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Aalto-1, multi-payload CubeSat: design, integration and launch

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Abstract

The design, integration, testing and launch of the first Finnish satellite Aalto-1 is briefly presented in this paper. Aalto-1, a three-unit CubeSat, launched into Sun-synchronous polar orbit at an altitude of approximately 500 km, is operational since June 2017. It carries three experimental payloads: Aalto Spectral Imager (AaSI), Radiation Monitor (RADMON) and Electrostatic Plasma Brake (EPB). AaSI is a hyperspectral imager in visible and near-infrared (NIR) wavelength bands, RADMON is an energetic particle detector and EPB is a de-orbiting technology demonstration payload. The platform was designed to accommodate multiple payloads

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while ensuring sufficient data, power, radio, mechanical and electrical interfaces. The design strategy of platform and payload subsystems consists of in-house development and commercial subsystems. The CubeSat Assembly, Integration & Test (AIT) followed Flatsat–Engineering-Qualification Model (EQM)–Flight Model (FM) model philosophy for qualification and acceptance.

The paper briefly describes the design approach of platform and payload subsystems, their integration and test campaigns and spacecraft launch. The paper also describes the ground segment & services that were developed by Aalto-1 team.

Keywords: Aalto-1, CubeSat, hyperspectral, radiation, Aalto Spectral Imager, Radiation Monitor, Electrostatic Plasma Brake

1. Introduction

Nowadays, there is an increased interest towards small satellite missions due to advances in Commercial Off-The-Shelf (COTS) technology miniaturization. Traditionally, the classification of small satellites is only based on their mass but the CubeSat standard also takes into consideration the volume [1]. Over the past decade, the applications of small satellites in general and CubeSats in particular have increased manifold due to the availability of low-cost design, testing and launch possibilities [2, 3, 4, 5]. Initially perceived for training and educational activities, the applications of CubeSats have expanded in vast application areas in the past few years [6]. Example application areas include remote sensing, Earth observation, disaster management, science, astronomy, space weather and technology demonstration etc. [7, 8, 9, 10, 11, 12, 13].

The abundant availability of COTS components with faster development cycles has led to the NewSpace movement [14]. This approach has led the transformation of CubeSat missions from educational and technology demonstration to real missions with potentially risky but higher commercial and science return [15, 16, 17]. A large number of commercial applications using CubeSats have evolved in the past few years with a promising future scope of commercial applications [2, 3, 4, 5]. Until now, more than one thousand CubeSats have been launched into space [18]. However, the forecast suggests an exponential increase in nanosatellite launches every year [19].

There has also been great advancement in the technology development

for nano and microsatellites [20]. A number of innovative platforms have been designed and demonstrated in space [21, 22, 23, 24]. Due to technology miniaturization, the capability of CubeSat platforms has been ever increasing [25, 26]. The current CubeSat missions are able to demonstrate innovative platforms with high power generation, precise attitude pointing and higher data downlink capabilities with potential to compete with their bigger satellite counterparts.

During the past decade, the worldwide trend of the first satellite by each university or Small & Medium Enterprises (SME) has been designing and launching relatively less complex single-unit (1U) CubeSat for capability demonstration. In contrast, we at Aalto university followed a more challenging approach, i.e., designing a multi-payload CubeSat with student teams. The mission objective was to build and launch a spacecraft with focus on science, imaging and de-orbiting technology demonstration while also providing hands-on educational training. This paper presents detailed design aspects of the Aalto-1 CubeSat with a capability description of payloads and the platform to accomplish the mission objectives. The in-orbit demonstration and lessons learned are presented in an accompanying paper [27].

This paper is organized as follows: Section 2 briefly introduces mission objectives and requirements, Section 3 presents the mission design, project implementation and educational outcomes, section 4 presents space segment design and implementation, section 5 presents all payloads: their specifications and designs, Section 6 introduces design approach of the platform subsystems, Section 7 presents the integration & testing, section 8 focuses on ground segment and Section 9 concludes the paper.

2. Mission objectives

The Aalto-1 satellite project was initiated from Aalto University student's aspiration to make the first satellite mission in Finland. The idea was supported by teachers and developed during a special assignment in Space Technology course in 2010 spring semester in the form of feasibility study of the satellite. The goal of the course was set to develop a realistic satellite concept which should be possible to implement (at least partly) by students. It was required that the main payload should be developed in Finland and it should be connected to Aalto University curriculum. During the feasibility study this was translated to the goal to build first Finnish satellite with Earth Observation payload.

60 For the university, the main driver for the Aalto-1 project was to provide
61 hands-on education in space engineering, science and entrepreneurship, while
62 taking advantage of the NewSpace movement [14, 2, 3, 4, 5] and harness the
63 enthusiasm of building the first national satellite. It was envisioned, that in
64 addition to satellite development, students will also learn to work with expe-
65 rienced space scientists and develop connections to industrial partners. The
66 mission was largely financed and led by Aalto University and integrated to
67 Aalto space technology curriculum.

68 Despite a main goal of building, launching and operating first national
69 satellite, the proposed payload selection introduced complex technology demon-
70 stration and science goals. The first feasibility study built the satellite con-
71 cept around four payload candidates and established a consortium for build-
72 ing a 3U Cubesat. The study also derived the main mission requirements.
73 As an outcome of the feasibility study, the satellite mission and platform
74 was to be developed by Aalto University students and payloads were to be
75 contributed by partner organisations.

76 The main payload candidate was a spectral Earth Observation imager,
77 AaSI, based on technology developed by VTT Technical Research Centre of
78 Finland (VTT). This further led to wider spectral device offering for space
79 applications by VTT. Another payload candidate, a radiation monitoring
80 device, later called RADMON, was proposed by a team from University of
81 Turku and University of Helsinki. The third payload candidate, selected by
82 the study, was e-sail experiment device, EPB, which was already in devel-
83 opment at Finnish Meteorological Institute (FMI) for ESTCube-1 CubeSat
84 mission [28, 29]. In the original feasibility study, a vibration monitoring sys-
85 tem was also proposed. However, the idea was later abandoned as impractical
86 for a rather monolithic nanosatellite.

87 Neither of the selected payloads had flight heritage. Moreover, AaSI and
88 EPB main technology was never demonstrated in space before for proposed
89 purpose. Earth Observation with tunable Fabry–Pérot Interferometer (FPI)
90 was a novel concept and also deorbiting a satellite with electrostatic force by
91 using a tether was never attempted before. This provided a technical chal-
92 lenge and scientific novelty for the project. The project consortium, which
93 consisted of Aalto University, University of Turku, University of Helsinki,
94 VTT and Finnish Meteorological Institute, decided to build a multi-payload
95 mission. In a retrospective, one can say that this decision elongated the
96 project significantly and enforced also several compromises in the design due
97 to contradicting requirements. It took slightly over five years from the first

198 idea to FM completion. The overarching Aalto-1 mission objective was to
199 build a satellite to carry out in-orbit demonstration of AaSI, RADMON and
200 EPB experiments, each of them with specific mission objectives.

201 AaSI's main objective was to demonstrate the operation of a tunable FPI-
202 based spectral imager for Earth Observation in the space environment. The
203 FPI technology developed at VTT allowed to build for the first time freely
204 tunable spectral EO camera to nanosatellite form factor. As a minimum,
205 the instrument was required to take wavelength calibration measurements
206 and record a spectrum of at least six wavelengths of a cloud-free land. For
207 a full demonstration, the instrument was required to take measurements to
208 investigate wavelength stability, thermal effects and long-term degradation
209 of filters, optics, the sensor and other components along with demonstrating
210 various operation modes.

211 RADMON's main objective was to operate and calibrate a CubeSat-
212 compatible radiation detector which registers protons in nine energy channels
213 with threshold energies of 10 – 40 MeV and electrons in five energy channels
214 with threshold energies of 1.5 – 12 MeV.

215 EPB's main objective was to deploy a tether and then charge it to esti-
216 mate the force exerted by the Coulomb drag between the tether's electric field
217 and the Earth's ionosphere, as well as to demonstrate de-orbiting by keeping
218 the tether charged for an extended period of time. This novel propulsion
219 concept was (and currently still is) never demonstrated in space. By now
220 two launched nanosatellites, ESTCube-1 and Aalto-1 have made attempts
221 to deploy this system. However, in the near future AuroraSat-1, Foresail-1
222 and ESTCube-2 are heading towards similar goals using the same technology
223 [30].

224 Detailed mission requirements kept developing along the project and were
225 not documented in detail. Therefore, it can be said, that the mission was
226 technology driven as it often happens in CubeSat missions. However, the
227 proper feasibility study in the very beginning and well established consortium
228 helped to keep the focus on results.

229 The finally launched satellite followed closely the original plan of payloads
230 and functionality, but seriously underestimated requirements due to many
231 constraints including time and resources.

132 3. Mission design and implementation

133 In order to satisfy the payload in-orbit demonstration requirements, Aalto-
134 1 was required to be launched to a polar orbit with an altitude of at least
135 500 km. A polar orbit provides sufficient conditions to estimate the Coulomb
136 drag force [31] and allows for RADMON to measure at various latitudes, in-
137 cluding the South Atlantic Anomaly. Polar orbit also allows coverage in
138 Finland and provides good opportunities for Earth Observation.

139 The attitude requirements were set by all payloads, but dominated by
140 EPB requirements. The lower limit of altitude was required by EPB. In lower
141 altitudes, the atmospheric drag might dominate the de-orbiting impact which
142 makes electrostatic drag estimation difficult. The highest altitude limit was
143 set by 25-year orbital decay requirement for space debris mitigation. The
144 EPB experiment requires spinning satellite of hundreds of degrees per sec-
145 ond in order to provide centrifugal force for tether deployment [32]. The
146 angular momentum was to be provided in steps: spin up the satellite, de-
147 ploy the tether, spin up again, etc. AaSI requires nadir pointing during
148 image acquisition and RADMON requires attitude knowledge, but the re-
149 quirement was not critical. Another notable requirement for the mission was
150 surface conductivity requirement by EPB to keep spacecraft potential during
151 Coloumb drag experiment. The satellite was designed for two years in the
152 orbit, which was estimated as sufficient time to carry out all experiments.
153 The mission design in terms of energy and thermal budget was flexible, as it
154 was decided that payload duty cycles can be adjusted in-orbit according to
155 the need.

156 The satellite operation from Aalto University was one of the key mis-
157 sion requirements. For this purpose, the ground segment was developed.
158 The ground station includes UHV, VHF and S-band steerable antennas and
159 associated transceivers. The mission operation software was designed and
160 implemented by Aalto students.

161 The product tree of Aalto-1 mission with ground segment, space segment
162 and launch segment description is shown in Fig 1.

163 3.1. Project Implementation

164 After successful feasibility study in spring 2010, the satellite project was
165 quickly funded and supported by Aalto MIDE (Multidisciplinary Institute
166 of Digitalisation and Energy)[33]. The project was also formally organized
167 by establishing posts for project responsible professor, project coordinator,

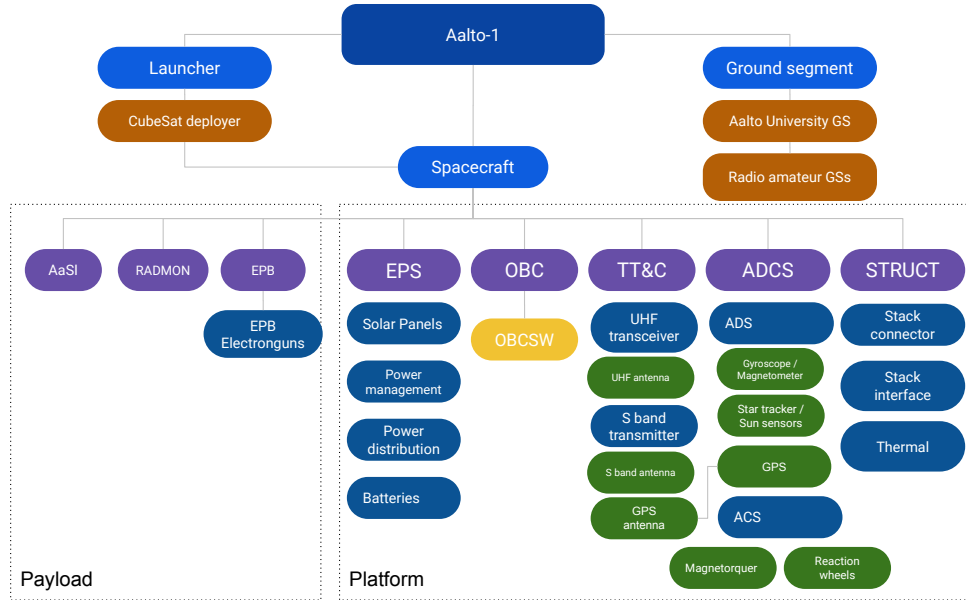


Figure 1: Aalto-1 product tree

168 Steering Group, and Scientific Advisory Board. Under the official project
169 umbrella, student groups developed their own organization for building the
170 satellite and ground segment. Thematic student teams were in most cases
171 oriented on single subsystem development or a single topic. The quality
172 assurance was maintained as a separate independent branch as it is practiced
173 in bigger satellite projects. During the semester, student teams had weekly
174 meetings and decisions were made by team-leader meetings.

175 Thanks to the available funding, it was possible to hire few doctoral
176 students, provide summer trainee positions and occasional master thesis po-
177 sitions with salary (usual practice in Finnish Universities). The doctoral stu-
178 dents formed a backbone of the project which helped accumulate the salient
179 knowledge. Many subsystems were developed as a master thesis project.

180 Payload teams formed separate project structure in their home organi-
181 zations and their team leaders were part of the Scientific Advisory Board.

182 Satellite bus and payload developments were financially independent and ap-
183 plied for funds independently. Several satellite project students made their
184 master thesis with payload team.

185 The project schedule was built to mimic larger space projects, where the
186 main project phases were separated by milestone reviews. The Preliminary
187 Design Review was arranged in November 2011, Critical Design Review in
188 May 2013, Test Readiness Review in May 2015 and Flight Readiness Review
189 in January 2016. Several smaller reviews were arranged along the project.
190 The flight model of the satellite was delivered to Netherlands in May 2016.

191 Review panels were assembled from space technology professionals and
192 CubeSat team members from other universities. Both, documentation based
193 review format (in the beginning of the project) and presentation based review
194 format (towards the end of the project) were used.

195 Flatsat, EQM and FM model policy with fast iterative development model
196 was implemented in the project. A single Flight Qualification Model (FQM)
197 approach was considered in the beginning of the project, but it proved to be
198 impractical. The students were inexperienced and learned most efficiently
199 by making prototypes and hardware versions, therefore rapid iterations and
200 frequent hardware models proved to be more efficient than waterfall design.

201 The main challenge for the project was to find and keep the knowledge
202 in the team during multi year project. Student teams were volatile and doc-
203 umentation often incomplete, despite a requirement for documentation to
204 retrieve study credits. The fact that key persons were hired and commit-
205 ted to the project, helped the project to continue. Constant support by the
206 university and project organization was also highly important. The project
207 was also well aligned with university goals: it was able to produce degrees,
208 research papers and positive publicity. Aalto University procured and fi-
209 nanced, along with few sponsoring partners, also the satellite launch, the
210 first in Finland.

211 3.2. Educational outcomes

212 The student work was incorporated to student's individual studies mainly
213 via special assignments, bachelor and master thesis projects and also as a
214 part of doctoral studies. The main challenge was to align project needs and
215 project documentation with teaching and outcome assessment in situation
216 where most of the work was done in groups.

217 During the project evolved a documentation and reporting approach
218 where project documentation was used for grading and individual contri-

219 butions were assessed by self evaluation and peer reviews. Assessment was
 220 done on the basis of provided snapshot of the evolving documentation and
 221 it was required that the documentation was available for entire project. The
 222 final grade was assigned by supervising professor [34]. Far more than 100
 223 students contributed in the design and development of the satellite. How-
 224 ever, the contribution varied from single semester participation in meetings
 225 to several years of design and implementation. Around 12 Master level and
 226 28 Bachelor level thesis were conducted in the satellite design and develop-
 227 ment activity during the course of the project. By now, also three doctoral
 228 dissertations are defended based mainly on Aalto-1 satellite related topics
 229 [35, 36, 37] and several are still on the way. Additionally more than 10 Mas-
 230 ter level thesis were conducted at partner institutes related to the design and
 231 development of payloads. The outcomes and results were also published in
 232 many scientific conferences and journals in the field. The project gathered
 233 a lot of media attention which led to the awareness of space technology and
 234 small satellites in vast areas [38].

235 Many of the Aalto-1 students became space engineers and scientists at
 236 partner institutions. A subgroup of Aalto-1 students established the ICEYE
 237 company, which builds operates a fleet of Synthetic-Aperture Radar (SAR)
 238 satellites. Another group formed Reaktor Space Lab company, which spe-
 239 cializes on nanosatellite missions.

240 **4. Space Segment design and implementation**

241 The feasibility study and preliminary design analysis proposed 3U Cube-
 242 Sat platform to carry out the mission. The CubeSat platform was selected
 243 because it provided affordable access to space and also available commercial
 244 subsystems for inexperienced team. The payloads were designed concurrently
 245 with the satellite platform, AaSI and RADMON were entirely new designs
 246 whereas EPB development was already started for ESTCube-1 satellite [28].

247 The 3U satellite platform was designed and manufactured mainly by stu-
 248 dents of Aalto University. However, in early stage of the project it was
 249 decided that electrical power system and attitude system should be procured
 250 from commercial provider. The main reason for that was the reliability con-
 251 cern of fresh designs.

252 The satellite design, as shown in Fig. 2, features 3U CubeSat body, 3-axis
 253 stabilization, body mounted solar panels, deployable UHF antennas, several

254 cameras and openings for payloads. Electronics of the satellite is accom-
 255 modated in two electronic stacks, connected by cabling. The Long Stack
 256 features all the avionics and AaSI payload. The Short Stack accommodates
 257 RADMON and EPB. The main reason for this separation was the design
 258 decision to align the EPB reel motor rotation axis with satellite rotation axis
 in spinning mode.

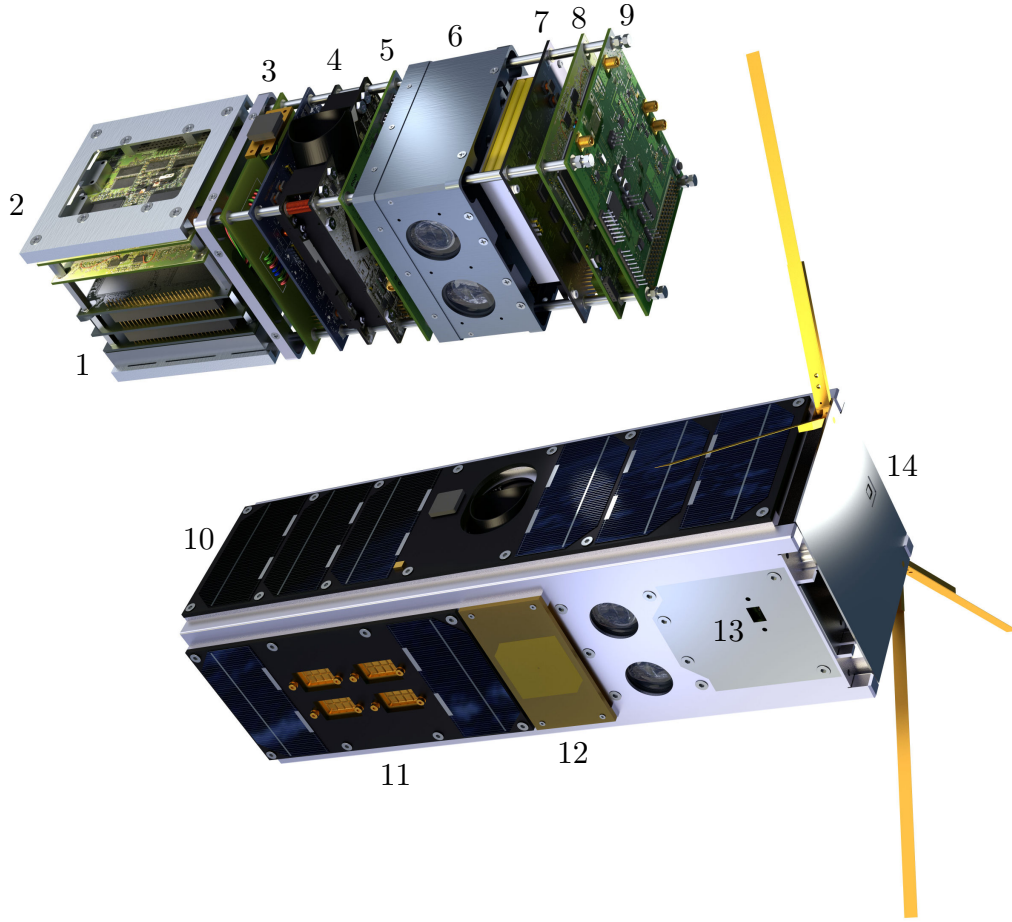


Figure 2: The structure and subsystems of Aalto-1 satellite. The highlighted subsystems are: 1) Radiation Monitor (RADMON), 2) Electrostatic Plasma Brake (EPB) 3) Global Positioning System's (GPS's) antenna and stack interface board, 4) Attitude Determination and Control System (ADCS), 5) GPS and S-band radio, 6) Aalto Spectral Imager (AaSI), 7) Electrical Power System (EPS), 8) On-Board Computer (OBC), 9) Ultra High Frequency (UHF) radios, 10) solar panels, 11) electron guns for EPB, 12) S-band antenna, 13) debug connector, and 14) UHF antennas.

259 The student designed Aalto-1 platform consists of a in-house developed
 260 cold-redundant On Board Computer (OBC) running Linux, Ultra High Fre-
 261 quency (UHF) and S-band radios, a navigation system based on Global
 262 Positioning System (GPS), aluminium structure, solar panels, Sun sensors,
 263 Antenna Deployment System (ADS) and commercially procured Electrical
 264 Power Subsystem (EPS) and Attitude Determination & Control Subsystem
 (ADCS).

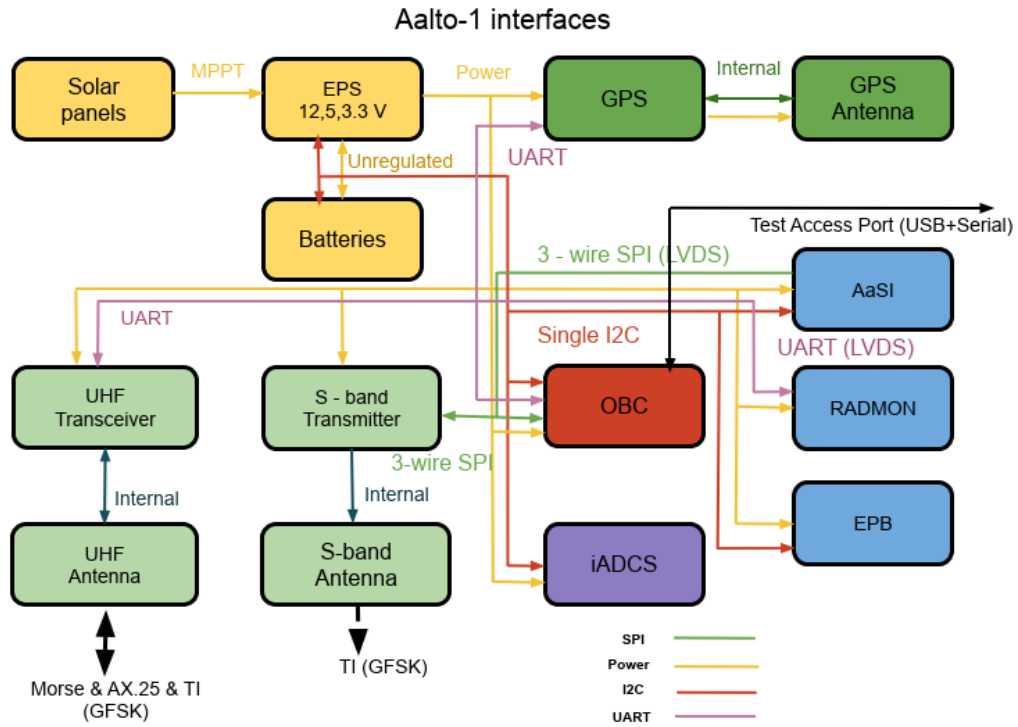


Figure 3: A block diagram of digital, RF and power interfaces

265 An overview of the Aalto-1 power, data and Radio Frequency (RF) inter-
 266 faces is presented in Fig. 3. The power interface provides regulated voltage
 267 levels (3.3 V, 5 V and 12 V) to the satellite avionics and payloads. Several
 268 digital interfaces including Inter Integrated Communication (I²C), Serial Pe-
 269 ripheral Interface (SPI) and Universal Asynchronous Receiver Transmitter
 270 (UART) have been implemented which are controlled by the OBC.
 271

272 5. Payloads

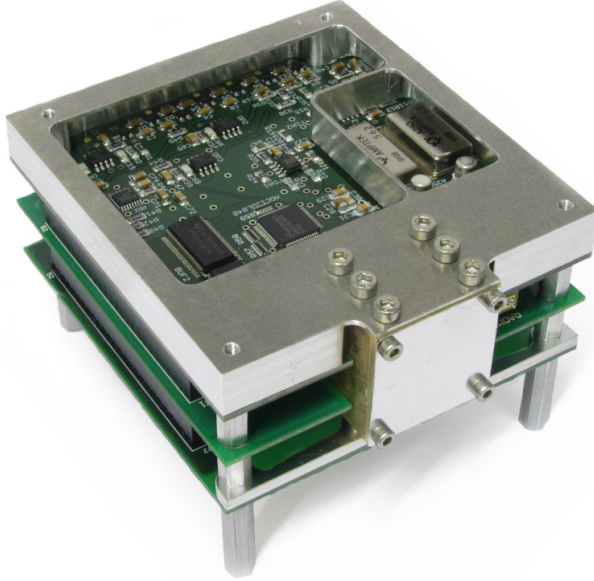
273 The instrument design technique, mass, volume, electrical and mechanical
274 interfaces and key design challenges of the Aalto-1 payloads is described in
275 detail in this and subsequent sections.

276 5.1. Radiation monitor

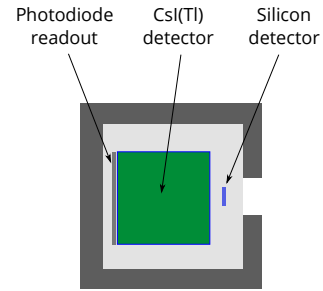
277 The RADMON instrument [39] is a compact low-power radiation monitor.
278 It has envelope dimensions of about $4 \times 9 \times 10 \text{ cm}^3$, a mass of 360 g and a power
279 consumption of 920 mW. The spacecraft supplies both +5 V and +12 V to
280 the instrument. The instrument consists of a detector assembly inside a
281 brass casing, a signal processing board, a digital board, and an electrical
282 power board. Three boards are connected by a 52-pin internal bus running
283 through all of the boards (see Fig. 4(a)). The instrument is integrated in
284 the short stack of the satellite with another bus connector as well as with
285 four spacers placed in the corners of the PCB stack. The bus connector also
286 provides the electrical interface to the satellite.

287 The detector unit consists of a rectangular $2.1 \times 2.1 \times 0.35 \text{ mm}^3$ silicon de-
288 tector and a $10 \times 10 \times 10 \text{ mm}^3$ CsI(Tl) scintillation detector that are enclosed
289 by the brass casing determining the acceptance aperture (see Fig. 4(b)). The
290 casing has an aluminum entrance window that protects the detector stack
291 from low-energy charged particles and photons. The scintillator has a thin
292 polytetrafluoroethylene (PTFE) wrapping on five sides and has a readout
293 photodiode on the sixth side. We have used a Hamamatsu S3590-08 PIN
294 silicon photodiode with dimensions of $10 \times 10 \text{ mm}^2$ and a depletion thickness
295 of about 0.3 mm. The silicon detector has a biased guard ring and a floating
296 one. The passive area of the silicon detector extends to about 0.7 mm around
297 the active spot. Two detectors produce electrical signals for a standard ΔE
298 – E analysis aimed at the determination of particle species and the energy
299 deposited in the detector. A coincidence logic prevents the registration of
300 particles coming from outside the aperture and bremsstrahlung X-rays gen-
301 erated in the brass container.

302 The aluminum window sets thresholds for electron detection at about
303 1 MeV and for proton detection at about 10 MeV. The brass case becomes
304 transparent for protons at about 55 MeV, approximately at the same energy
305 as protons incident through the aperture start to penetrate the scintillator.
306 RADMON registers protons in nine energy channels with threshold energies
307 of 10 – 40 MeV and electrons in five energy channels with threshold energies



(a) The RADMON radiation monitor assembly.



(b) The detector unit.

Figure 4: The assembly compounds three printed circuit boards and a brass container with a detector unit inside. An aluminum entrance window in front of the brass container covers the detector unit. A scintillator with a readout photodiode and a silicon detector are housed within the brass container.

of 1.5 – 12 MeV. The detailed analysis of the instrument response to electrons and protons is described in [40]. The data rate can be adjusted by changing the polling frequency of the instrument. Nominally, science data is collected every 15 seconds and housekeeping data every 60 seconds. This gives a data rate of about 25.4 kBytes per hour, including the packet overhead.

Testing and ground calibrations of RADMON were performed using radioactive sources and a proton beam from the MGC-20 cyclotron at the Åbo Akademi University, Turku, Finland. The maximum beam energy available in the cyclotron was about 17 MeV. The beam was scattered at about 60 degrees from a thin tantalum foil to lower the beam intensity and achieve a low-enough flux for the calibrations. The proton beam energy was step-wise decreased by adding absorbers between the foil and the detector. This setup allowed to successfully calibrate the instrument for the low-energy proton response, and Geant4 [41, 42] simulations were used to extend the proton

322 response over the full energy range. The electron response was monitored
323 utilizing beta particles from different radioactive decay sources.

324 Radiation tolerance of RADMON electronics has been tested in the RA-
325 diation Effects Facility (RADEF) of the University of Jyvaskyla, Finland.
326 The device was tested in a 50-MeV proton beam for total dose up to 10 krad,
327 which it survived without observable degradation [39]. As the device relies
328 on a commercial version of the Xilinx Virtex-4 field-programmable gate ar-
329 ray, we have implemented a triple-redundant memory with active scrubbing
330 running parallel to the normal operations of the instrument [43]. The sys-
331 tem was tested in RADEF to be able to cope with a 50-MeV proton flux
332 of $10^6 \text{ cm}^{-2} \text{ s}^{-1}$, after which the rate of double bit errors became significant
333 [39].

334 The instrument, being integrated into the satellite short stack, is also
335 sensitive to electromagnetic interference. Especially the scintillator detector
336 signal path is affected by the electromagnetic emission of other spacecraft
337 subsystems. This has led to an increase of the noise levels in this signal and
338 the inability to detect at the smallest signal levels, which has increased the
339 threshold of the electron measurements from the nominal 0.7 MeV [39] to 1.5
340 MeV achieved in space [40].

341 5.2. *Electrostatic Plasma Brake*

342 The plasma brake payload is based on the Coulomb drag principle, which
343 is the driving phenomenon behind the Electric Solar Wind Sail (E-sail) inven-
344 tion [44, 45]. The brake itself consists of a 100 meter long tether; a storage
345 reel; a vacuum qualified piezo motor and control electronics for tether de-
346 ployment; a high voltage source; and four electron guns. Once the tether has
347 been deployed, it can be charged with a voltage of either +1 kV or -1 kV,
348 with respect to the surrounding ionospheric plasma. As the satellite moves
349 through the plasma with its orbital speed, the electrostatic interaction be-
350 tween the tether and the plasma introduces a force opposite in direction to
351 the satellite's velocity, thus slowly reducing the orbital speed.

352 Fig. 5 shows the tether reel Printed Circuit Board (PCB)(left panel) and
353 the high voltage board (right panel). These two boards were stacked together
354 to have dimensions of $5 \times 9 \times 10 \text{ cm}$. The spacecraft EPS supplied 3.3 V, 5 V,
355 and 12 V to EPB. Power consumption of EPB depends on the operation
356 mode: launch locks use 1.25 W each for about 20 sec (locks are not released
357 at the same time), high voltage tether system consumes a few hundreds of
358 mW depending on the ambient ionospheric plasma density, and the reeling

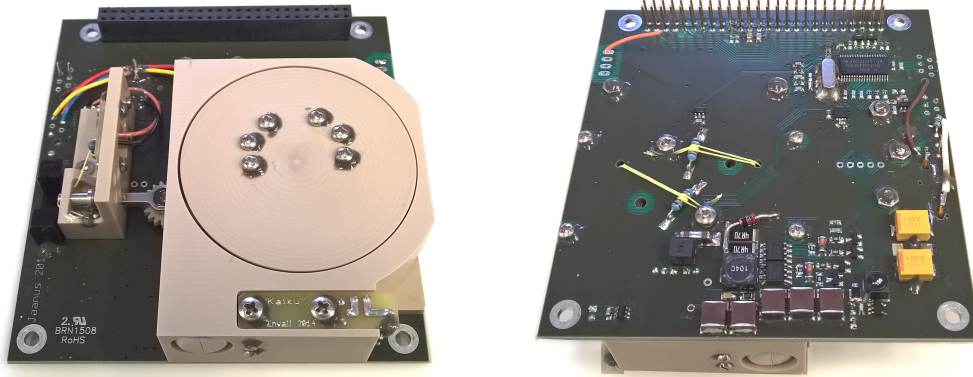


Figure 5: Tether Reel FM board from both sides. On the left, tether reel lock Kieku ready and locked. The black object left from Kieku is the optical feedback Kyylä. The electron guns are located satellite side panel X+, next to the payload which deploys from Z- end of the satellite.

359 system draws 2.3 W during the deployment. None of these tasks are executed
 360 simultaneously, and the tether is not deployed all at once. The data rate is
 361 low throughout the mission as only the tether voltage and current are sampled
 362 with a frequency of 10 Hz.

363 The tether itself is constructed of four aluminum filaments and it is based
 364 on the Heytether geometry [46]. The tether is deployed with the help of
 365 centrifugal force, the satellite must therefore be spinning around a suitable
 366 axis. Once the proper spin mode is reached, the tether reel motor is activated
 367 and the tether is slowly unreeled out to space. At the tip of the tether there
 368 is an aluminum tip mass, whose task is to assist in tether deployment by
 369 increasing the pull force experienced by the tether. On the bottom side of the
 370 reel there is a slip ring serving as the contact point for the high voltage source
 371 through two cantilever spring sliders being the only mechanically redundant
 372 subsystem of EPB. When a positive voltage is applied, one or more electron
 373 guns are activated in order to eject excess electrons and thus maintain the
 374 positive voltage, as the surrounding plasma attempts to neutralize it. In
 375 negative tether voltage mode, the tether gathers positive ions from the plasma
 376 and the conducting parts of the satellite surface collect the same flux of
 377 thermal electrons from the plasma to maintain current balance..

378 The plasma brake payload was tested prior to the system level tests. Vi-
 379 bration tests were carried out to qualify the mechanical components, PCBs,

380 and the reel motor, especially, as the motor was designed for laboratory use.
381 It was noted that the high voltage sliders dug two dents to the slip ring that
382 were able to stop reel rotation. Simple resistor-based launch locks were in-
383 troduced to the bottom side of the reel PCB to keep the sliders apart from
384 the slip ring during the launch. The functionality of the payload was success-
385 fully tested in thermal-vacuum. Furthermore, specific to EPB payload, high
386 voltage tests were made, and the tether outreeling was tested to determine
387 the minimum centrifugal force required for the tether deployment.

388 5.3. Aalto-1 Spectral Imager AaSI

389 The miniaturized spectral imager, AaSI, as shown in Fig. 6, is the main
390 payload of the Aalto-1 nanosatellite. The imager is based on a tunable FPI,
391 which is used as an adjustable passband filter. This enables the imager to
392 acquire images at freely selectable wavelengths. The operational range is
393 500–900 nm and the spectral resolution is 10–20 nm. In addition to the
394 spectral imager, a visible (VIS) spectrum Red–Green–Blue (RGB) camera is
395 included in the instrument [47, 48, 49].

Table 1: Main parameters of AaSI.

Wavelength range	500–900 nm
Spectral resolution	10–15 nm
Field of view	10° × 10° (SPE), 15° × 10° (VIS)
Spectral image size	512 × 512 pixels
VIS image size	2048 × 1280 pixels
Number of spectral bands	6, 25 or 75
Size	97 × 97 × 48 mm ³
Mass	600 g

396 Table 1 introduces the main parameters of AaSI.

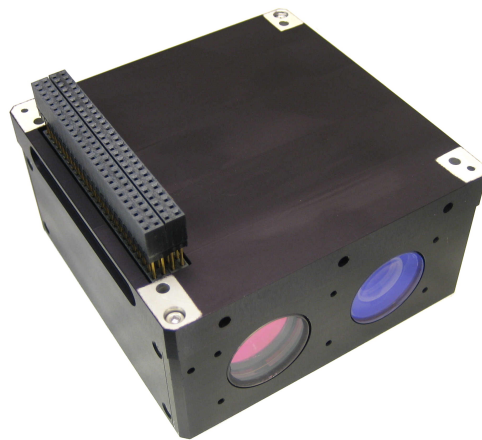


Figure 6: The Aalto-1 Spectral Imager AaSI. The size of the instrument is ca. 0.5 U and it is compatible with the PC104 interface. The instrument has two cameras: a visible spectrum RGB camera (left) and a spectral imager (right).

397 6. Platform

398 The Aalto-1 platform subsystems include an EPS [50], an ADCS [51], a
399 GPS-based navigation system [52], a UHF [53, 54] and S-band [55] radios
400 for Telemetry, Telecommand & Communication (TT&C), and a Linux-based
401 OBC [56, 57, 58, 59]. The electronic subsystems are placed in two circuit
402 board stacks, the Long Stack and the Short Stack, which are connected using
403 a stack interface board. The electronics followed CubeSat electronics format,
404 whereas the bus pin-layout followed PC-104 standard.

405 The design philosophy of the CubeSat platform is a hybrid combination of
406 subsystems developed in-house and commercial products. The satellite struc-
407 ture, solar panels, Sun sensors, TT&C and OBC were fully designed in-house
408 whereas the ADCS and the EPS were procured from commercial partners.
409 The CubeSat structure, antenna and antenna deployment system were also
410 developed in house. The in-house developed subsystems were fully designed,
411 integrated and testing by student teams. The PCB designs were manufac-
412 tured by commercial PCB provider, whereas the component soldering and
413 stuffing was performed in our facility. Special consideration was employed
414 in the design of the critical subset of subsystems, consisting of EPS, OBC
415 and UHF. Redundant parts, fault detection and recovery procedures were
416 added to increase their reliability and fault tolerance. The agile development
417 approaches were followed in the design and verification of the satellite. The
418 development process of the subsystems has been iterative, since the proto-
419 type of each subsystem was developed and qualified in quick iterations. The
420 waterfall verification approach was followed in the Flatsat, EQM and FM in-
421 tegration [60]. The detailed design description of each platform subsystem is
422 presented in the subsequent subsections. The in-orbit performance of major
423 platform subsystems can be read from [27].

424 6.1. Electrical Power Subsystem

425 The EPS ensures the power generation, conditioning, storage and distri-
426 bution to each subsystem and payload [61]. The EPS was procured from
427 a commercial partner Clyde Space (Currently AAC-Clyde). The solar pan-
428 els were designed in-house in order to accommodate the conductive surface
429 requirements of EPB. A block diagram of Aalto-1 EPS is presented in Fig. 7

430 The incident solar radiation is converted to electrical power by the solar
431 panels, developed at Aalto University [62]. The Solar Panel design featured
432 thermally conductive PCB design. In-house made design provided freedom

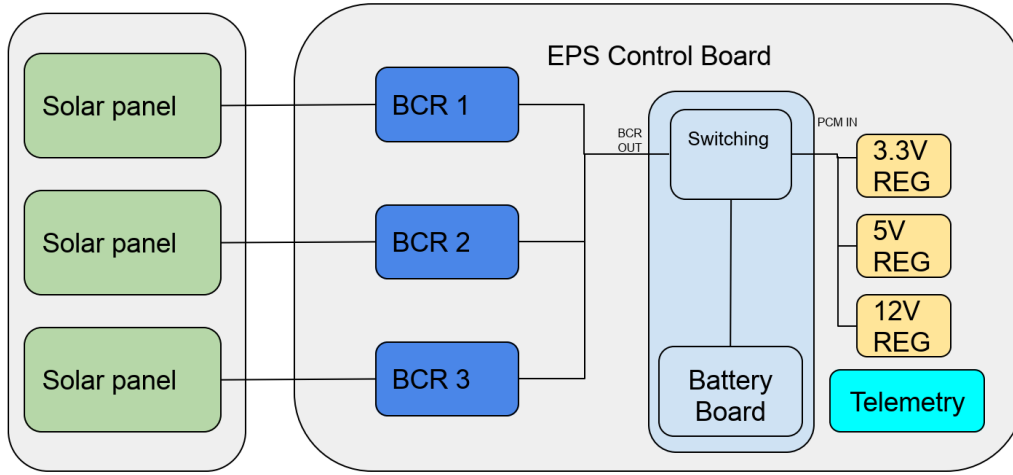


Figure 7: Block diagram of EPS

on sensor and payload location and satellite structure design. The power from panels is transferred to the Electrical Power System Control Board (EPSCB) where Battery Charge Regulators (BCR) convert the input voltage from the solar panels to battery charging voltage (6.2 V to 8.4 V). The Power Conditioning Modules (PCMs) are responsible for regulating the voltages produced by solar panels and the battery unit. The dc-dc converters convert the voltage levels to the ones used by subsystems and distribute the power to the Satellite Bus (SB). The major subsystems have dedicated power lines, which are controlled by switches located on the battery board and accessible through stack connector. The operating voltages, standby and peak power consumption of platform avionics and payloads are provided in Table 2.

The EPS has several safety features implemented for increased reliability of the platform. It monitors the I²C bus lines for inactivity and erroneous behaviour (see Fig. 3), which if detected will cause a power cycle event of the whole platform. Battery power level is monitored as well and the low power mode is activated if depth of discharge is below the critical value. In this mode only the EPS is active, operating the battery charging circuits. Lastly, a timer feature, which starts a 30 minutes countdown after the first EPS power up, was set as a redundant antenna deployment trigger, in addition to the main dedicated countdown timers.

Table 2: Aalto-1 Power budget.

System	Details	Operating voltage (V)	Standby Power (W)	Peak Power (W)
TT&C	UHF	12	0.2	1.55
	S-band	3.3	0	3.5
	ADS	12	0	7
GPS	Active Antenna		0.03	0.03
	GPS Receiver		0.015	0.1
OBC			0.25	0.55
ADCS	Coils and Electronics	5	0.5	1.8
	Sun sensors	5	0	0.06
Payloads	RADMON	12, 5	0	1
	EPB	12, 5, 3.3	2.3	3
	AaSI	12, 5	0	4
Total			3.295	22.59

453 6.2. Attitude determination & control subsystem

454 The ADCS is the most critical subsystem to ensure the required point-
455 ing and spin modes for payloads. Aalto-1 ADCS (iADCS100), provided by
456 commercial partners Berlin Space Technologies (BST) and Hyperion Tech-
457 nologies, consists of an integrated solution of attitude determination sensors
458 and attitude control actuators. The attitude sensors include, gyroscopes,
459 magnetometers and a star tracker. The Sun sensors were developed in-house
460 by Aalto University and integrated to the solar panels[63]. The attitude ac-
461 tuators include magnetorquer rods and reaction wheels. Aalto-1 was the first
462 satellite carrying iADCS100 attitude system and the Aalto students partici-
463 pated in the development. The FM of Aalto-1 ADCS is shown in Fig. 8.

464 6.3. Global positioning system subsystem

465 The GPS subsystem of Aalto-1 is shown in Fig. 9 which contains a Fastrax
466 IT03 GPS receiver and an Adactus ADA-15S patch antenna. When operated,
467 the GPS subsystem consumes approximately 160 mW of power [64, 52]. The
468 main purpose of the subsystem has been to provide more accurate positioning
469 than Two Line Element (TLE)-based solutions, for example, during plasma
470 brake operations. The Fastrax receiver was selected as the manufacturer

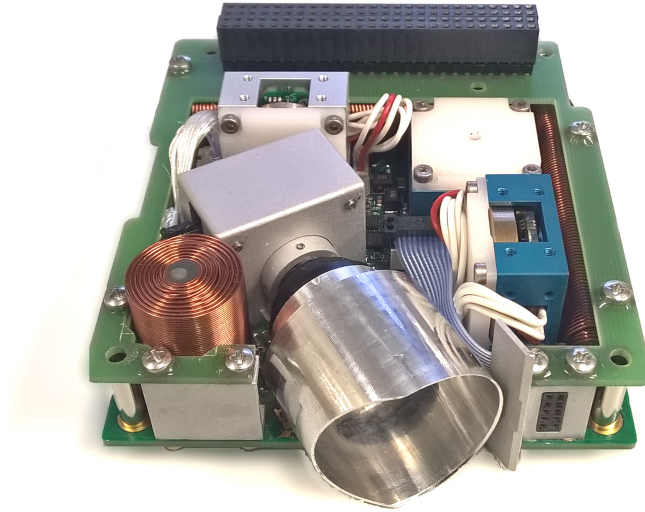


Figure 8: FM of the ADCS iADCS100 with star tracker by Berlin Space Technologies GmbH and Hyperion Technologies.

471 was willing to provide the receiver without the usual altitude and velocity
 472 restrictions [65]. In early 2010s, when a GPS subsystem was included in
 473 the Aalto-1 design, there were not many GNSS subsystems for nanosatellites
 474 available as commercial off-the-shelf products.

475 6.4. Telemetry, tracking & communication subsystem

476 A UHF transceiver was used as the primary radio on the Aalto-1 satellite.
 477 The UHF radio supported transmission power of up to 1.2 W. The unit, as
 478 shown in Fig. 12, is fully redundant, equipped with two cold redundant TI
 479 CC1125-based transceivers and an MSP430 microcontroller (MSP430FR5729).
 480 It is capable of half-duplex bidirectional communication at 437.220 MHz. The
 481 UHF communication system is equipped with two dipole UHF antennas, each
 482 connected to one of the two redundant radios [53]. The OBC software and
 483 the arbiter can perform the switching from active to redundant radio.

484 A UHF antenna deployment system, as shown in Fig. 10, consists of
 485 timer control board for antenna release and two L-shaped doors to keep
 486 the antennas stowed during launch. After the spacecraft is deployed, the
 487 antenna release mechanism burns the dyneema strings thereby deploying
 488 the antennas. Additionally, the redundant timer on the EPS can trigger the
 489 antenna deployment. The deployed antenna configuration is shown in Fig. 11.

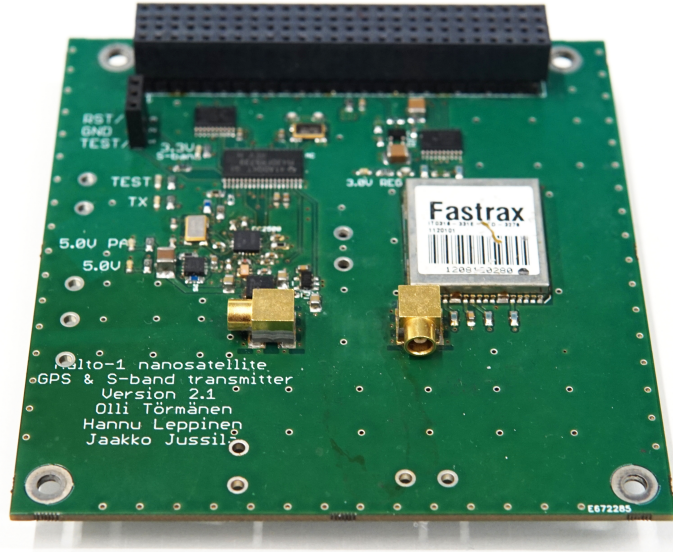


Figure 9: Aalto-1 S-band transmitter and GPS subsystem

490 An automatic UHF beacon is transmitted every two minutes by default. The
 491 UHF beacon containing a static Morse code is transmitted every two minutes
 492 by default.

493 Along with the UHF radio, Aalto-1 has an S-Band transmitter used for
 494 high speed telemetry downlink. Because of regulations, the S-band trans-
 495 mission can be active only above the Aalto Ground Station. The S-band
 496 transmitter featuring a single transceiver (TI CC2500) and a microcontroller
 497 (MSP430FR5739) is shown in Fig. 9. The communication frequency is 2.402
 498 GHz with the design data rate of 500 kbps. The S-band communication sys-
 499 tem uses an in house designed single circular polarization patch antenna. It
 500 also forms a secondary downlink channel [55].

501 6.5. Onboard computer

502 The Aalto-1 OBC consists of two cold-redundant 32-bit AT91RM9200,
 503 microcontrollers from Microware. The architecture hosts a 256-Mbit Syn-
 504 chronous Dynamic Random Access Memory (SDRAM) volatile memory (AS4C16M16S),

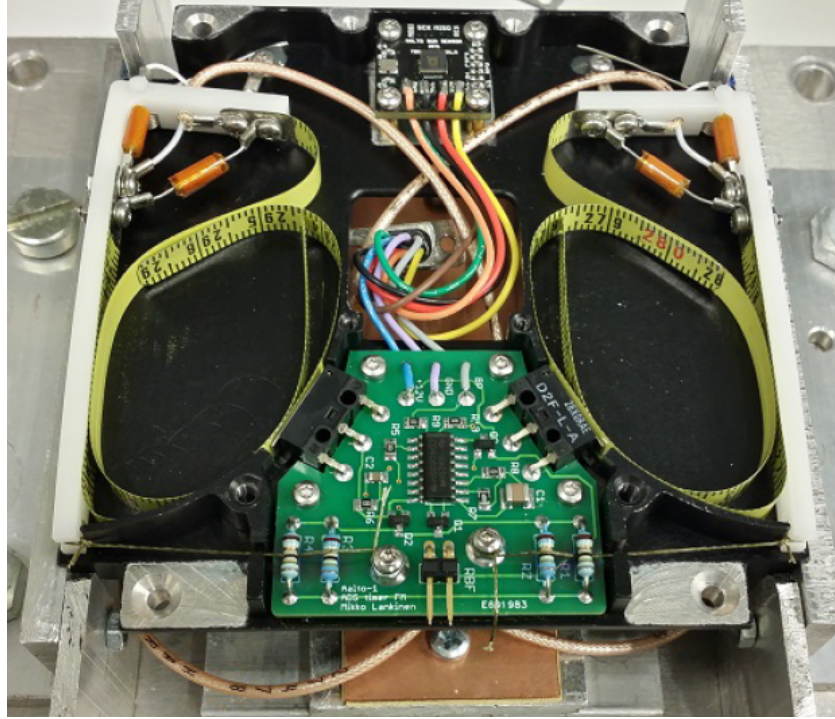


Figure 10: Aalto-1 Antenna deployment system and UHF-band antenna in stowed configuration

505 a parallel/NOR flash (S29JL064J), a dataflash (AT45DB642D), and a NAND
 506 flash (S34ML02G1). These different memories are used to store boot-loaders,
 507 kernel images and file systems. The architecture uses three different bus inter-
 508 faces including I²C, UART and SPI. The UART, SPI and USB are supported
 509 by the processor itself, while I²C is handled by an external controller
 510 (PCA9665).

511 The OBC consists of several components that can be classified as watch-
 512 dogs, the most important one being the arbiter [57]. An MSP430-based
 513 arbiter selects which of the two processors is powered therefore preventing
 514 mission failure due to hard failure of one of the OBCs. In the arbitration
 515 logic, a full reboot is required to switch from the active to redundant OBC.
 516 The switching procedure was set to execute when the arbiter powers up and
 517 it does not receive a heartbeat signal of the active OBC. A further read
 518 on system description, arbitration logic, Failure mode and effects analysis
 519 (FMEA) and error handling procedures, can be found in [56]. A detailed

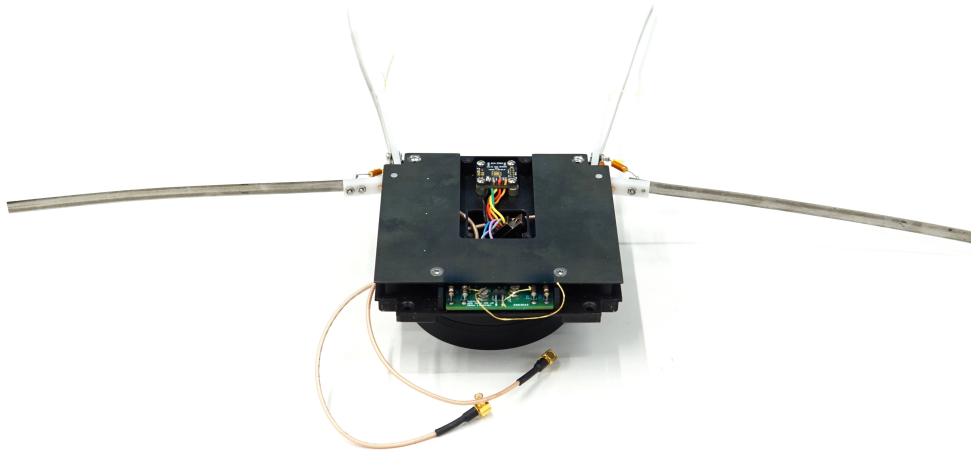


Figure 11: Aalto-1 UHF antenna in deployed configuration

520 block diagram of the OBC representing the data interfaces with payloads
 521 and platform subsystems is shown in Fig. 3 whereas the flight spare model
 522 of the OBC is shown in Fig. 13.

523 The OBC runs the Linux operating system and bash for scripting dif-
 524 ferent command sequences. At the time of its selection in 2010, Linux was
 525 not a common choice for satellite OBCs, but has since become popular in
 526 small spacecraft [66], [59]. Software of the OBC, due to high complexity of
 527 the Linux operating system, has been thoroughly analysed and additionally
 528 strengthened against various identified failure scenarios[57]. The version con-
 529 trol in software development approach was followed in the software design
 530 with regular commits to the Github repository.

531 6.6. Software

532 Several on-board data handling tools have been used in existing Cube-
 533 Sat designs [67]. The Aalto-1 on-board data handling and flight software is
 534 built around applications running on Linux which was quite a new choice for
 535 nanosatellites at the design selection stage. The applications utilise certain
 536 libraries to communicate with satellite subsystems and the satellite internal
 537 data bus.

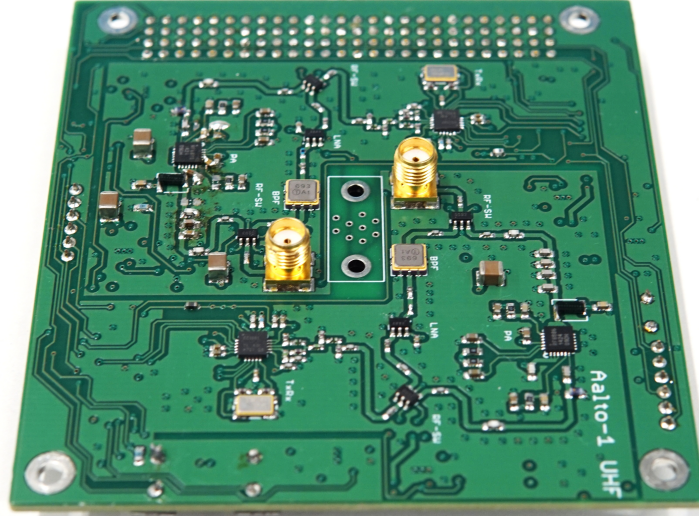


Figure 12: Aalto-1 UHF-band cold redundant transceiver

538 A number of libraries were developed for several subsystems, the most
 539 prominent being libarbitr for arbiters, libeps for communication with EPS,
 540 libicp for communications with subsystems on I²C and libradio and libsb
 541 for radio communication. The detailed description on design choices and
 542 lessons learned on Aalto-1 software design approach can be followed in [59].

543 Linux is a feature full operating system with mature, stable and time-
 544 proven core code base. This simplified the development of flight logic and
 545 utilities. Well known Linux library ecosystem and APIs assured a proper
 546 separation of concerns.

547 Nonetheless, Linux is a complex software which necessitated a thorough
 548 analysis to ensure reliability on the [57]. During the system's boot proce-
 549 dure there is no possibility for intervention and thus needed to be made fault
 550 tolerant by adding an emergency boot procedure with reduced functionality
 551 and less dependencies. A memory storage was divided into sections with pri-
 552 mary and recovery file systems. Unsorted Block Image File System (UBIFS)

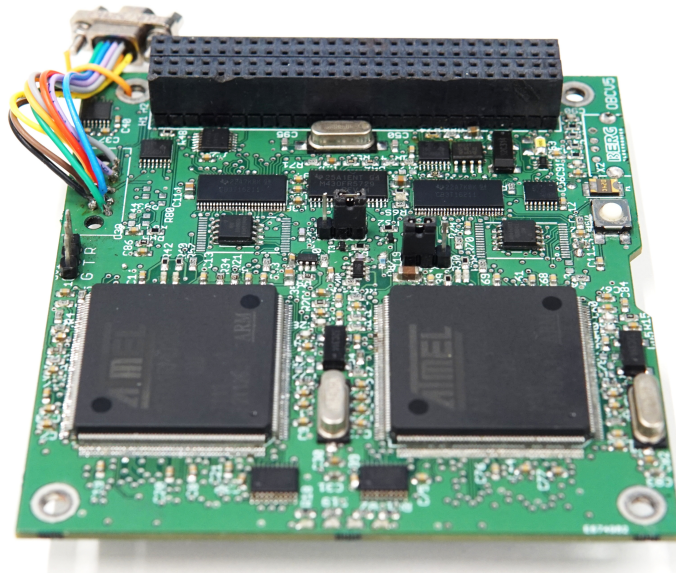


Figure 13: Aalto-1 On-Board Computer's flight spare

553 have been used and it supports wear leveling and due to its use of journals
 554 is power loss tolerant. On the overall system level, a number of software and
 555 hardware watchdog timers are used in conjunction with the arbiter heart-
 556 beat output. Bus and radio communication libraries are strengthened with
 557 appropriate checksum and implementing non-blocking procedures.

558 *6.7. Thermo mechanical subsystem*

559 There are two long and a short standard PC-104 stack to route power and
 560 data signals among platforms and payloads . The long stack, required by few
 561 subsystems, is 2U long whereas the short stack is 1U long. As evident from
 562 Fig.2, the orientation of subsystems on one unit is different than those on
 563 the other two units, therefore a stack interface board was used. The in-house
 564 built structure is compatible with standard dimensions and provides mechan-
 565 ical interface to internal subsystems, solar panels and antenna deployment.
 566 The breakdown of total spacecraft mass is provided in Table 3

Table 3: Aalto-1 mass budget

System	Details	Quantity	Unit mass (g)	Total mass (g)
Structure	3U and harness	1	1180	1180
TT&C	UHF antenna	4	1	4
	UHF transceiver	1	90	90
	S-band & GPS board	1	75	75
	S-band antenna	1	50	50
EPS	Solar panels	4	130	520
	Control board	1	83	83
	Batteries	1	258	258
OBC		1	75	75
ADCS	Coils and Electronics	1	360	360
	Sun sensors	6	10	60
Payloads	RADMON	1	360	360
	EPB	1	300	300
	AaSI	1	600	600
Total			3572	4015

567 The spacecraft used passive thermal control system [68]. The structure
 568 rails were anodized black since it provides optimum emissivity/absorptivity
 569 ratio. The electrically conductive surfaces were masked before anodization
 570 and later chromate coated. The unused areas of solar cell PCBs were gold
 571 plated. In order to increase the thermal conductivity from solar cell to the
 572 structure, indium foil washers were placed in the screw joints. For better
 573 thermal conductivity, many grounding vias were also placed in the solar
 574 cell footprints. The telemetry data of Aalto 1 reveals that the equilibrium
 575 temperature is well maintained inside the spacecraft.

576 7. Satellite integration & testing

577 The model philosophy of the project followed a Flatsat, EQM and FM
 578 approach. This approach was selected mainly due to the fact that all sub-
 579 systems and payloads were new development items and early verification of
 580 them was seen as highly beneficial. Additionally, lessons learned from other
 581 CubeSat projects in other universities often highlighted the importance of
 582 leaving significant amount of time for the integration and testing campaign
 583 on the system level prior to a launch.



Figure 14: Aalto-1 Flatsat model integrating (starting from top left continuing clockwise) AaSI, EPB, UHF, S-band with patch antenna and GPS, OBC and commercial ADCS

Most of the testing performed prior to system level integration was done on subsystem level by each development team. Usually, development kits and other test equipment was utilized rather than other satellite subsystems as their development was performed concurrently by separate teams. The first full interface tests were performed on the Flatsat model which is shown in Fig. 14. A number of interface mismatches were identified and troubleshooted at this stage. While being rather typical for any project with many concurrent developments, earlier testing of the system as a whole, if possible, would probably have saved required redesign and manufacturing effort at the EQM level. As most of the satellite subsystems were in-house developed, making small modifications was however relatively easy at this point of the project.

A full environmental qualification test campaign was performed with the satellite EQM. No major issues were found during these tests, which raised the confidence level on the system level design. However, not all satellite functionalities had been implemented at this point and thus were not fully reference tested prior and during the environmental testing campaign. This

600 left some uncertainties to be fully verified later at the FM test campaign. It
601 also highlighted the importance of thorough reference testing and reporting
602 prior to environmental tests.

603 The satellite FM was built soon after the EQM environmental test cam-
604 paign, including some necessary minor modifications. Testing with the EQM
605 also continued throughout the FM campaign and allowed simpler software
606 development and testing on the system level, as well as tests that could have
607 caused unnecessary stress to the FM. Such tests were, for example, long
608 duration durability testing, outdoor long-range testing and magnetic testing.
609 Some issues were still found using the EQM and it was possible to implement
610 necessary fixes to the FM. One of such issues was related to a component in
611 the Telemetry/Telecommand (TM/TC) radio and may not have been noticed
612 without the long duration durability testing, and could possibly have caused
613 mission failure soon after the launch.

614 A full acceptance test campaign was performed with the satellite FM.
615 The FM testing consisted of pre-built test scripts to command the subsys-
616 tems and receive respective telemetries. No major issues were found during
617 these tests, as was expected thanks to the successful EQM tests. Due to
618 the late readiness of the third party provided ADCS and flight software, it
619 was not possible to perform a thorough enough functional or performance
620 test campaign for it. Testing of the ADCS algorithms was planned to be
621 performed using a hardware-in-loop approach, which did not work as ex-
622 pected through the satellite main communication bus due to communication
623 delays. Rather, access to the ADCS internal sensor and actuator bus would
624 have been required, but it was not possible at that point. This highlighted
625 the importance of early delivery of third-party systems with final and fully
626 tested flight software and should be considered a high risk regarding any new
627 developments by a third party.

628 The Aalto-1 launch campaign started after the assembly, integration and
629 verification stage. A lot of issues were addressed even when the satellite was
630 in the launch pod. As an example, the batteries had become empty and it
631 was a trouble charging them because no such interface was provided on the
632 access port. The batteries in FM were charged with a solar lamp transported
633 to the launch pod.

634 A photograph of the integration of the FM into the commercial orbital
635 deployer is shown in Fig. 15.

636 In the end, the selected model philosophy proved to be a very suitable ap-
637 proach for the project that had significant amount of new development items.

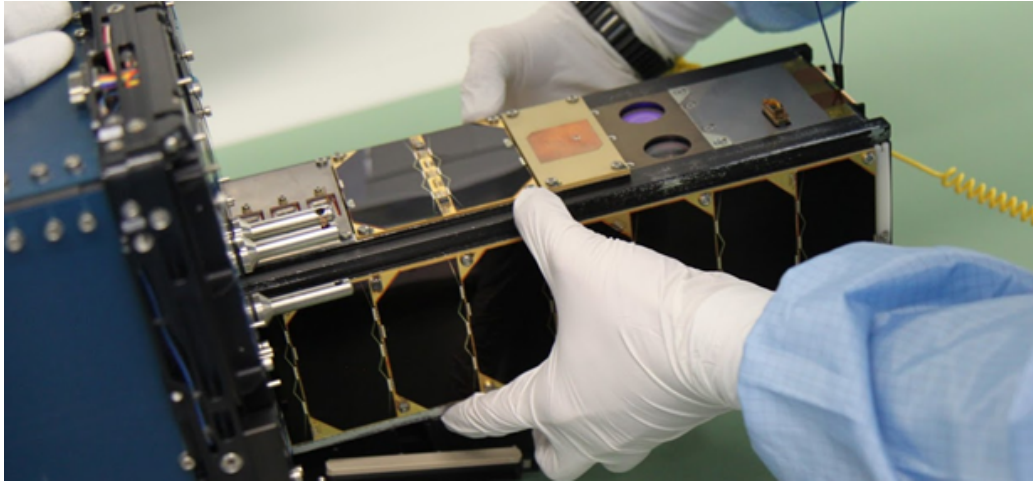


Figure 15: Aalto-1 integration with the deployer

Like in many other CubeSat projects, schedule issues were encountered to perform system level testing in the most ideal and thorough way possible. Emphasis on system level testing from the very beginning of development can solve some of such issues encountered at later stages of the project. A most-viable-product approach, used typically in software engineering, has been followed and determined beneficial in projects after Aalto-1, where the highest importance functions of the satellite are implemented and tested as early as possible on the system level, and later incremented with additional features in order of priority towards the full satellite integration and testing. Such an approach however requires agile in-house development and close cooperation with third parties. Ultimately, the most suitable development approach for any CubeSat project is highly dependant on many aspects, such as the available resources, experience, the number of development items and the usage of in-house or third-party systems. The development approach should be carefully planned only after such aspects have been identified.

8. Ground segment & services

The ground segment originally used an Icom IC-910H radio transceiver and a relay based pre-amplifier that was designed for voice communication. Reception was implemented using an RTL-SDR. Five months after launch, the setup was updated with a newly developed solid state switched pre-amplifier due to problems with the relay-switched pre-amplifier.

659 In 2018, the transceiver was changed to a USRP B200 Software Defined
 660 Radio (SDR). To ease operation of multiple missions from the ground sta-
 661 tion, the digitized radio signal is distributed to multiple programs through
 662 a shared memory buffer. With the OpenWebRX software, the spectrum
 663 between 431 MHz and 439 MHz can be monitored using a web browser.

664 Brushed motors in the antenna rotator caused strong, broadband inter-
 665 ference close to the antenna while rotating during a satellite pass. The issue
 666 was reduced with upgraded rotators and an upgraded controller.

667 Block diagram of the relevant parts of the currently operational ground
 668 station is presented in Fig.16.

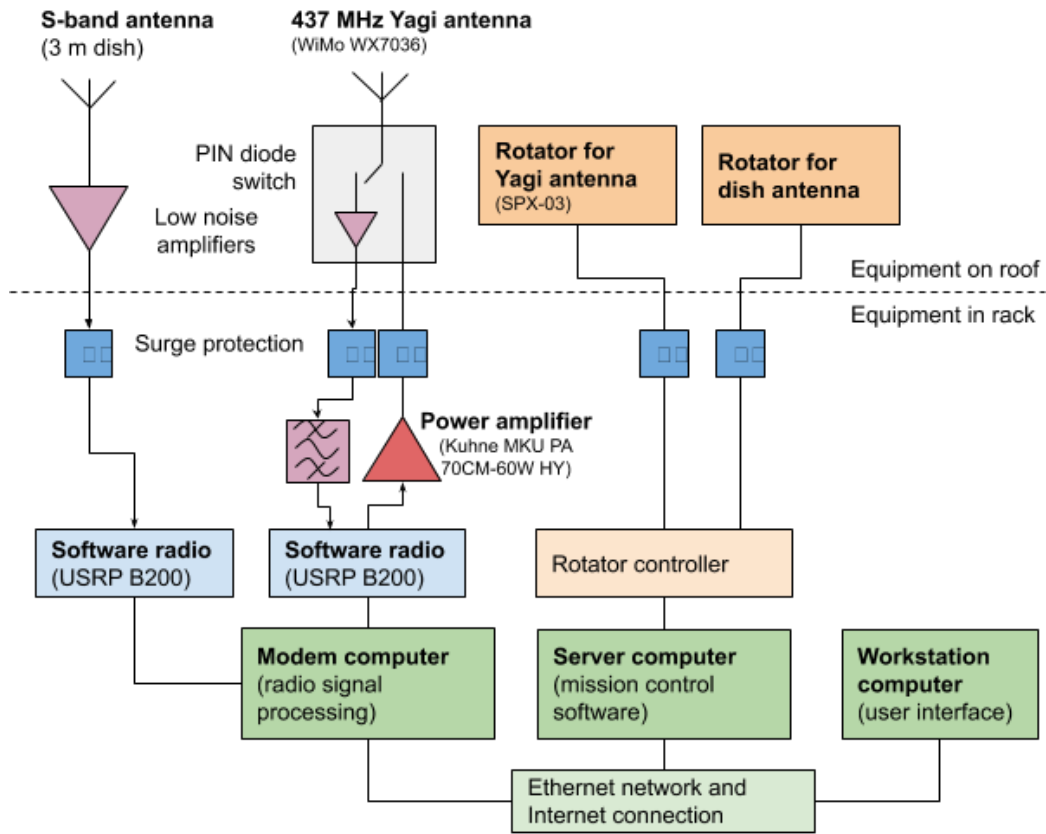


Figure 16: A block diagram of Ground station.

669 The ground segment is controlled using the Mission Control Center (MCC)
 670 software developed by the Aalto-1 team. The back-end is based on a Post-

671 greSQL database that stores every received packet with a timestamp. Fur-
672 thermore, the housekeeping system stores every housekeeping value sepa-
673 rately with timestamp which has proven cumbersome and ineffective due to
674 storage of large set of values in the database. For upcoming missions, we plan
675 to use a database structure that uses one table per subsystem and one row in-
676 cludes an entire housekeeping package of that subsystem. This should reduce
677 query time substantially as not every value has to be queried independently.

678 We have developed a Graphical User Interface (GUI) for the MCC based
679 on Qt. Qt was chosen since the application programming interface (API) does
680 not change very quick which ensures long compatibility in the future. The
681 GUI shows information about the current position of the satellite, satellite
682 passes, received and transmitted packets and housekeeping most recent data.
683 In addition, the history of a single housekeeping value can be plotted.

684 9. Conclusion

685 The Aalto-1 projects key scientific and technological and educational ob-
686 jectives were achieved. The platform and payloads were successfully de-
687 signed, developed and integrated with many student teams getting hands-on
688 learning. The integration, testing, verification and launch activities were suc-
689 cessfully accomplished. The subsystems and payloads demonstrated partial
690 mission success with many lessons learned which have been briefed in an
691 accompanying paper.

692 This project started a new era of space activities in Finland. A number
693 of new space start-ups were founded as an outcome of this project. The
694 (former) Aalto satellites group members have started and joined a number of
695 new missions, such as ICEYE SAR satellite constellation, Aalto-3, Reaktor
696 Hello World, FORESAIL [69], and Comet Interceptor [70]. The Aalto-1
697 design has been beneficial in the space technology curriculum and a
698 source of inspiration for new students in the space technology lab.

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The design and development of multi-payload CubeSat, Aalto 1

The miniaturized hyperspectral imager based on piezo-actuated tunable FPI

The de-orbiting payload consisting of coulomb drag force

The miniaturized low power radiation monitor

The platform design approach consisting of inhouse development and commercial subsystems.

Declaration of interests

☒ The authors declare that they have no known competing financial interests or personal relationships that could have appeared to influence the work reported in this paper.

☐ The authors declare the following financial interests/personal relationships which may be considered as potential competing interests: