference and top-limit volume bound. Since a 6U CubeSat has a standard volume and shape, it is simple to evaluate the circumference of the enveloping shrouds, and thus the size of the chamber itself.

During TV testing of a satellite or a tested item at a lower level of assembly, the unit under test is usually operated using an external DC power supply or a similar method. In this event, deployment of the solar panels is unnecessary, and does not contribute to the success of the test. If possible, disassembling the solar panels completely, and inserting the satellite without them, may allow usage of a smaller chamber, thus significantly reducing the cost of the required test infrastructure.

The tested satellite may contain components that will not allow operation of certain sub-systems before the solar panels are deployed. In this case, a solar panel emulator (with similar weight or attaching mechanism) can solve this problem and allow testing in a smaller chamber. The solar panels can be placed alongside the satellite during the TV test or be tested separately.

A standard 6U CubeSat, without its solar panels, measuring $0.34 \times 0.1 \times 0.226$m, and can be conveniently installed inside a cylindrical TVC with a diameter of 0.6m and a length of 0.5m. A TVC with combined cold plate and thermal shrouds will contain the mentioned thermal shrouds, and a cold plate with a minimal size of $0.4 \times 0.2\text{m}^2$, that will allow attachment of the $0.33 \times 0.1\text{m}^2$ surface area of a 6U CubeSat. The thermal shrouds should be about 0.6m in diameter with a minimal length of 0.5m, so that the CubeSat is fully enveloped.

Alternatively, a cold plate-based chamber, including only a cold plate without thermal shrouds, may contain a cold plate measuring 0.4×0.2m, sufficient to conduct heat from any item with a maximal size of a 6U CubeSat.

• Vacuum Pump System

The vacuum pump system shall be capable of reaching a pressure of 10^{-5} Torr or lower inside the above chamber volume, in about 30 minutes or less (this evacuation time requirement can be found in most space environmental testing standards). Given the minimal chamber dimensions mentioned above (diameter of 0.6m, and length of 0.5m), the initial air volume inside the chamber is about $0.94m^3$ (or $33.2ft^3$). Thus, the vacuum pump must be capable of average flow of at least 1.1cfm. This requirement is considered quite achievable for most vacuum pumps.

• Temperature Range

In previous sections, we concluded that for a 6U CubeSat, in a nanosatellite-class weight, a typical temperature range of 10 to 50°C can be predicted on external surfaces, excluding solar panels. Adding a qualification temperature margin of 10°C to the expected temperature range, the TVC should be able to expose the tested item to a temperature range of 0 to 60°C. To estimate the required radiation and conduction heat sink temperatures, a thermal analysis is presented in the next section. For more extreme cases, a wider temperature range of −5 to 65°C can be expected (including uncertainty margin). This will be discussed in the next section.

• Test Duration

As presented in Section 1, there are methods of conducting a TV test using an open-loop cooling system, resulting in a time-limited test that is dependent on the coolant capacity. According to the verification standards presented in Sections 2 and 3, it is clear that a complete TV test, of either assembly level, is a continuous procedure consisting of several stabilization cycles, each lasting for several hours. Thus, for a TV test-bed facility optimized for nanosatellites, it is best to adhere to a closed-loop cooling method that will not limit the duration of the test. A test of eight cycles, each with two soaking periods of two hours (in accordance with ESA-10-03A) is the minimal duration limit of the governmental standards; however, since no duration limit is present, any combination of cycle number with soak duration can be used.

• Cooling/Heating Concept

The tested nanosatellite shall be exposed to radiative environment, which will impose the desired temperature range on the satellite's external surfaces. This is usually achieved using temperature-controlled thermal shrouds that engulf the satellite, acting as a radiative heat sink. The thermal shroud temperature is controlled by coolant flowing inside it, and through a closed-loop cooling system located outside the vacuum chamber. The pipes running from the cooling system to the shrouds are also running, in parallel, through a cold plate fixed inside the chamber, allowing it to be temperature-controlled as well. The external thermal control system is also known as a "chiller." The chiller is a closed-loop cooling unit, containing a liquid pump, compressor, coolant-air heat exchanger, and a control system. Chillers are a common product of commercial use, and may be found in various sizes, cooling/heating capacity and temperature range. These products usually contain a user control system that enables the user to define a temperature set point, either constant or dynamic. The control system must display the current set point and measured reference temperature inside the chamber at a specified location. The coolant itself is chosen according to the desired temperature range, and desired heat removal rate.

To evaluate the required cooling capacity of the cooling system, an estimation of the total heat dissipated from the tested item shall be performed. As energy conservation exists, the dissipated heat cannot be greater than the total power consumed by the tested item. The largest item required for testing is a fully integrated 6U CubeSat, so a calculation of the maximal power consumption of the satellite will yield its maximal heat dissipation.

A 6U CubeSat includes a set of two deployable 3U and two deployable 6U solar panels. It is possible to mount additional solar arrays on the satellite; however, it may be in vain, since those arrays would not experience direct solar radiation, and may not produce enough electricity to justify the cost and effort.

Deploying solar panels may result in a maximum of 2 × 3U + 2 × 6U area of solar panels for a 6U nanosatellite such as Adelis-SAMSON. The solar arrays mounted on the panel are smaller than the actual panel size. Assuming 30% efficiency for the 75 × $75mm^2$ solar array, it is easy to calculate that for the solar heat flux at LEO ($1370\frac{W}{m^2}$), nanosatellite class solar panels will produce about 2.3W of electricity per

1U. Thus, the heat dissipation of a standard 6U CubeSat will be no more than 2.3W × 2 × 3U + 2.3W × 2 × 6U = 41.4W, or ~ 40W. Moreover, recent CubeSat designs follow this trend, with numerous examples of 3U-6U satellites with a power consumption profile peaking under 50W (Praks et al., 2021; Bulut, 2021; Lucchetta, 2021; Prokopyev et al., 2021)]. Thus, a closed-loop cooling system of a thermal vacuum chamber specified for nanosatellite-type CubeSats will be required to dissipate no more than 40W per satellite, in addition to parasitic heating from the surrounding walls.

The vast majority of nanosatellite-class CubeSat missions have been launched into LEO, at a low-eccentricity, almost-circular orbit. There are currently only a handful of nanosatellite missions planned or launched outside of LEO. Thus, we can assume that a thermal test-bed optimized for nanosatellites should be sufficient to simulate the thermal conditions of a nanosatellite orbiting at a circular orbit in LEO, with altitude between 400 – 1, 000 km.

To choose the correct thermal control concept, we evaluate the required temperature of the heat sink, in both hot and cold scenarios, and the required heat dissipation. Then, we may decide on the most suitable coolant and heat sink design for the test chamber.

• <u>Temperature Measurement System</u>

The TVC should be able to support the measurement of at least 15 temperature sensors, since system level testing may require a large number of measurements for gradient evaluation. A common temperature sensor is a thermocouple of type K, with high accuracy in the desired temperature range, but other sensor types can also be considered. In any case, the chamber should have a designated connecting flange to allow passage of the sensor wires from the inside to the outside of the chamber. The temperature sensors are placed at different points of interest inside the chamber and on the tested item. Additional reference temperature measurements may be recorded on the shroud and cold plate; however, it is best if the reference temperature measurement is fed directly to the chiller control system.

• <u>Power Supply</u>

During the test, the tested item may be operated using an external DC power supply, as the solar panels cannot produce any voltage inside the chamber and thus battery life may not be sufficient. To activate the tested item, whether it is a sub-system or a completely integrated satellite, a wiring connection should be possible between the item and the outside of the chamber. To keep the number of potential leakage points to a minimum, it is best to use the temperature sensor connecting flange for power supply connectors as well.

4.2. TVC Benchmark TMM Description

With the aforementioned parameters in mind, we now present the TMM of the nanosatellite mounted inside a TVC. The TMM will be used to verify several features, with different design solutions, and the performance of these solutions will be compared:

- Cylindrical thermal shrouds (partial encirclement, radiation);
- Combination of cylindrical and spherical shrouds (complete encirclement, radiation);
- Combination of cylindrical shrouds with cold plate (partial encirclement, radiation and conduction); and
- Cold plate (partial encirclement, conduction).

After the design concepts above are compared, a preferred solution is recommended, and an example of a complete qualification on the Adelis-SAMSON satellite, with the recommended TVC design and thermal conditions, is presented.

4.3. TV Chamber TMM and Comparison of Design Concepts

The thermal analysis of the TVC is based on the Adelis-SAMSON benchmark TMM presented in the previous chapter, with different boundary conditions. Whereas in the original TMM the boundary conditions were time-varying radiation view factors to space and external heat sources, the conditions are simpler in the TVC TMM. The boundary conditions in the TVC TMM can be either radiation or conduction. Radiative boundary conditions are set to represent the non-contacting thermal relation between the tested item and the thermal shrouds. Conduction conditions represent the thermal connection between the tested item and the cold plate (if it exists), as the tested item is physically attached to the cold plate. Figure 5 shows Adelis-SAMSON, without its solar panels, mounted inside a benchmark TVC. Similar to the TMM of the satellite itself, the shroud and cold plate geometries are meshed, so that the thermal problem may be solved using a numerical simulation with a finite element grid (see Figure 6).

As heat is dissipated from both thermal shrouds and the cold plate using a closed-loop cooling system to an external cooling system, it is assumed that the shrouds and cold plate remain at a relatively constant temperature while exchanging heat with the tested item.

Since the chamber walls are mostly affected by external natural convection to the surrounding ambient, they remain very close to the ambient temperature (or room temperature: ~25°C). For analysis purposes, we assume the chamber walls remain at room temperature throughout the test, whether the tested item is heated or cooled.

For comparison reasons, the set-point of the boundary conditions (either shrouds or cold plate) was set to 0°C, so that the temperature gradient on the satellite is readily apparent. The thermal model was run for a simulation duration of 24 hours, so that steady state could be achieved. Temperature gradients and stabilization times can then be compared. The satellite heat dissipation was according to Adelis-SAMSON's safe-mode profile. For this reason, temperatures do not reach a fixed value stabilization, but rather temperature oscillations are observed as the PCU profile changes with time.

4.4. Comparison of Designs

One of the compared parameters is the temperature range required for complete qualification 6U CubeSat testing. In the previous chapter, we concluded that a 6U CubeSat may experience a temperature gradient (on its radiators) that varies from about 10°C to about 50°C in different scenarios. A 1U CubeSat may even experience a range of 0°C to 50°C, as discussed in section 2.2. Qualification testing requires an additional uncertainty margin with values varying between the different standards. An uncertainty margin value of 10°C suffices for all the reviewed standards. Moreover, during ejection, a 6U CubeSat may cool down to a minimal temperature of about −10°C, while a 1U CubeSat may even reach a low temperature of −20°C, in case of a delay between ejection and start-up.

Thus, to induce on the satellite a temperature range of −10°C to 60°C (including uncertainty margin), and a minimum temperature of −20°C for cold start-up, the thermal heat sink must be set to a specified temperature. The last parameter compared in this section is the

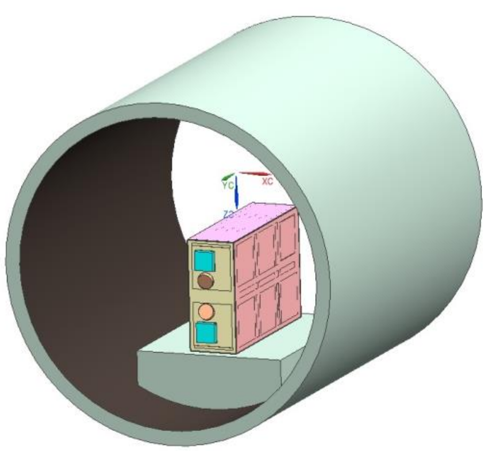

Figure 5: A 6U CubeSat mounted inside the benchmark TVC.

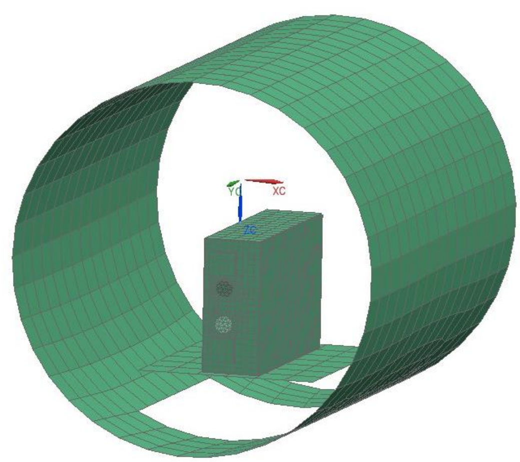

Figure 6: Finite element grid of the benchmark TVC TMM.

specified temperature range on the heat sink, required to induce the above conditions on the satellite. The results are shown and compared in Table 4. The term 'Total temp, gradient' means the maximal temperature difference between the between the heat sink and the hottest external surface of the satellite.

Table 4: TVC Concept Comparison

	Cylindrical Shrouds	Complete Shroud Envelopment	Combined Shroud/ Cold Plate	Cold Plate
Total temp. gradient, °C	45.7	45.8	47.5	47.9
Radiators temp. gradient, °C	3.0	3.1	4.7	5.1
Radiators stabilization time, hours	~16	~16	~4	~4
Complete stabilization time, hours	~22	~22	~7	~7
Desired temp. range for qualification test, °C	-40 to +30	-40 to +30	-20 to +55	-20 to +55
Total cost evaluation	High	Very High	High	Low

It can be seen that the results of the cylindrical shrouds and the complete envelopment designs are almost identical. This proves that the shrouded doors have no advantage over the cylindrical shrouds concept, thus allowing the complete envelopment design to be rejected.

The same conclusion can be achieved while comparing the combination of cylindrical shrouds plus cold plate concept to the cold plate concept. In the combination concept, the temperature gradient on the external surfaces of the 6U nanosatellite is slightly lower than in the cold plate concept. However, this is almost negligible for a 6U CubeSat, so for a smaller type CubeSat (or sub-system), the temperature gradient is expected to be even lower. Comparing the total cost of manufacturing and operating infrastructures for these concepts, the cold plate-only design is by far the most economical and thus, preferable.

Simply placing the satellite on the cold plate surface will result in a high contact resistance between the surfaces, meaning low heat transfer. This is not necessarily a problem, and may be desirable in certain scenarios. However, for the purpose of comparison between a conduction solution (cold plate) and a radiation-only solution (shrouds), a somewhat high-contact heat transfer coefficient of 500 $\frac{W}{m^2K}$ was assumed. For a surface area of 10 x 30cm as discussed here, values of 100 $\frac{W}{m^2K}$ to 500 $\frac{W}{m^2K}$ may be achieved using a gap filler or by using about 15 bolts to attach the satellite onto the cold plate (Gilmore and Bello, 1994).

Comparing the cylindrical shrouds concept to the cold plate concept, we conclude that the cylindrical shroud concept's advantage is a slightly lower temperature gradient. On the other hand, the cold plate concept has several advantages, include shorter stabilization time (resulting in shorter test duration), higher minimum required temperature and lower total cost. For these reasons, the cold plate concept is preferable among all the four designs considered. While it is preferred for several reasons, the cold-plate method may impose an unrealistic temperature gradient in some applications (such as a unique satellite internal structure, or if a very low contact resistance is applied between the satellite and the cold plate). However, it is important to remember the goal of the TV test, which is to demonstrate the ability of the satellite to perform in worst-case conditions in flight. If such conditions are met using the cold plate method (i.e., internal components at acceptance/qualification temperatures), then the test is valid. The TV test is not designed for calibration of thermal analysis this is performed by a TB test, as discussed earlier. A TB test is out of scope for this paper.

4.5. Analytical Flow Analysis for Cooling System Selection

For complete coverage of every possible scenario for a 6U satellite, the heat sink of the TVC should be capable of reaching temperatures of −20°C to +55°C, as long as it is based on a cold plate.

To choose the appropriate equipment to maintain such conditions, a simple analytical calculation was performed, based on several assumptions. The minimal temperature of the heat sink dictates which substance can be used, as the freezing point of certain fluids is lower than the desired temperature, and its properties must be compatible with the requirement. The maximal temperature may also limit the usable substances. However, since the maximal temperature of the heat sink is not very high (less than 60°C), it is not limiting, as almost any coolant is usable in this temperature.

There are several compatible substances available in the market, each with its own advantages and disadvantages for different usage purposes. A comparison was performed among some of those substances, with some of their known advantages and disadvantages summarized in Table 5. Given its low-cost, simple application, and good heat capacity, PGW50% was used as a reference in this article and is the recommended substance for usage in low-cost test facilities, such as the one described in this article. However, each of the specified substances may be suitable as a coolant for the recommended TV test facility. The calculation presented below is for PGW 50%.

Table 5: TVC Coolant Comparison

	Lowest working temp., °C	Advantages	Disadvantages
Silicone oils (Siloxane)	-40	Non-toxic, adjustable	Tendency to leak at pipe fitting and corners
Gaseous nitrogen (GN2)	-180	No contamination during leakage, wide temp. range	Requires blower, larger shroud tubing and headers
Liquid nitrogen (LN2)	-180	Refrigeration to low temp.	Expensive infrastructure
Gaseous helium (GHe)	-253	Refrigeration to extremely low temp.	Expensive infrastructure
Ethylene glycol water (EGW50%)	-30	Good heat capacity, low-cost	Toxic, may cause bio-growth
Propylene glycol water (PGW50%)	-30	Good heat capacity, low-cost, less toxic than EGW	May cause bio-growth

While maintaining the tested item at a high temperature of +55°C, the cooling system must be capable of dissipating approximately 10 to 30 W of heat, produced by the satellite in the nominal + geolocation profile.

In addition to the heat dissipated from the satellite due to power consumption of the different sub-systems, the total heat flow dissipated from the heat sink includes the parasitic heat as well. The parasitic heat is the heat flow transferred from the chamber walls to the heat sink by radiation, resulting from the temperature difference between the heat sink and the walls. Naturally, the parasitic heat is larger than 0 only when the heat sink is cooled to a temperature lower than the ambient (room temperature). The assumptions for the analytic calculation are as follows:

- The minimal temperature of the cold plate is −20°C during cold start-up;
- During cooling down to low temperatures, the maximal internal heat dissipation is compatible with Adelis-SAMSON's safe-mode profile (~ 10W);
- During heating to high temperatures, the maximal internal heat dissipation is compatible with Adelis-SAMSON's geo-location profile (~ 30W);
- The walls of the TVC remain at about 25°C (room temperature) throughout the test, inducing parasitic heat on the TVC's heat sink;
- The emissivity of the internal TVC walls is about 0.11 (polished stainless steel);
- The coolant temperature inside the cold plate equals the average temperature between the inlet and outlet temperature to and from the cold plate;
- The chamber is based on cold plate concept. The cold plate area is $0.4 \times 0.3 m^2$;
- TVC diameter is 0.6m and its length is 0.65m; and
- Radiation view factor between the tested item and the chamber's walls is 1.

The parasitic heat induced from the temperature difference between the TVC walls and the tested item can be calculated using:

$$Q_{par} = \in F\sigma(T_{wall}^4 - T^4) = 22.1\text{W},$$

so that the total heat dissipated through the cold plate, during cool-down, is equal to:

$$Q = Q_{par} + Q_{internal} = 32.1\text{W}.$$

An illustration of the cold plate principle is shown in Figure 7, where the coolant is heated from its inlet temperature of T_{cold} to its outlet temperature of T_{hot}, using the total heat dissipation Q. The heat conveyed from the cold plate to the coolant depends on the cold plate thermal resistance. Each cold plate has a certain value of thermal resistance different from zero, due to the internal convection coefficient and the fact that the piping inside it has a finite length. The resistance of the cold plate varies with coolant mass flow inside it, as the coolant velocity effects the convection coefficient. A common value of about 0.02°C/W for either EGW or PGW mass flow of 5LPM can be easily found with several manufacturers.

With these values, the average temperature of the coolant inside the cold plate (during cooling) is:

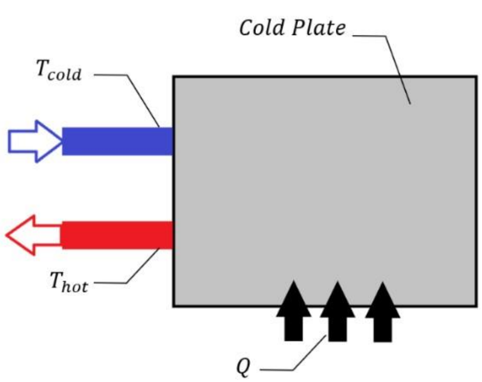

Figure 7. Schematic description of the cold plate and inlet and outlet flow

$$QR_{c.p} = (T_{c.p} - T_{\infty})$$

$$T_{\infty} = -20°C.$$

In other words, during cool-down the temperature difference between the cold plate and the coolant is about 0.6°C for a mass flow of 5LPM. Notice that during heating of the tested item, the total heat dissipation is about 30W, as parasitic heat cancels (the tested item radiates heat to the TVC as it is hotter). Either way, the temperature difference is preserved during either heating or cooling. As a mass flow of 5LPM is assumed, we can now calculate the inlet and outlet temperatures using:

$$Q = \dot{m}_{cp}(T_{hot} - T_{cold}).$$

Pump pressure capacity is dependent on piping length, diameter and bending between the cooling system and the chamber. However, a margin can be added so that the cooling pump shall suitable. Observe that the temperature difference between the inlet and outlet flow is negligible. Thus, the cooling system shall consist with the following properties:

- Usage of PGW (or EGW) as coolant;
- Operating fluid temperature range: −25°C to +60°C; and
- Maximal flow capacity of at least 15LPM.

The example given here is relevant for most CubeSats, assuming a total power consumption of 40W or less. A higher power consumption will result in a higher required coolant flow capacity. The required flow capacity can be easily calculated using the equations and example given above.

5. Example of a Complete Qualification Test

With the recommended design concept of the conduction cold plate, we performed a TMM simulating a complete qualification test for a 6U nanosatellite, with the following features:

- The ambient temperature range between the cycles is −5°C to +55°C;
- The induced temperature range upon the tested 6U CubeSat is 0°C to +60°C;
- The number of cycles is two;
- The soak duration is four hours;
- The maximal ambient temperature transition rate is about 2°C/min;

- Pressure inside the chamber is 10^{-5} Torr;
- The test begins with a hot start-up followed by a cold start-up. This is performed in this way and not the opposite, so that condensation will not occur on the satellite;
- Hot start-up temperature was set to 40°C, but it may be lower dependent on the launcher conditions. Both GSFC-7000 and ESA-10-03A require hot and cold start-up temperatures to be similar to those expected during flight, with no margins; and
- Cold start-up temperature is set to −20°C. Recall during activation failure scenario is about −5°C, as discussed in section 3.2.3.

The described test meets the requirements of ISO19683:2017 for CubeSat verification testing. It is worth mentioning that by a slight change of increasing the number of cycles, and the soak duration, the test can be adjusted to meet GSFC-7000 and ESA10-03A verification standards. An example of the suggested complete TV qualification test on the Adelis-SAMSON benchmark model is shown on Figure 8, including the set point of the cold plate, and the resulting temperatures on the external surfaces of the 6U CubeSat. Required action, such as turn-on, turn-off and functionality test of the satellite are mentioned on top of the presented graph, as described in its legend.

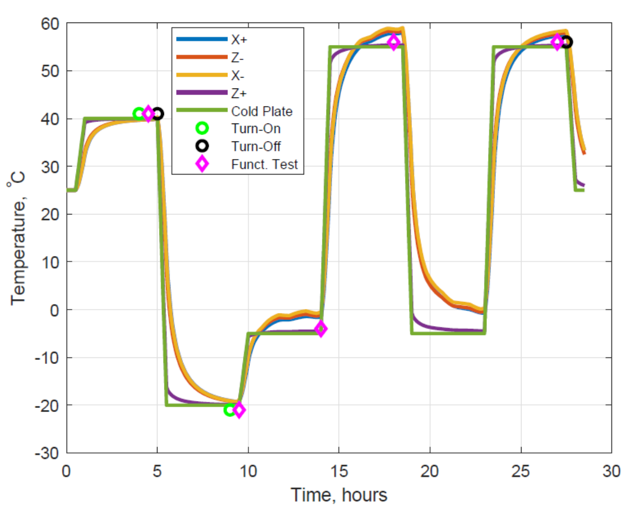

Figure 8: A complete qualification test using cold plate concept for TVC.

6. Conclusions

As the use of nanosatellites is becoming more widespread, it is essential to ensure that their reliability will increase while their design and launch costs stay the same. In this work, it has been shown that nanosatellites, with similar design concepts among multiple projects, are exposed to thermal conditions that require testing before launch, so that their reliability and success rates remain high. To simulate such conditions, a thermal vacuum test should be performed on the satellite to test its functionality under real-life conditions, as well as to test workmanship and other parameters that may lead to failure during launch or in orbit.

As thermal vacuum tests are extremely expensive and require special facilities, this work asks a fundamental question: is it possible to design a TVC specifically for the testing of nanosatellites, such that its construction and operation costs will remain relatively low? If so, this may encourage more manufacturers and universities, which often skip these test series, to perform the TV tests, thus increasing the success rate and reliability of the satellite.

In conclusion, small satellites, from picosatellites of 1kg to nanosatellites of about 10kg, may experience limited temperature ranges, as their typical designs dictate their thermal behavior. With this limited temperature range, TV chamber is proposed based on a relatively low-cost, closed-loop, temperature-control unit, and a typical mechanical design of the chamber. With a low-cost cooling unit, the design and construction costs of the chamber, and more importantly its operation costs, may be reduced significantly. The guidelines for choosing the appropriate hardware and the thermal design of the chamber were discussed and presented, so that such a chamber can be readily designed.

Several design concepts for a TV chamber were examined. After inspecting several parameters and comparing the different concepts using these parameters, a single concept was found to be best suitable for the required test facility. A recommendation was given to design a TVC based on a conduction heat sink (cold plate), capable of maintaining a temperature range between −20°C and +55°C on a tested item, and dissipating ~ 30W while doing so. For this temperature range, the usage of PGW50%, EGW50% or silicone oil as coolant is appropriate, as it meets the temperature requirement. A recommendation was given to use PGW50% as coolant for his advantages over the other options.

An example for complete qualification test at the system-level of a benchmark 6U nanosatellite was presented. The test meets the standards requirements of ISO19683:2017 for CubeSats verification testing.

Acknowledgments

This work was funded by Adelis Foundation and the Israeli Ministry of Science and Technology under Grant 3-9707.

References

Baker, A. M., da Silva Curiel, A., Schaffner, J., and Sweeting, M. (2005): You Can Get There from Here: Advanced Low-Cost Propulsion Concepts for Small Satellites Beyond LEO. *Acta Astronautica*, Vol. 57, pp. 288–301.

Baturkin, V. (2005): Micro-Satellites Thermal Control Concepts and Components. *Acta Astronautica*, Vol. 56, No. 1-2, pp. 161–170.

Bouwmeester, J. and Guo, J. (2010): Survey of worldwide pico and nanosatellite missions, distributions and subsystem technology. *Acta Astronautica*, Vol. 67 (7), pp. 854–862.

Bulut, M. (2021): Thermal Design, Analysis, and Testing of the First Turkish 3U Communication CubeSat in Low Earth Orbit. *J. of Thermal Analysis and Calorimetry*, Vol. 143(6), pp. 4341–4353.

Chandrashekar, S. (2016): Thermal Analysis and Control of MIST CubeSat, Dept. of Space and Plasma Physics School of Electrical Engineering, KTH Royal Institute of Technology, Stockholm, Sweden.

Cho, M., Masui, H., and Hatamura, T. (2012): Overview of Nanosatellite Environmental Tests Standardization Project: Test Campaign and Standard Draft, presented at the 26th Ann. AIAA/USU Conf. on Small Satellite, Logan, UT. Paper SSC12-VII-10.

Cho, M. and Graziani, F. (2015): IAA Study on Lean Satellites Definition and Requirements, presented at the 66th Int. Astronautical Cong., Jerusalem, Israel, IAC-15 Symposium B4. Paper 7.3.

Cho, M. (2016): Recent Status of Lean Satellite Related Standards, presented at the 7th Nano Satellite Symp., Kamchia, Bulgaria. Paper NSS-07-0501.

Corpino, S., Caldera, M., Nichele, F., Masoero, M., and Viola, N. (2015): Thermal Design and Analysis of Nanosatellite in Low Earth Orbit. *Acta Astronautica*, Vol. 115, pp. 247–261.

Deepak, R. A., and Twiggs. R. J. (2012): Thinking Out of the Box: Space Science Beyond the Cube-Sat. *J. of Small Satellites*, Vol. 1, No.1, pp. 3–7.

Diaz-Aguado, M., Fowler, W., Greenbaum, J., and Lightsey, G. (2006): Small Satellite Thermal Design, Test, and Analysis. In *Modeling, Simulation, and Verification of Spacebased Systems III* (P. Motaghedi), Vol. 6221, Int. Soc. for Optics and Photonics (SPIE) Conf. Series.

ECSS Secretariat Requirements & Standard Division (2004): *ECSSQ-700-71A Space Product Assurance - Data for Selection of Space Materials and Processes* (Rev. 1). The Netherlands, ESA Publication Division.

ECSS Secretariat Requirements & Standard Division (2002): *ECSS-E-10-03A Space Engineering – Testing*. The Netherlands, ESA Publication Division.

Fortescue, P., Stark, J., and Swinerd, G. (2003): *Spacecraft Systems Engineering* (4th ed.). Hoboken, NJ: John Wiley and Sons.

Gilmore, D. (1994): *Satellite Thermal Control Handbook* (Vol. 1). Segundo, CA: Aerospace Corporation Press EI.

Gurfil, P. and Herscovitz, J. (2012): The SAMSON Project – Cluster Flight and Geolocation with Three Autonomous Nanosatellites, presented at the 26th Ann. AIAA/USU Conf. on Small Satellite, Logan, UT. Paper SC12-VII-2.

Jacques, L. (2009): Thermal Design of the OOUFTI-1 Nanosatellite, Applied Science Faculty, University of Liège, Liège, Belgium.

Jayaram, S. and Gonzales, E. (2011): Design and Construction of a Low-cost Economical Thermal Vacuum Chamber for Spacecraft Environmental Testing. *J. of Engineering Design and Technology*, Vol. 9, No. 1, pp. 47–62.

Kauder, L. (2005): *Spacecraft Thermal Control Coating References*. Greenbelt, Maryland: NASA-Goddard Space Flight Center, Technical Report NASA-TP-2005-212792.

Klesh, A. and Krajewski, J. (2015): MarCO: CubeSats to Mars in 2016, presented at the 29th Ann. AIAA/USU Conf. on Small Satellites, Logan, Utah, Aug. 8-13. Paper SSC15-III-3.

Klesh, A., Baker, J., Castillo-Rogez, J., Halatek, L., Murphy, N., Raymond, C., Sherwood, B., Bellardo, J., Cutler, J., and Lightsey, G. (2013): INSPIRE: Interplanetary NanoSpacecraft Pathfinder in Relevant Environment, presented at the AIAA Space 2013 Conf. and Exposition, San Diego, CA, Sept. 10-12, Paper 5323.

Koller, M., Loidoldand, L. M., Lffler, T., Meier, J., Klinkner, S., Manoliu, L., Schoch, B., and Kallfass, I. (2021): The EIVE CubeSat - Developing a Satellite Bus for a 71-76 GHz E-Band Transmitter Payload, presented at the 35th Ann. Small Satellite Conf., Online, Aug. 7–12. Paper SSC21-VIII-02.

Lucchetta, G. (2021): MeVCube: a CubeSat for MeV astronomy, presented at the 37th Int. Cosmic Ray Conf. (ICRC 2021), Online (Berlin, Germany), Jul. 12-23.

Maldonado, C., Rodrigues, S., and Ketsdever. A (2015): A Ground-based Facility for Nanosatellite Systems Testing in Relevant Environments, presented at the 2015 IEEE Aerospace Conf., Big Sky, MT, Mar. 5-12, pp. 1-11. doi: 10.1109/AERO.2015.7118933.

Nishino, K., Yamashita, S., and Toriim, K. (1995): Thermal Contact Conductance Under Low Applied Load in a Vacuum Environment. *Experimental Thermal and Fluid Science*, Vol. 10, pp. 258–271.

Nishioka, K., Takamoto, T., Agui, T., Kaneeiwa, M., Uraoka, Y., and Fuyuki, T. (2006): Evaluation of In-GaP/InGaAs/Ge Triple-Junction Solar Cell and Optimization of Solar Cell's Structure Focusing on Series Resistance for High-Efficiency Concentrator Photovoltaic Systems. *Solar Energy Materials and Solar Cells*, Vol. 90 (9), pp. 1308–1321.

Nuss, H. (1987): Space Simulation Facilities and Recent Experience in Satellite Thermal Testing. *Vacuum*, Vol. 37, pp. 297–302.

Praks, J., Mughal, M. R., Vainio, R., Janhunen, P., Envall, J., Oleynik, P., Nsil, A., Leppinen, H., Niemel, P., Slavinskis, A., and Gieseler, J. (2021): Alato-1, Multi-payload CubeSat: Design, Integration and Launch. *Acta Astronautica*, Vol. 187, pp. 370–383.

Prokopyev, V. Y., Bakanov, S. S., Bodrov, V. K., Chernodarov, E. N., Doroshkin, A. A., Gorev, V. N., Kolesnikova, A. Y., Kozlov, A. S., Kus, O. N., Melkov, A. V., and Mitrokhin, A.A. (2021): NORBY CubeSat Nanosatellite: Design Challenges and the First Flight Data. *J. of Physics: Conference Series*, Vol. 1867 (1), 012038.

Qiao, L., Rizos, C., and Dempster, G. (2013): Analysis and Comparison of CubeSat Lifetime, in *Proc. 12th Australian Space Development Conf.*, Adelaide, Australia, pp. 249-260.

Swartwout, M. (2013): The First One Hundred CubeSats: A Statistical Look. *J. of Small Satellites*, Vol. 2, No. 2, pp. 213-233.

Tsitas, S. R and Kingston, J. (2012): 6U CubeSats Commercial Application. *Aeronaut. J.*, Vol. 1176, pp. 189–198.