ISTANBUL TECHNICAL UNIVERSITY ★ GRADUATE SCHOOL

DUAL CAMERA SYSTEM FOR MEDIUM RESULATION EARTH OBSERVATION IN CUBESATS

M.Sc. THESIS

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Department of Aeronautical and Astronautical Engineering

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ISTANBUL TEKNİK ÜNİVERSİTESİ ★ LİSANSÜSTÜ EĞİTİM ENSTİTÜSÜ

KÜPSATLARDA ORTA ÇÖZÜNÜRLÜKLÜ ÇİFT KAMERALI DÜNYA GÖZLEMİ GÖRÜNTÜLEYİCİSİ TASARIMI

YÜKSEK LİSANS TEZİ

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Uçak ve Uzay Mühendisliği Ana Bilim Dalı

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OCAK 2023

Onur Öztekin, a M.Sc. student of İTU Graduate School student ID 511191224, successfully defended the thesis/dissertation entitled “DUAL CAMERA SYSTEM FOR MEDIUM RESULATION EARTH OBSERVATION IN CUBESATS”, which he/she prepared after fulfilling the requirements specified in the associated legislations, before the jury whose signatures are below.

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Date of Submission : 30 December 2022
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To my family and friends,

FOREWORD

Thanks to my advisor Prof. Dr. Alim Rüstem Aslan for his support. Special thanks to MSc. Boğaç Karabulut for his support as my senior for many years.

Thank you to all my collages in Space Systems Design and Testing Laboratory for their invaluable contributions in this project. Special thanks to Eng. Aybüke Ağırbaş, MSc. Egemen Çatal, Eng. Eser Gül, and Eng. Kaan Sarıca.

2023 Onur Öztekin
Astronautical Engineer

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ABBREVIATIONS

GSD	**:** Ground Sampling Distance
ADCS	**:** Attitude Determination and Control System
CCD	**:** Charged Coupled Device
CFA	**:** Color Filter Array
CMOS	**:** Complementary Metal–Oxide Semiconductor
CoG	**:** Center of Gravity
CoTS	**:** Commercially off the Shelf
dTDI	**:** Digital Time Delay Integration
EPS	**:** Electrical Power System
EQM	**:** Engineering Qualification Model
FPA	**:** Focal Plane Array
FPGA	**:** Field Programmable Gate Array
GPS	**:** Global Positioning System
ILMH	: Integrated Lens Mount Holder
iXRD	**:** improved X-ray Detector
LEO	**:** Low Earth Orbit
NDA	**:** Non-Disclosure Agreement
OBC	: On Board Computer
RAM	**:** Random Access Memory
SNR	**:** Signal to Noise Ratio
SPI	**:** Serial Peripheral Interface
SSDTL	**:** Space Systems Design and Test Laboratory
SU	**:** Sabancı University
TDI	**:** Time Delay Integration
UHF	**:** Ultra High Frequency
UoS	**:** University of Sharjah
VHF	**:** Very High Frequency

SYMBOLS

$\boldsymbol{a}$	**:** Angle of view in degrees (for width or height)
λ	**:** Wavelenght of the light
Pa	**:** Pascal

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DUAL CAMERA SYSTEM FOR MEDIUM RESULATION EARTH OBSERVATION IN CUBESATS

SUMMARY

Here in this thesis design, testing, manufacturing, assembly, image processing, and some software information for a dual camera imaging system for medium resolution earth observation CubeSats is given. System is designed to use CoTS imagers, and lenses. CoTS components are tested for their suitability for space usage. Selection criterias for component selection is detailed, factors such as mechanical design of the components, their cost, and availability is important. Imager types, their working principles are explained. Optical limits, and selection criteria with lens and imaging sensor compatibility is also discussed. Design choices for space suitability, environmental testing suitability and specific mission requirements are explained. Effect of these requirements and their direct results have been explained in the thesis. Assembly procedures for system is explained in detail, also satellite assembly procedure is mentioned to some extend.

SharjahSat-1 satellite, its mission and its configuration is shared in this work, to further explain design choices and boundary conditions created by rest of the mission is explained.

Material selection, manufacturing details, some critical tolerances and used thread types are shared.

Software structure for ground testing software, satellite ground station software, and satellite firmware, ground testing board firmware and concept of operations is shared in this work.

Tests done on the EQM and FM model of the payload have been explained in this work. Testing logic, procedures, and their results have been explained. Tests are done to determine performance of the system, and suitability of the system to a space environment, and launch conditions. TVAC, vibration, and optical tests are explained. A failed design is also shown and explained. Testing done on the system showed that, system is indeed suitable for space craft usage, within its temperature ranges. Too high or too low temperatures results in image focus shifting, and for high temperature too much image noise. Some possible future work, and future tests are also mentioned.

KÜPSATLARDA ORTA ÇÖZÜNÜRLÜKLÜ ÇİFT KAMERALI DÜNYA GÖZLEMİ GÖRÜNTÜLEYİCİSİ TASARIMI

ÖZET

Bu tezde KüpSatlar için orta çözünürlükte çift görüntülüyücüye sahip kamera sisteminin tasarımı, testleri, üretimi, görüntü işlemesi ve yazılımı hakkında bilgiler verilmiştir. Sistem piyasadan hazır temin edilebilen lens ve görüntüleyiciler kullanılarak tasarlanmıştır. Sistem parçalarının seçiminde göz önünde bulundurulan etkenlere değinilmiş ve açıklanmıştır. Bu etkenler genel olarak parçaların mekanik tasarımı, temin edilebiliriği, yüksek vakum ortamına uyumlu olma ihtimalleri, titreşime olan dayanımları, fiyatları, ve kullanım kolaylıkarı olarak değerlendirlimiştir. Sistemin tasarımı sırasında ana uydu görevi dolayısı ile ortaya çıkan limitleyici etkenler, ana uydu görevi yüzünden kaynaklanan limitler ve bunların ortaya çıkardığı isterler açıklanmıştır. Tez içerisinde bu isterlerin tasarım üzerinde oluşturduğu etkiler açıklanmıştır. Uzay ortamından ve roket sırasında oluşan yüklerden dolayı sistem üzerinde dikkat edilmesi gereken konular ve bunların tasarıma olan etkileri ayrıca açıklanmıştır. Uzay ortamının yüksek vakumunun malzemelere olan etkileri açıklanmıştır ve bu etkilerin sonuçları ve sistemler üzerindeki sınırlayıcı etkileri açıklanmıştır.

Piyasadan hazır temin edilebilen parçaların seçimlerinden sonra bunların uzay uyumluluğu için çevresel testleri yapılmış, testlerin mantığı ve sonuçları açıklanmıştır. İlk yapılan tasarım için denenen ve başarılı olunan parçalardan oluşan görev yükü tasarımının optik performans bakımından başarısız olması sonucu, gidilen parça değişiklikleri ve bunların tekrar TVAC testinde denenmesi anlatılmıştır. Bu testin sonuçları değerlendirilmiştir. Bu 2. yapılan çalışmanın optik performansı test edilmiş ve yeterli bulunduğundan dolayı tasarıma devam edilme kararı alınmıştır. Parçaların kendi başlarına testlerinden sonra SharjahSat-1 uydu görevi detaylandırılmış ve bu görevin gerçeklenebilmesi için ana görev yükünün ihtiyaçlarından doğan ve tasarımı yönlendiren sebepler açıklanmıştır. Bu sepeler doğrultusunda sistemin tasarımına deyinilmiş ve detayları verilmiştir. Tasarımın üretim açısından ihtiyaçlarına deyinilmiş detaylar verilmiştir. Tasarım için kritik olan tolerans ve diş tiplerine değinilmiştir. Tasarımın montaj aşamaları açıklanmış detaylandırılmıştır. Montaj sırasında parçalar üzerinde yapılması gereken değişikliklerin sebepleri açıklanmış ve yapılması gereken değişiklikler gösterilmiştir. Montaj sırasında görüntüleyicilerin odak noktalarının nasıl ayarlanması gerektiği belirtilmiş ve optik labarotuar da yapılan bu işlem detaylandırılmıştır. Bu aşamadan sonra mühedislik modeli ve uçuş modellerinin testlerindeki farklılıklar açıklanmıştır. Mühendislik modeli için yapılan titreşim ve TVAC testlerinden sonra optik labarotuarda yapılan odak testi açıklanmıştır ve bu testin neden uçuş modeli için yapılmadığı belirtilmiştir ve sonuçlar sunulup yorumlanmıştır. Uçuş modeli için yapılan TVAC ve titreşim testleri açıklanmıştır. Bu testlerden sonra görev yükü üzerinde yapılan diğer çalışmalar ve hazırlıklar açıklanmıştır.

Tez içerisinde görüntüleme sensörlerinin çalışma mantıkları açıklanmıştır ve bu cihazların aldıkları ışık bilgisini kullanılabilir görüntü formatlarına nasıl dönüştürdükleri detaylandırılmıştır. Açıklanan görüntü sensörleri arasından yapılan seçimin sebepleri belirtilmiş ve kullanılan görüntüleme kartının tasarım ve çalışma mantıkları açıklanmıştır ve iç yazılımı hakkında bilgi verilmiştir. Görev yükünün denenmesi için geliştirilen çeşitli geliştirme kartları ve kartlar için kullanılan gömülü yazılım ve bu gömülü yazılımı denetlemesi için kullanılan bilgisayar yazılımlarına değinişmiştir. SharjsahSat-1 için kullanılan gerçek zamanlı işletim sisteminin çalışma mantığı açıklanmıştır ve bunun uydunun sahip olduğu bilgisayar ve depolama sisteminin çalışma mantığı açıklanmıştır. Uydu ile kullanılan yer istasyonu yazılımı gösterilmiş ve sahip olduğu yeteneklere değinilmiştir. Çekilen fotoğraflar yanında kayıt edilen "metadata"nın içeriğine değinilmiş ve açıklanmıştır. Bu kaydedilen verinin olası değerinden bahsedilmiştir. Uçuş modelinin yazılım testi sırasında yapılan denemelerden bahsedilmiş ve veri indirme yöntemleri açıklanmıştır.

Görüntünün kaydedilmesi için gereken ve yapılabilen ayarlar açıklanmıştır ve bu ayarların yörünge parametreleri ile ilgili ilişkileri belirtilmiştir. Sensör üzerinden kaydedilen işlenmemiş verinin nasıl anlamdırılabileceği ve kayıt edilen verilerin sahip olduğu anlam açıklanmıştır. Sistem için uçuş öncesi yapılan kontroller ve testler açıklanmış ve çalışma sonlandırılmıştır.

1. INTRODUCTION

1.1 CubeSats

A CubeSat is a class of satellites that adapt a standard size and form factor defined as ‘U’. 1U is a 10cm cube with a mass of up to 2kg [1]. Size of a CubeSat can be up to 27U [2]. Due to this relatively large range CubeSats can either be a nano (1–10kg) or a micro (10-100kg) satellite [3]. A CubeSat, just like other satellites, includes an onboard computer (OBC), electronic power system (EPS), communication equipment (transmitter and receivers) at various radio frequency (RF) bands. Also depending on the mission, a CubeSat could include an attitude determination and control system (ADCS), propulsion system etc. These subsystems are there to support payload of the satellite. CubeSats are design to reduce cost and development time, increase accessibility to space [1]. Just in 2022 around 600 CubeSats have been launched [4]. CubeSats are usually launched in to low Earth orbits (LEO) [4], but there are plans to deploy CubeSats into lunar orbits [5].

1.1.1 CubeSat research at İTÜ

CubeSat research at İstanbul Technical University is done by Space Systems Design and Testing Laboratory (SSDTL). It is one of the first makers of a CubeSat in the world, and first in Türkiye with ITUpSAT-1 as shown in Figure 1.1. Since establishment in 2007, 7 CubeSats have been designed and launched. The imaging system explained in this document is designed for this satellite, as seen in Figure 1.2.

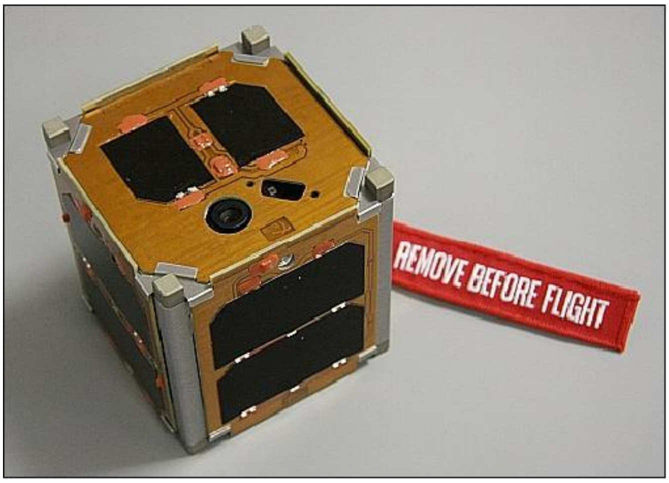

Figure 1.1 : Image of ITUpSAT-1.

Figure 1.2 : Image of SharjahSat-1 flight model.

1.2 Satellite Camera Information

1.2.1 Sensor types

Most common imaging sensors used are either charge-coupled device CCD or complementary metal oxide semiconductor CMOS. These can be line scanner or area scanner. These sensors are accompanied by filters in different spectral bands to achieve imaging requirements. Main difference between these sensors are in CCDs photons are converted to voltage at the output node, on CMOS they are converted to voltage at the pixel [6, 7]. An image explaining their working principles can be seen at Figure 1.3 and Figure 1.4 respectively.

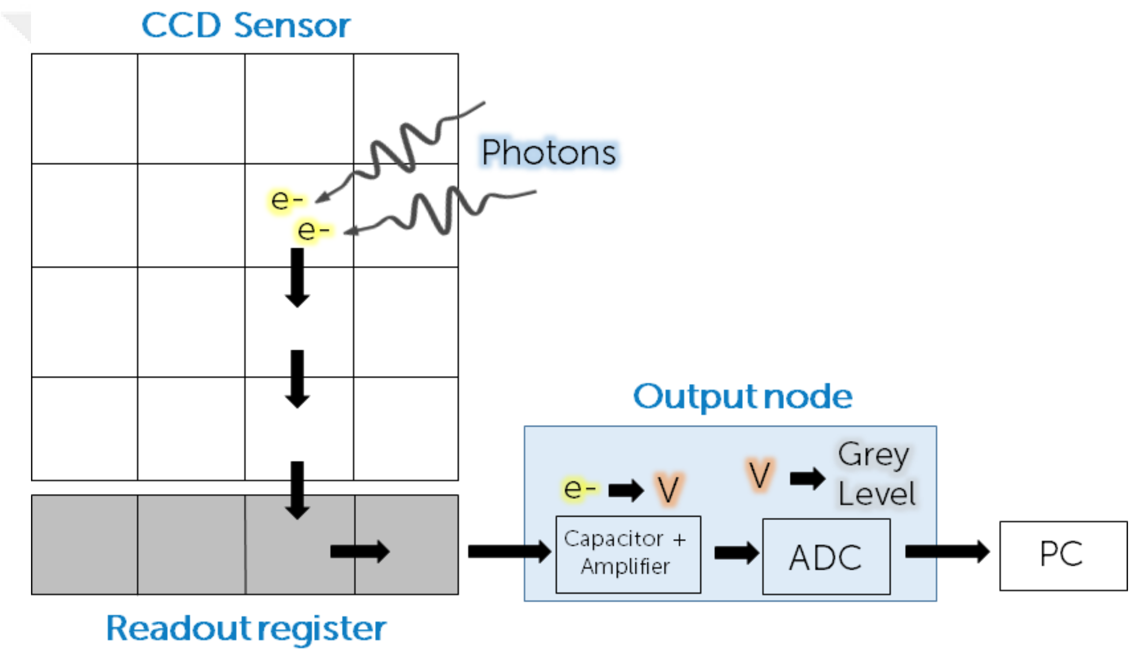

Figure 1.3 : CCD sensor working principle [6].

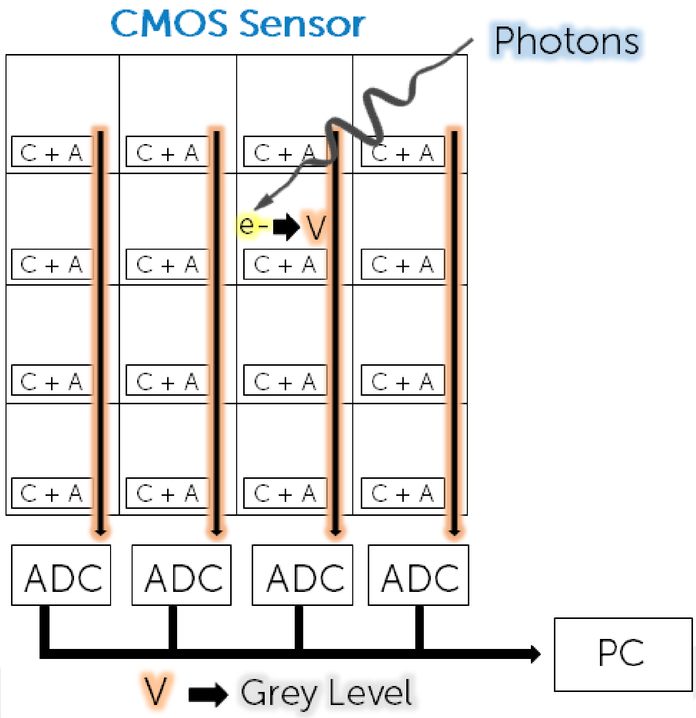

Figure 1.4 : CMOS sensor working principle [6].

1.2.2 Along-track imaging

Along track imaging utilizes satellites orbital motion to scan the ground in line by line fashion to create an image as shown in Figure 1.5. Depending on the design imager could also include its own mechanical components to move optics or its sensor.

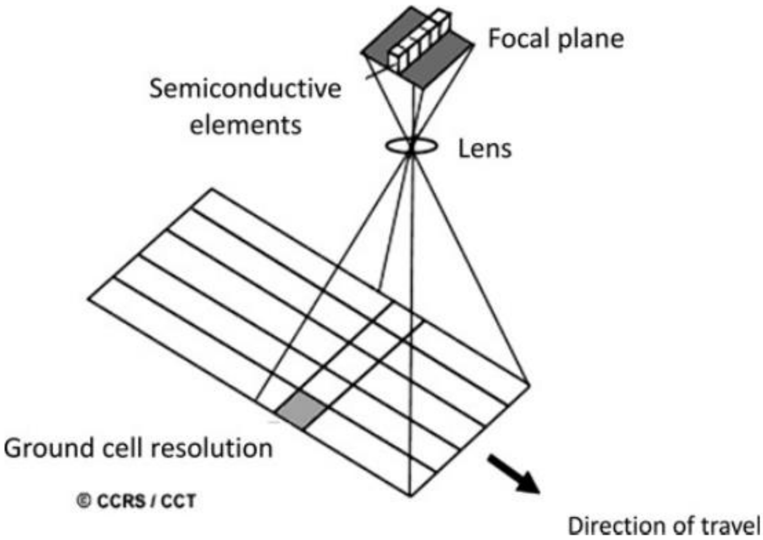

Figure 1.5 : Along track imaging [8].

Used imagers could be in the shape of a line as shown in Figure 1.6, and a satellite could have more than one of them to capture data in different spectral band [8, 9].

Figure 1.6 : a line CMOS sensor [9].

Another method is to use a full area imaging sensor (as shown in Figure 1.7) with a filter of different spectral bands applied to it. This way one image sensor utilizing satellites orbital movement can have multiple spectral bands while also ability to support digital time delay integration (dTDI) [10].

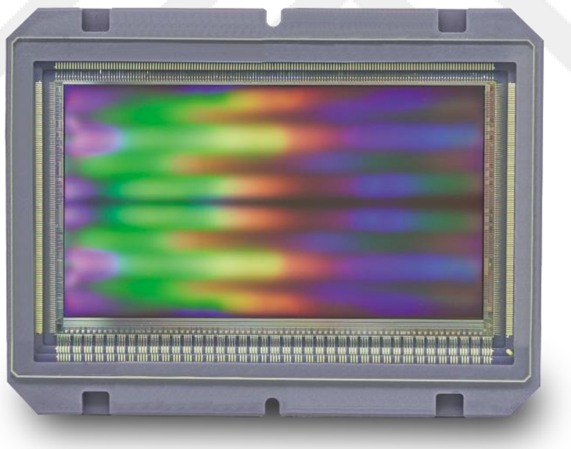

Figure 1.7 : a full area sensor [11].

1.2.3 Across-track imaging

Across track imaging also utilizes satellite’s motion, but it also scans the ground in perpendicular to motion axis too as shown in Figure 1.8.

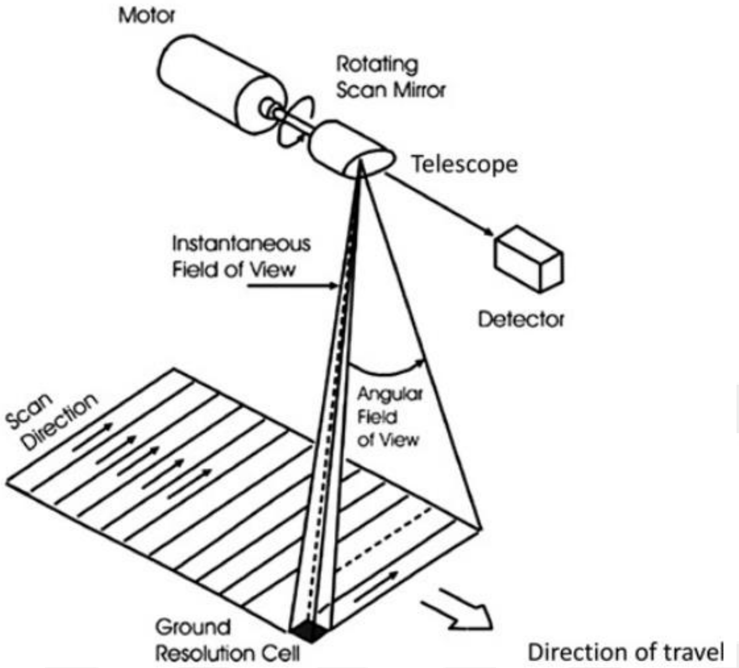

Figure 1.8 : Across track imaging [8].

1.2.4 Time delay integration

Time delay integration is a solution for taking images of quick moving subjects using low integration times. Concept is that acclamation of cumulative exposures of the same object that is moving linearly under the imager. This allows for better signal to noise ratio (SNR) [12, 13]. This have opened new possibilities and higher quality imaging for CubeSat applications [10, 14]. An example image for a full sized line scanning TDI enabled can be seen in Figure 1.9.

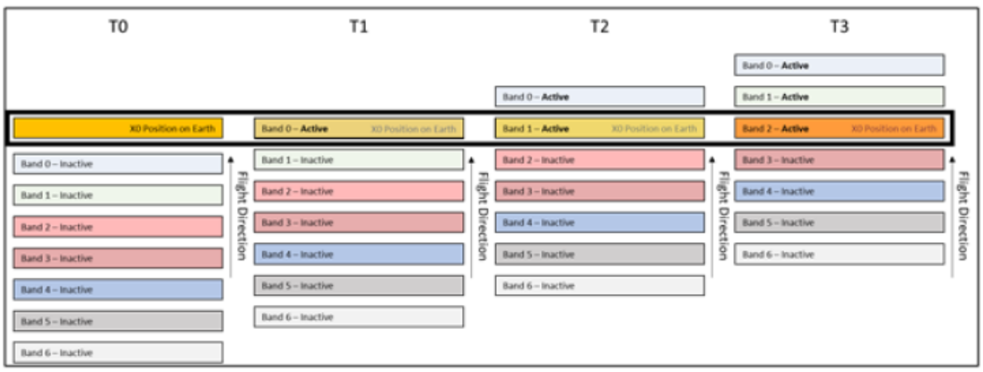

Figure 1.9 : Line scanning TDI sensor [10].

1.2.5 Snap-shot imaging

This type of imaging is most like what is being used in consumer electronics. A sensor such as Figure 1.7 is used with a color filter array (CFA) on top of it. When image is taken whole sensor is read, and then with a demosaicing algorithm raw data is converted to image data.

1.3 Nanosatellite Camera Systems

1.3.1 Custom imagers

These imagers have been designed for their respective mission requirements. Although there are commercially available imagers for CubeSat usage, for some missions imagers are needed to be designed for that specific mission.

1.3.2 Commercially available imagers

Some of the commercially of the shelf imagers are as follows. An imager from GOM space with 1U size and 30m/px [15]. 3 different offerings from Simera with 1U, 1.5U and 12U sizes with 30m/px, 4.75m/px and 1.5 m/px respectively. These available imagers show that there are not many medium resolution imagers with small sizes.

2. BACKGROUND

2.1 Sharjahsat-1 Mission

SharjahSat-1 is a 3U+ sized CubeSat that is developed by Sharjah Academy for Astronomy, Space Sciences, and Technology (SAASST), University of Sharjah (UoS), Space Systems Design and Test Laboratory of Istanbul Technical University (SSDTL-ITU), and Sabancı University (SU). Satellite bus, and camera payload is developed by SSDTL, while main payload, improved X-ray detector (iXRD), is developed by SU. SSDTL is also responsible for mission design, instrument design, bus design, manufacturing, testing, and programming of the satellite. Besides from 2 payloads, satellite houses an EPS, battery, S-band transmitter, VHF-UHF modem, UHF beacon, OBC, interface board, real time clock, ADCS system with a star tracker, GPS, and solar panels which can be seen in Figure 2.1. Currently satellite awaits launch in deployer.

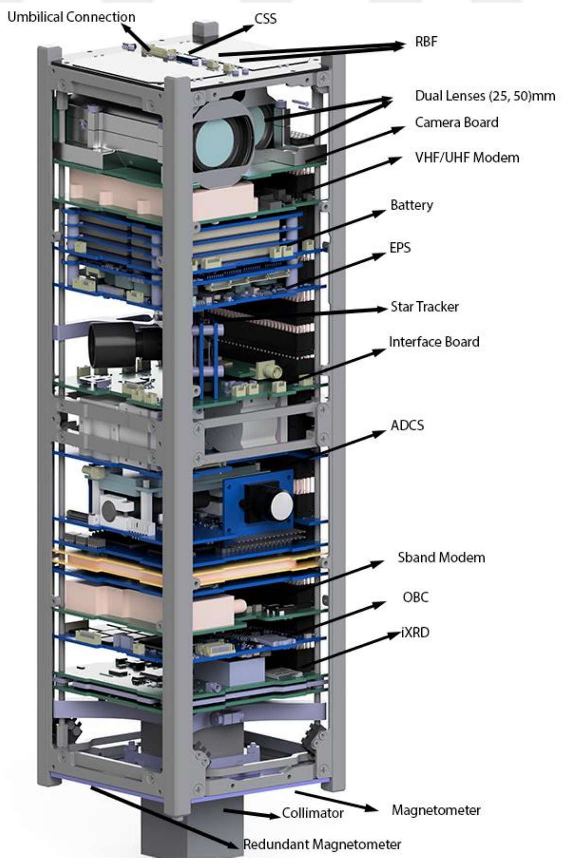

Figure 2.1 : Top down of SharjahSat-1.

2.1.1 Main payload

The SharjahSat-1 mission's primary scientific objective that is iXRD is to observe bright and hard X-ray sources in our Galaxy and the solar coronal holes [16]. This is done by utilizing CdZnTe crystal as the active element that is sensitive to hard X-rays in 20-200 keV range. The crystal is surrounded by a tungsten collimator with square holes to bring field of view to 4.26 degrees. On the top of the collimator there is an optical blocker made from 0.3mm aluminum which is attached to collimator by adhesives and polyamide tape. There is also a tungsten mid-shield shield between "motherboard" and "daughter" board to stop radiation from the Earth's atmosphere reaching to the crystal [16]. There are also aluminum shielings covering both boards against interference from subsystems of the satellite. These all can be seen in Figure 2.1. Because of the nature of the instrument, it requires relatively large volumes and large masses on the payload, which is already limited.

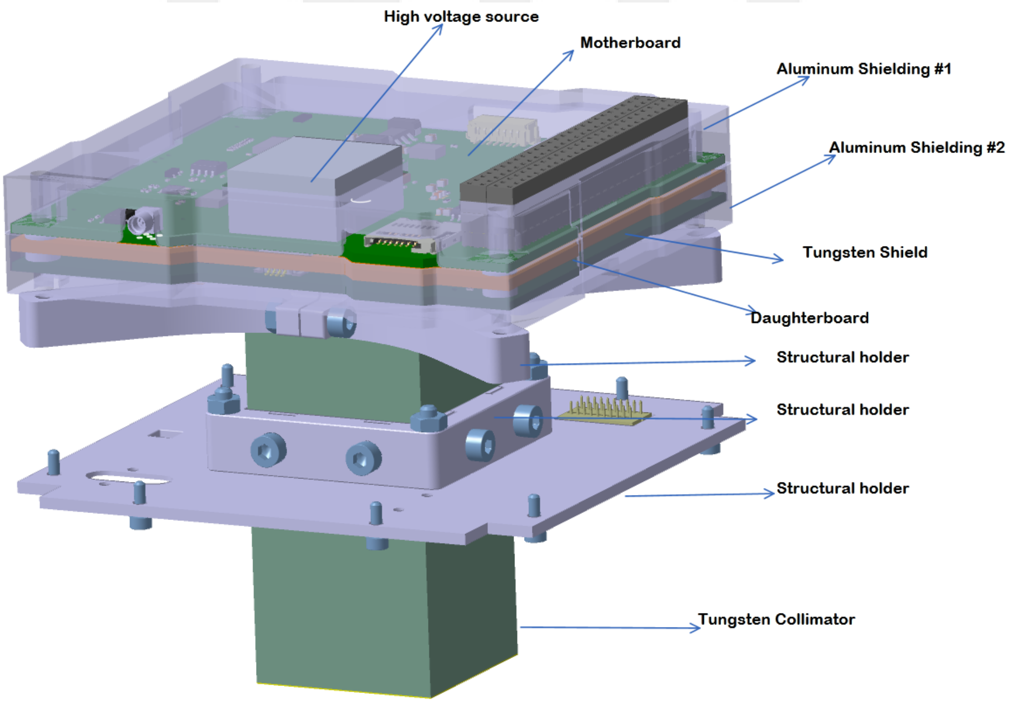

Figure 2.2 : Top down of iXRD.

2.1.2 Secondary Payload

In this thesis design, and testing of the secondary payload will be explained. The end product is an imaging system that uses dual lenses and sensors to capture images in visible spectrum. Design uses CoTS parts for its sensors and optics. Images of the imager could be seen in Figure 2.3, Figure 2.4 and Figure 2.5. Design is very much optimized for volume usage and specifics of the SharjahSat-1 mission.

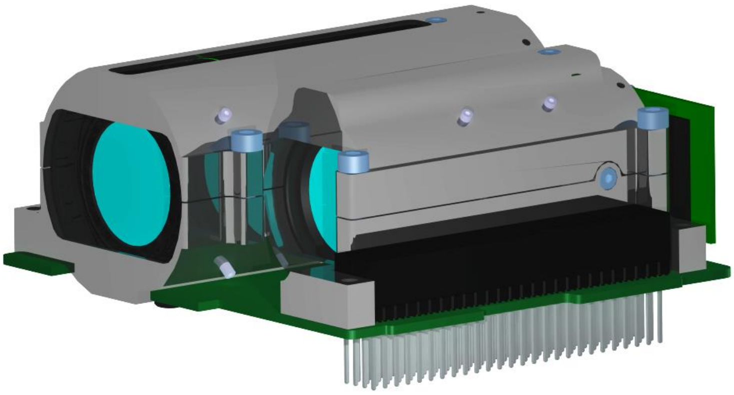

Figure 2.3 : Rendering of the imaging system.

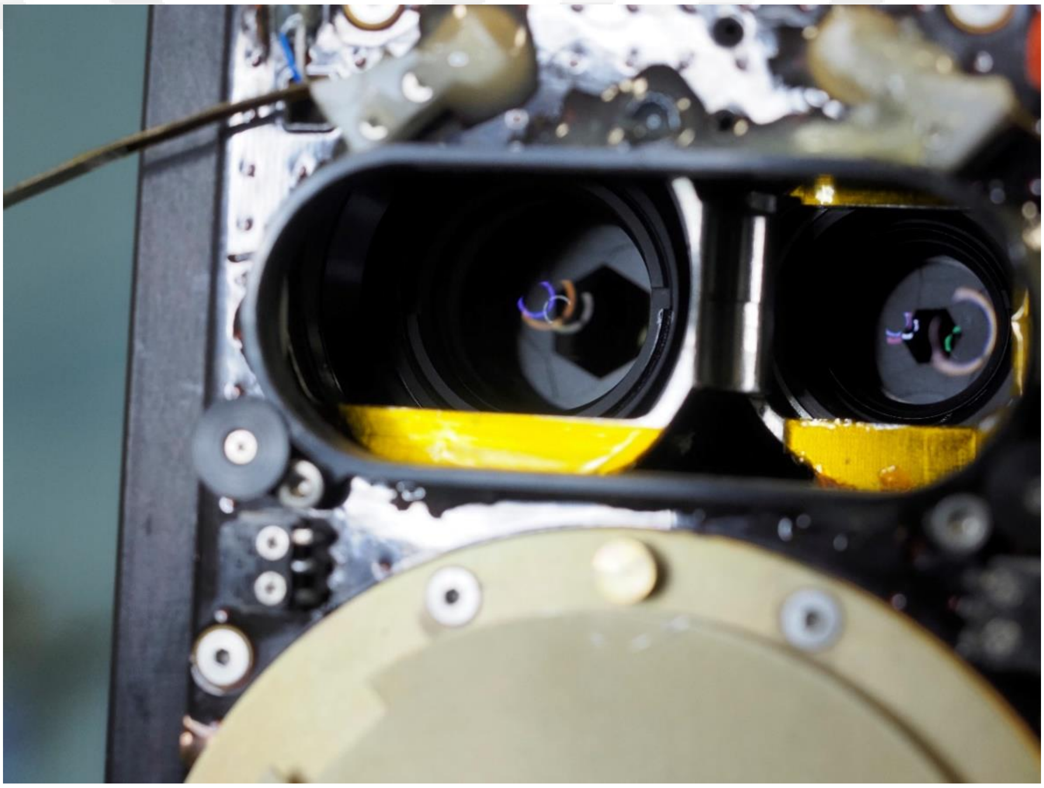

Figure 2.4 : Image of the optics on the flight model of SharjahSat-1.

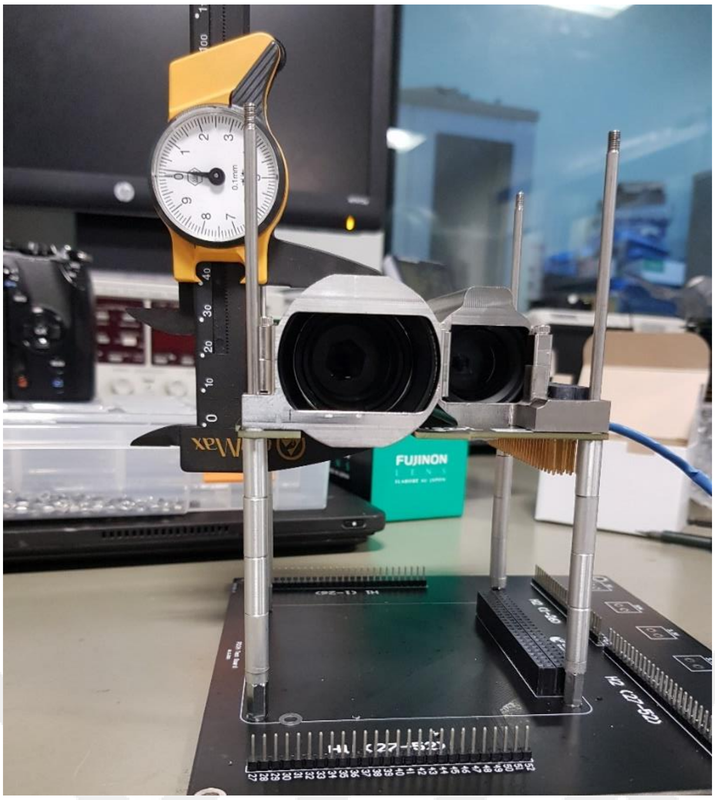

Figure 2.5 : Image from assembly of the second payload.

2.1.3 Mass requirements

The mass of a 3U+ satellite could be up to 6kg [1]. Center of gravity of a CubeSat should be in limits described in Table 2.1, this distances are measured from geometric center of the CubeSat.

Table 2.1: CubeSat acceptable center of gravity locations [1].

	X Axis	Y Axis	Z Axis
1U	+ 2 cm / -2 cm	+ 2 cm / -2 cm	+ 2 cm / -2 cm
1.5U	+ 2 cm / -2 cm	+ 2 cm / -2 cm	+ 2 cm / -2 cm
2U	+ 2 cm / -2 cm	+ 2 cm / -2 cm	+4.5 cm / -4.5 cm
3U	+ 2 cm / -2 cm	+ 2 cm / -2 cm	+7 cm / - 7 cm
6U	+4.5 cm / -4.5 cm	+ 2 cm / -2 cm	+7 cm / - 7 cm
12U	+4.5 cm / -4.5 cm	+4.5 cm / -4.5 cm	+7 cm / - 7 cm

2.1.4 Manetic requirements

ADCS system of the satellite utilizes dual magnetometers, one being redundant and other being deployable. Operation of the ADCS is dependent on the magnetometers on board. These devices are very sensitive to disturbances in magnetic fields, and can become in operatable under large magnetic fields. In order to get accurate data, magnetic fields over these sensors should not be affected by disturbances from satellite. This limited materials on the satellite to diamagnetic or paramagnetic materials, which is another limiting factor over already limited material compatibility of the spacecrafts caused by vacuum of space [17, 18].

2.1.5 Volume requirements

The volume available to use in a 3U+ CubeSat could be seen in Figure 2.6 and Figure 2.7. Because of shape of the CubeSat, and the designs of the subsystems; vertical (z axis) space is very valuable for SharjahSat-1 as ADCS system, StarTracker and iXRD takes considerable amount of space. This leaves very limited amount of volume for rest of the systems, and even less for the secondary payload.

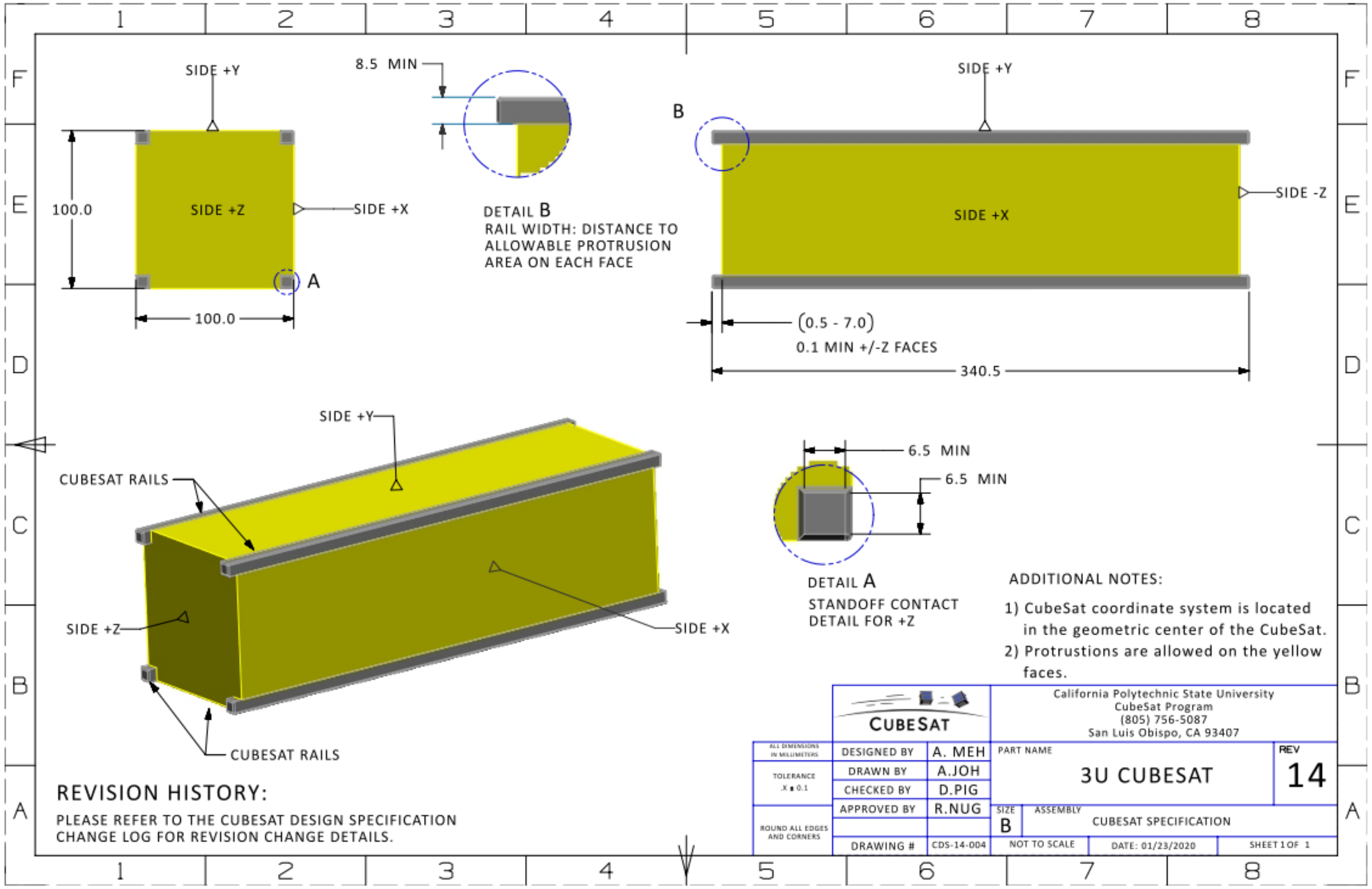

Figure 2.6 : 3U CubeSat volume [1].

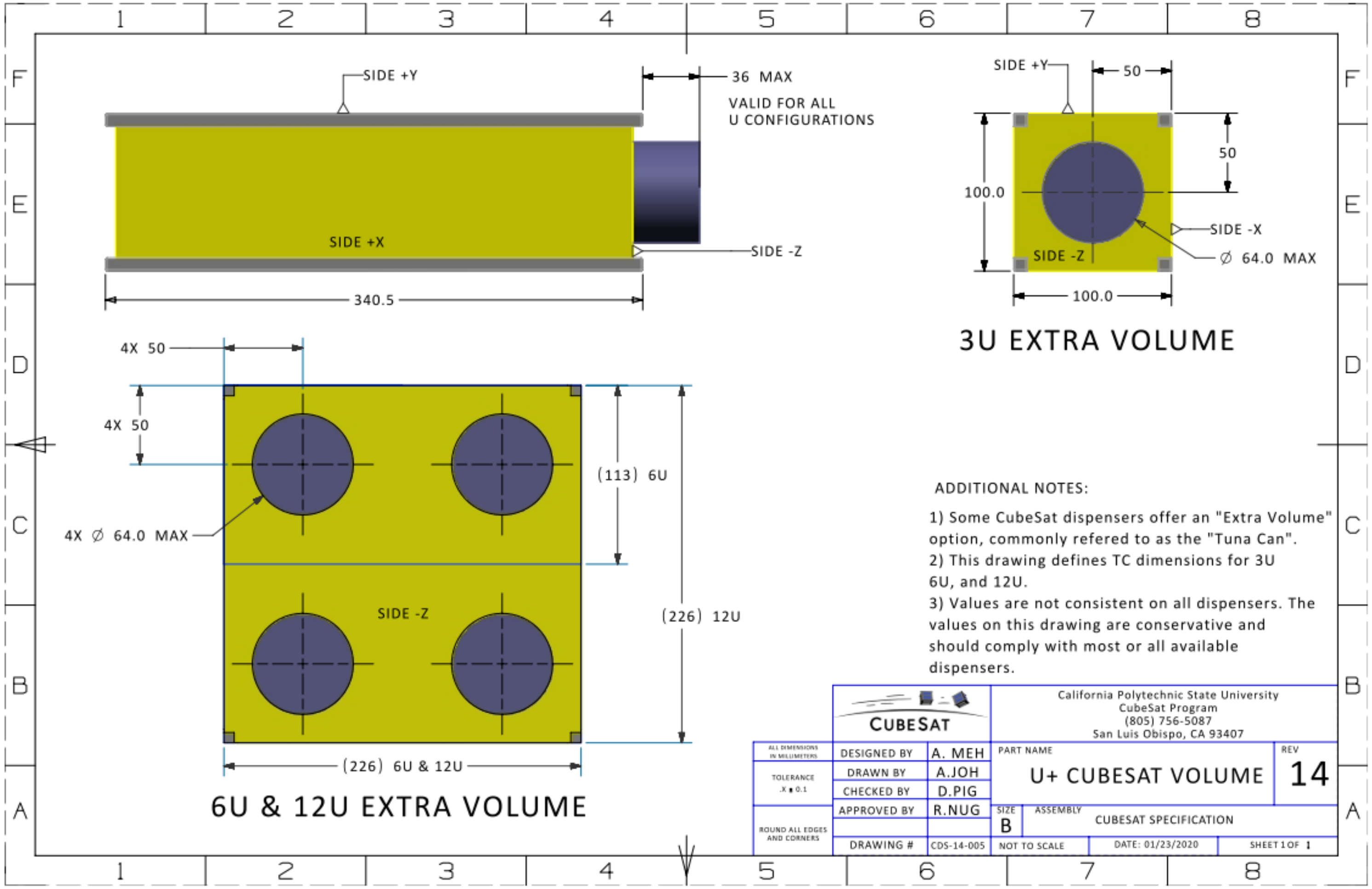

Figure 2.7 : Extra volume (tunacan) [1].

2.1.6 Payload-2 cost requirements

SharjahSat-1 main payload is iXRD. This leaves secondary payload with low budget for cost, and volume. Many of the design considerations are done with strict adherence to budgets. After evaluating available CubeSat imagers, it was decided that designing and manufacturing an imager from CoST parts would be the more affordable option, this was possible because of USTTL’s internal testing capabilities. Designing imager specifically for SharjahSat-1 mission also meant better performance for the available volume.

2.1.7 Parts availability

Parts availability is another important concern. Locally available parts are often preferred as it can reduce complications with importing, shipping, legal work etc. Being able to work with local suppliers were a contributing factor on the selections of the parts.

2.1.8 Processing power limits

Satellite OBC is usually not the most powerful computing unit, because of concerns of reliability, radiation toughness, and other various reasons. Computing power could get to backseat on the deciding process. Some CoTS imaging sensor boards also include processing equipment for image taking and recording. This is a valuable asset, as it eases up software development process and coincidentally reduces development time. Opening up computing resources for other processes increases the reliability, and performance of the real time operation system that runs on the satellite OBC.

2.1.9 Satellite bus requirements

CubeSats usually follow PC104 connectors for their subsystems, and many of the boards use same spacing for their mounting holes. Limited volume and fixed positioning of the PC104 connector creates design limitations on the length of the optics of the imager, as it will be explained in further sections.

2.2 Satellite Imaging Systems

2.2.1 Imager quality

Image quality for a remote sensing space craft imager is a complex relationship between impulse response of spaceborne optical imagery system, its radiometric noise, its spatial sampling period, or frequency, and the on-ground digital post-processing [19]. Sampling period is usually defined by GSD the Nyquist frequency of GSD corresponds to the highest spatial frequency that can be represented by the imaging system [19]. The impulse response of the spaceborne optical imagery system, called point spread function (PSF), includes optical effects –diffraction and aberrations–, spatial and temporal integration of the photosites –detectors effects–, and motion blur effects due to line of sight movements and perturbations during the integration time [20].

2.2.2 Lens mounts

Lens mount is a mechanical and often also electrical interface between an imager and its lens. Besides from aligning electrical contacts, mounts are also responsible for aligning optical axes, and placing lenses at their designed flange distance. Mounts can be threaded type, bayonet type or friction lock type. Threaded types provide less precision on the rotational position. CoTS optics usually implement a standard mount type. While many of the mounts are specific to the company developing them, some of them are open standard. Lens mounts could limit the maximum size of the image circle. Machine vision, surveillance, industrial lenses etc. usually implement D, S, CS, and C mount as their interface.

2.2.3 Ground sampling distance (GSD)

Ground sampling distance is distance between pixel centers measured on the ground an example can be seen in Figure 2.8. It is one of the theoretical limiting factors of the spatial resolution. GSD does not take temporal, or spectral limitations nor limitations of the optical front end of the imager. It is widely used in remote sensing application, but does not represent final image quality.

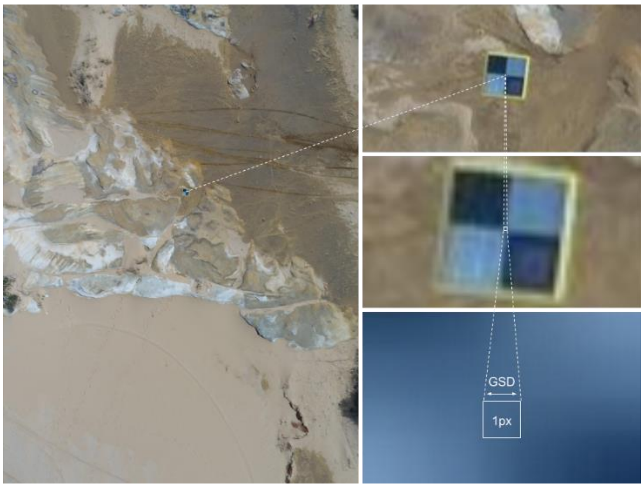

Figure 2.8 : Ground sampling distance example.

Calculations for ground sampling distance **(GSD)** in meters are done by:

$$GSD = (H*px) /f \quad (2.1)$$

GSD = Ground Sampling Distance (m/px)

px = Linear dimension of the pixels (in either width or height)

f = Lens' focal length

2.2.4 Modulation transfer function (MFT)

Modulation transfer function is one of the most important criteria that is describing performance of an optical system. It could be for a whole imaging system or just a part of it. It could be analyzed or it could be tested.[21 - 23]. Results can be represented as line pairs per millimeter (lp/mm). In Figure 2.9 MTF graph for different abbreviations effects, and the whole imaging system could be seen (graph is dimensionless).

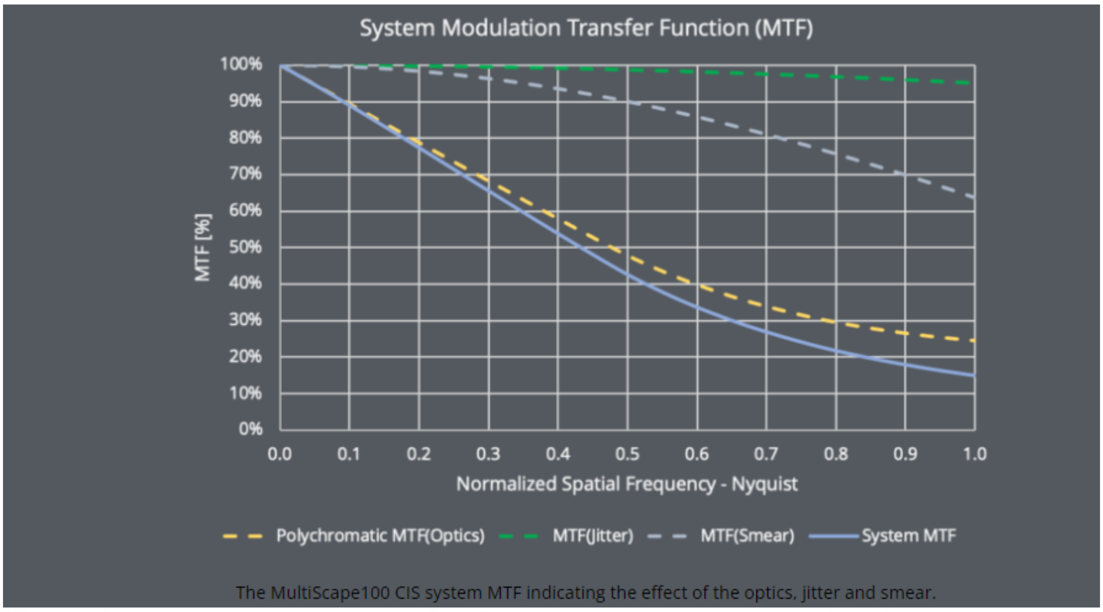

Figure 2.9 : MTF for a imaging system [24].

2.2.5 Angle of view (AoV)

Angle of view is shown in Figure 2.10 and it is angular extend of scene that is seen by the imager.

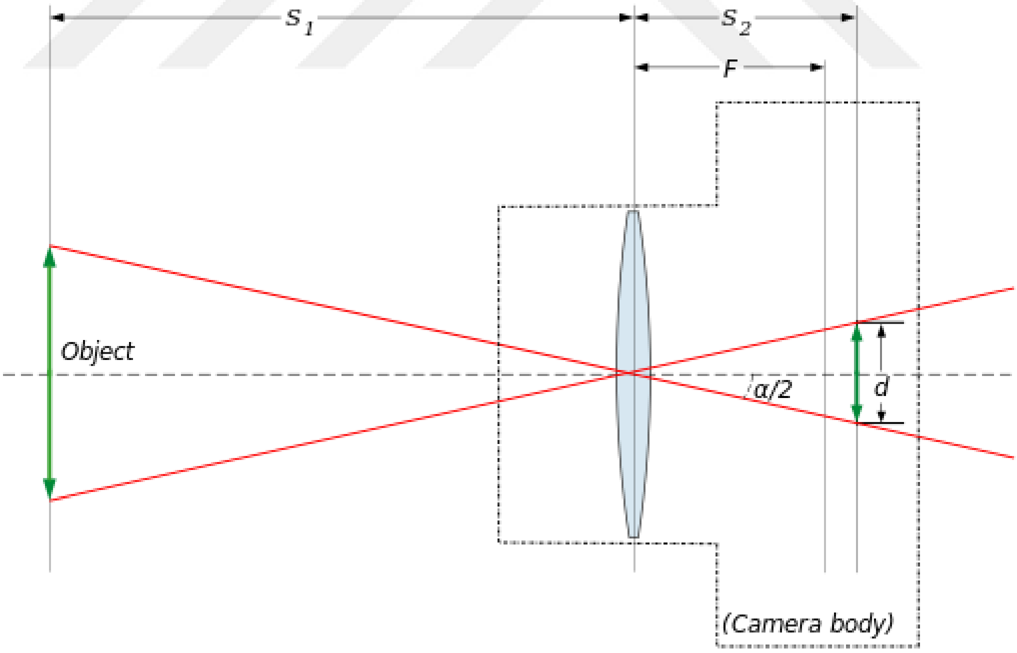

Figure 2.10 : Angle of View.

Calculations for AoV (angle of view) are done by:

$$a = 2 * arctan(\frac{d}{2f}) \tag{2.2}$$

$\boldsymbol{a}$ **=** angle of view in degrees (for width or height)

d = chosen dimension of the sensor (width or height)

f= focal length of the lens

2.2.6 Diffraction limited system

Diffraction is defined as the interference or bending of waves around the corners of an obstacle or through an aperture into the region of geometrical shadow of the obstacle/aperture [25]. In Figure 2.11 diffraction of the laser light could be seen (assuming a coherent, monochromatic wave emitted from point source).

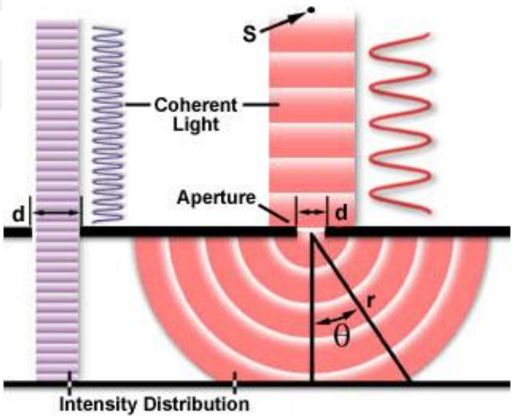

Figure 2.11 : Single-Slit optical diffraction experiment [26]**.**

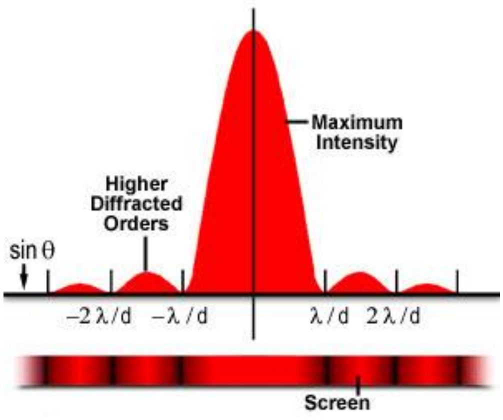

Figure 2.12 : Diffracted light intensity distibution [26]**.**

In Figure 2.12 light intensity distibution of a diffracted light could be seen.

Diffaraction of light is a paramount limiting factor on the performance of the any optical system. Wave like nature of light forces an ultimate limit on the performance of all optical instruments [26]. Disregarding effects from aberrations, angular resulation of a system effected by diffraction could be estimated by Rayleight criterion, it assumes two point sources are regarded as resolved when the principal diffraction maximum (center) of the Airy disk of one image coincides with the first minimum of the Airy disk of the other as it could be seen in Figure 2.13 : Airy disk spacing [25].

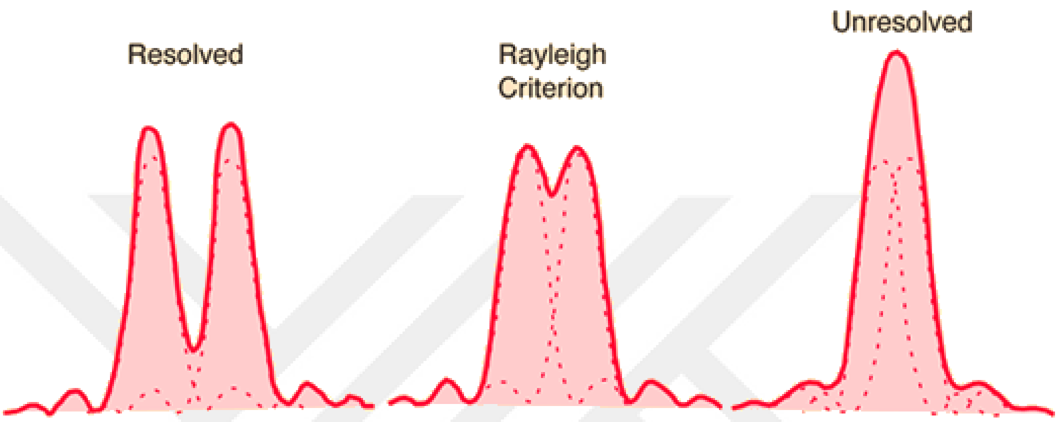

Figure 2.13 : Airy disk spacing [27].

This effect shows itself on imaging sensors as in Figure 2.14.

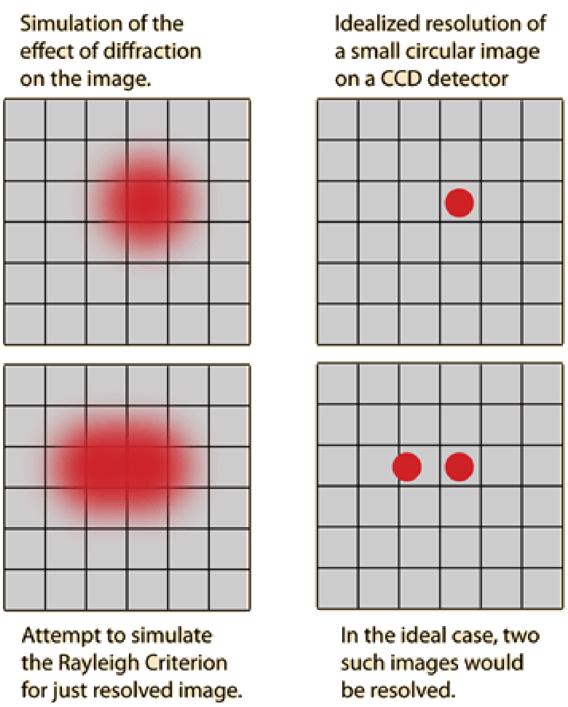

Figure 2.14 : Diffraction on a image sensor [27].

This effect on a imager could be calculated by [28]:

$$1.22*\lambda*(F/\#) \tag{2.3}$$

where:

λ = wavelenght of the light.

F/#= f number of the imager's lens.

2.2.7 Determination of integration time

Integration time is the length of time that is given to imager to gather light to create an image. To keep the same exposure, less integration time will result in a noisier image. Longer the integration time, less gain noise will be on the image. Integration time needs to be a length where imager will have the ability to gather enough light to have an appreciable level of noise, but not show motion artifacts. Estimation for a suitable integration time could be assumed to depend on; satellites orbital velocity, Earth's velocity if satellite is assumed to have a suitable ADCS system where there are no significant disturbances from thumbling and vibration of the satellite. Further calculations will be shown in following chapters.

2.2.8 Data formats

Analog values imposed by light rays on the imaging sensor is converted to digital numbers in one of the steps of the imaging chain. This data could be recorded in a Lossy or Non-Lossy formats.

2.2.8.1 Lossy formats

When imaging data is recorded in a lossy format such as JPEG some spectral information is lost. This is done to keep file sizes low. To achieve this most of the information that is not required to create the exposure is removed from the image. Decision of the desirable exposure is done on the image processing chain on the imager or image processor, which usually means limited user input. When a lossy format is considered, if final exposure is not a desired one (wrong white balance, dark image etc) there might not be much to do to fix the image. Benefit of this format is much lower file sizes, which could be very useful for a satellite imager where link budgets are not very generous.

2.2.8.2 Non lossy formats

Non-lossy format means there is no information lost from the original sensor data. There could be compression on the data. Some image sensors could transfer digital numbers recorded on pixels of the imager. This digital numbers needs to be processed to show a meaningful image. There are many kinds of imaging sensors with various kinds of band filters (usually called color filter arrays CFA). Here in this thesis scope is kept on Bayer filtered imaging sensors, as they are the most common type of imaging sensor available as CoTS. Sensor with a Bayer filter could be seen in Figure 2.15.

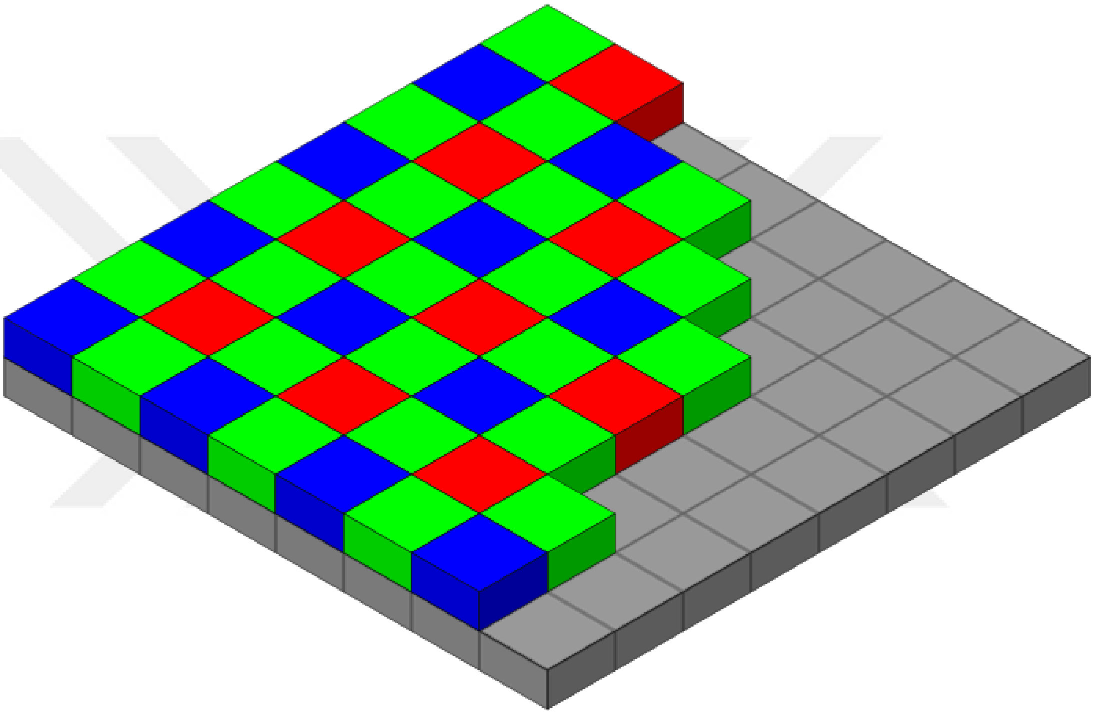

Figure 2.15 : Example bayer type color filter array.

Every sensor has a different spectral response. In order to render an image from raw sensor data, conversations must be done.

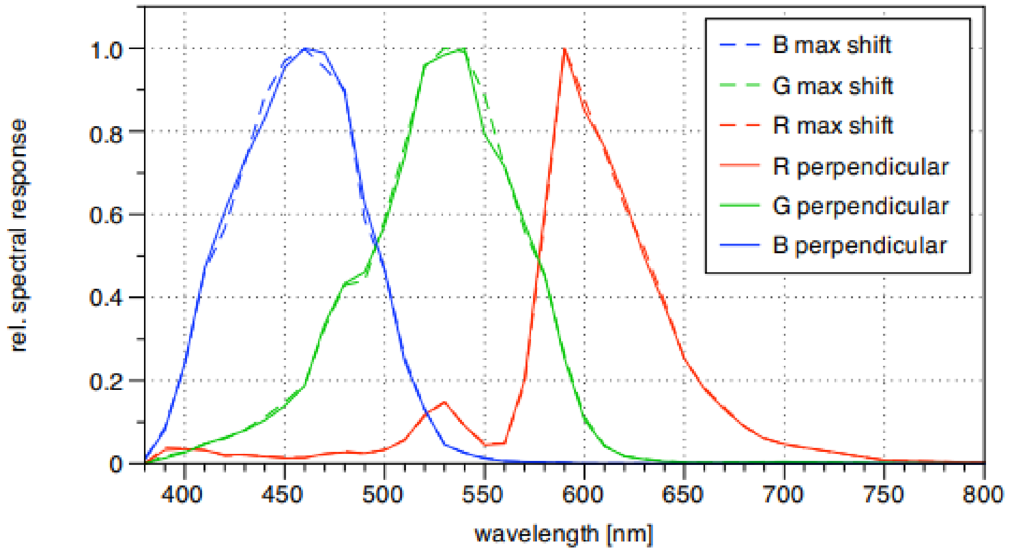

Figure 2.16 : Example spectral response of an imager.

There are many steps to the rendering processes, to give a general idea these steps could be described as follows. First step, could be loading of the linear data from the sensor raw file, and subtracting of black levels. Black level of the sensor at the time of imaging could be found from masked pixels that exists on some imagers. Next step is to white balance image, which is done by multiplying linear data with constants to shift color balance of the image. Up until now image is still in grayscale, because of CFA. Now image needs to be reconstructed to be in color. There are different kinds of algorithms, some opensource some closed to debayer or demosaic an image; which is the process of combining pixels to construct full color information. Detailed compression between different algorithms could be seen in [29]. Every sensor has different spectral response which is usually similar to Figure 2.16. Previous step has provided RGB color information for every pixel. To get accurate color information this RGB values needs to be transformed by a forward transformation matrix.

2.3 Environmental Constraints

Space environment has its own challenges for design of any system, and in the following chapters some design challenges for space use shall be explained.

2.3.1 Vacuum environment

High vacuum of space is a major limiting factor for any device. Space environments high vacuum can range between $1.33*10^{-12}$ Pa to $1.33*10^{-16}$ Pa [30]. This results in outgassing from materials, and makes some materials unsuitable for space usage [31]. Outgassing could effect optical properties of the material or materials around it, also outgassing could create structural faults such as cracks, pores, and disturbance in bondings between materials. High vacuum also increases chemical activity of materials and the effects of the atomic oxygen. In organic materials, behavioral change could be unpredictable. [17, 30, 32, 33]

Mechanical sealing of voids could propose a problem as well, on CoTS systems such as lenses there could be seals between optics, if there are no adequate path for air to escape it could create structural problems, cracks etc. Optical adhesives, could move. Optics themselves could move while getting into high vacuum and reduce optical alignment. Light bends differently in vacuum this could oppose a problem when the device is focused under clean room conditions by changing the focus plane this requires testing.

2.3.2 Vibration

Launch conditions could create misalignment in the optical path, designs should be suitable for launch vibrations. Besides from alignment of optical path launch vibrations creates problems on the general design of the satellite. Another vibration that should be considered is vibrations caused by ADCS momentum wheels. To test if satellite could survive launch conditions, analyses and tests are done [34–36].

2.3.3 Radiation

Radiation is a constant factor to be considered for designs in space, in this section thermal radiation is not considered. Cosmic radiation effects are considered. These are high energy charged particles of galactic cosmic rays, solar particle events, secondary protons and neutrons [37]. These particles could create single event effects, and bit flips on satellite electronics [38]. There are some protections, but for some missions they are not critically important besides from some basic protection. Another problem solar radiation, some electro magnetic radiation from light is at harmful frequencies optical surfaces and their coatings, so special care should be shown on the selection of the optics.

2.3.4 Thermal

Thermal design of a satellite is crucial for a healthy mission. Satellites could heat from incoming radiation from sun, albedo, cosmic rays, and heat generation inside satellite components. Satellites could also get very cold during eclipse parts of the orbits because of large radiative heat loses. Optical systems should be tested thoroughly during environmental testing to test their capabilities. Optical systems could loose critical focus when heated up or cooled down too much. Noise levels on an imaging sensor could increase with heat too. These all should be tested for a design to see suitable performance is achievable for payload on the satellite bus.

3. INSTRUMENT DESIGN

In this thesis design of secondary payload of SharjahSat-1 is discussed. In Section 3 design goals and design limitations; equipment selection, detailed design of hardware and software, manufacturing, instrument & satellite assembly, and calibrations is discussed. In section 4 test setups, analyses, tests are presented.

3.1 Design Goals

Following goals are set for the system:

- System should be suitable for a limited budget.
- System should be able to take images of SAASST and its surroundings.
- System should have a GSD better than 50m/px, with acceptable image quality.
- It should take up as little volume as possible; since, space in Z direction is very scarce in SharjahSat-1.
- System should utilize CoTS components, and should have dual lenses and imaging sensors.

3.2 Design Considerations and Limits

Majority of CubeSats follow similar specifications and design rules with slight differences between them depending on the launch provider, and some design limitations that come from systems inherent nature. This creates limits on the design of the system in question. Such as; position of the center of gravity, external dimensions, material selections etc. All of this usually manifest themselves on the design as volume and sizing problems, since a CubeSat is already a largely volume constrained system. Result of this is that size of any system is very crucial. SharjahSat-1 mission proposed its own unique challenges on the design of the payload. Design of the secondary payload carried out simultaneously with the main payload and the satellite subsystems. This created its own separate challenges.

Because of the main payload’s strict requirements on the attitude control, SharjahSat-1 required to have 3 wheel ADCS as seen in Figure 3.1. This system takes up nearly 90mm in Z height of the satellite. Because of the precision needed to observe sources SharjahSaT-1 is also equipped with a Star Tracker, which also takes up valuable space in Z axis nearing 35mm. Mounting if the Star Tracker inside satellite is shown in Figure 3.2.

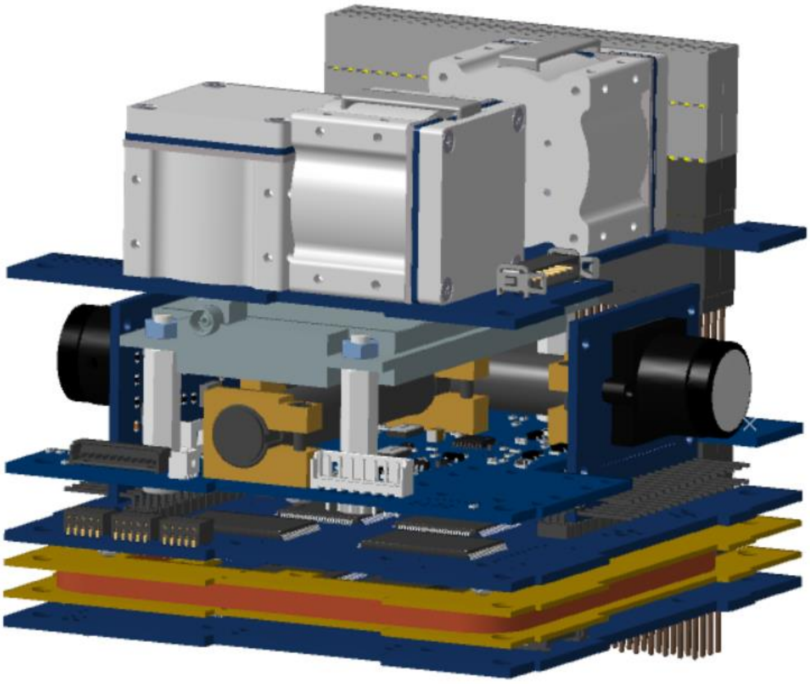

Figure 3.1 : SharjahSat-1 3 wheel ADCS system.

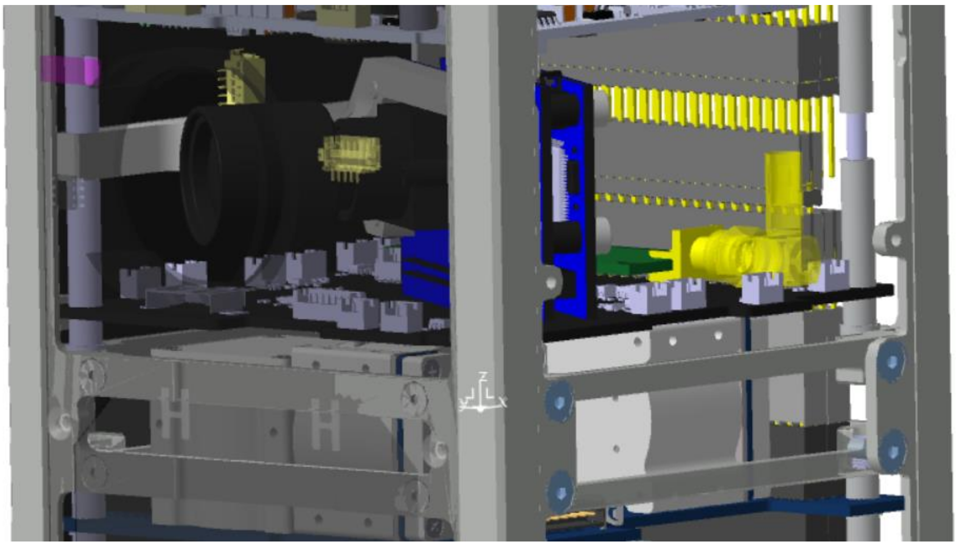

Figure 3.2 : Star Tracker assembled in satellite.

From the start of the design phase it was predetermined that iXRD would require a relatively long tungsten collimator to limit incoming X-rays. This collimator's length were calculated to be around 75mm. Best solution regarding size limitations were to place this collimator protruding outside of the satellite. In Figure 3.3 tungsten collimator and its mass could be seen. When remaining parts of the iXRD is assembled, it occupies length around 95mm.

Figure 3.3 : SharjahSat-1 iXRD collimator.

A 3U CubeSat could be 340.5mm [1]. Additional 36mm of Z height could be gained by utilizing space that is defined as "Tuna Can" in Figure 2.7 by making SharjahSat-1 a 3U+ sized satellite. To achieve that iXRD collimator needs to be protruding outside into this extra volume, as shown in Figure 3.4. Collimator is 454gr, and iXRD system with its many shields made from tungsten and aluminum is about 1024gr in total.

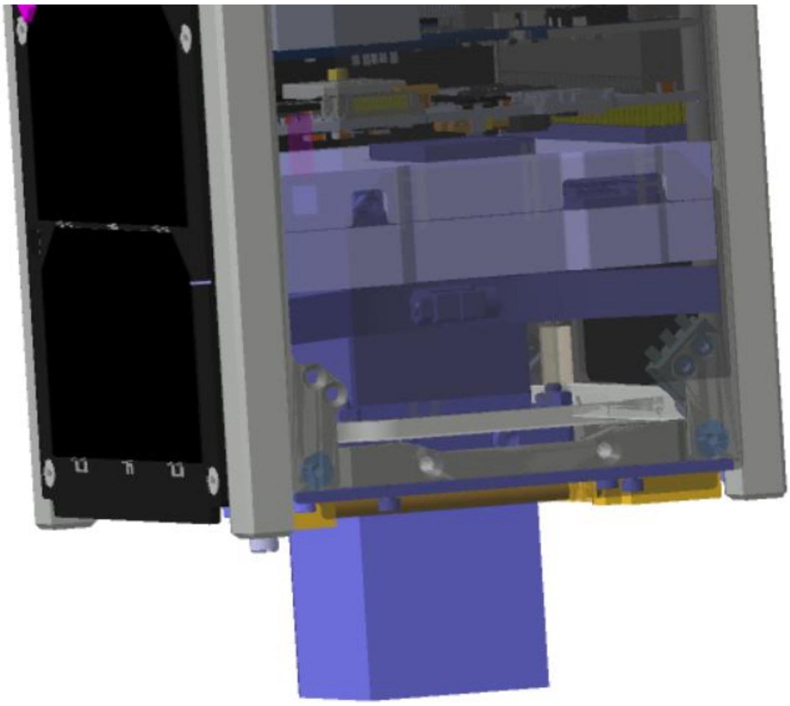

Figure 3.4 : Protrusion of iXRD collimator.

Since, iXRD is at one of the extremities of the satellite its effect on center of gravity is paramount. Because of this outcome, and mass of the other subsystems, secondary payload also assumed duty of additional mass (given it did not have any negative outcome on the lifetime of the satellite orbit.)

Constraints resulted by supporting the main payload left secondary payload with a Z height of around 40mm. Figure 3.5 shows imaging system in this space.

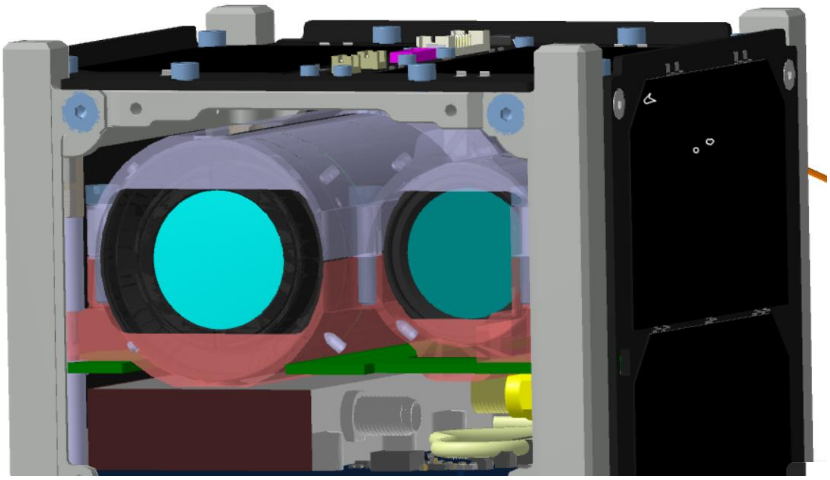

Figure 3.5 : Imaging system in space left.

Wiring harnesses were an important factor in the design of the imaging system, since they required cabling for external sensors, umblical connectors, solar arrays, antennas etc. Vibration resistance and interaction between components were also considered during the design phase. Available limits of manufacturing, such as tolerances, clearances, and allowances) and accuracy limits of assembly were also limiting factors during design. Global supply chain issues, and known available time frames were also important limiting factors. Selection of the imaging sensors were limited by availability, capabilities of OBC, available time for development.

3.2.1 Sensor selection

Sensors in Table 3.1 were considered to be used in imaging system. These were readily available imagers during the development time. There were no available space heritage information for any of these sensors. Any of them required testing.

Table 3.1: Sensors considered.

Board Manufacturer	Sensor	Size	Resolution	Global Shutter	Color	Interface
Arducam	OV5642	1/4"	5mp	No	yes	SPI&I2C
Arducam	OV2640	1/4"	2mp	No	yes	SPI&I2C
EconSystems	IMX290	1/2.8"	2mp	No	Yes	MIPI
NA	IMX252	1/1.8"	3.2mp	Yes	Yes	Camera Serial Interface
Sony	XCL-SG510C	2/3"	5.1mp	Yes	yes	CameraLink

Satellite OBC only supported SPI. Both sensor boards from Arducam were selected, as they were the only ones that could be natively supported by OBC. Specifications of the sensors used on the boards are shown in Table 3.2.

Table 3.2: Sensor Specifications.

Manufacturer	Sensor	Size	Resolution	Pixel Size	Sensitivity	Interface
Omnivision	OV5642	1/4"	2592x1944	1.4 µm x 1.4 µm	600 mV/Lux-sec	SPI&I2C
Omnivision	OV2640	1/4"	1600x 200	2.2 µm x 2.2 µm	600 mV/Lux-sec	SPI&I2C

Arducam boards handle data communications with OV5642, and OV2640 sensors. Imaging sensors are directly connected satellite OBC via I2C. I2C bus is used for configuring imagers. Arducam boards house a FPGA and ram chip, which handles data connection with imaging sensors, and then stored it on on board ram. OBC could access this image data via SPI to store it on flash. SPI connections are natively supported by OBC. The OV5642 is an image sensor with low-voltage, 1/4-inch 5-megapixel (2592x1944px) CMOS imager. Sensor offers selectable image quality, formatting and output data transfer. All required image processing functions are programmable through the SCCB interface, MIPI interface or embedded microcontroller. It has an operating range of -30°C to +70°C. The sensor is fed with 5V/70mA power supply during normal operating mode, but it can reach 5V/390mA during peak consumption. Sensor accommodates a low power mode which consumes 5V/20mA power by shutting down the sensor and memory circuits.

OV2640 image sensor is the secondary sensor. It has a resolution of 2 megapixels(1600x1200px). It has an identical operating range to OV5642.

During its operations the sensor consumes 125mW and 140mW on normal mode and compressed mode, respectively. Selected imaging boards could be seen in Figure 3.6.

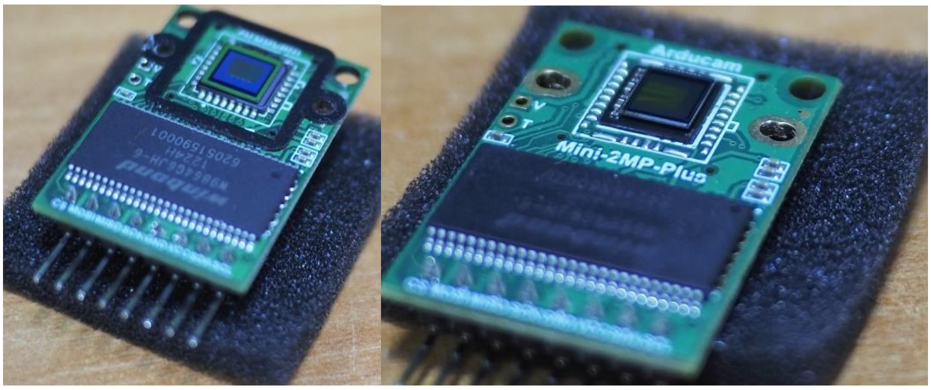

Figure 3.6 : Imaging boards (5mp on left, 2mp on right).

3.2.2 Lens selection

Many CoTS lenses were considered. Selection criterion's were;

3.2.2.1 Focal length of the lens

Focal length is directly affects GSD, and FoV. Suitable focal length is a balance between GSD and FoV while also staying within limitations of the ADCS system. Longer focal lengths could also mean larger physical size, which is a variable that is very limited for this design.

3.2.2.2 F number

F number is important to stay within diffraction limit. Lower F numbers means more light on the imaging sensor. Lenses are usually softer at their largest aperture opening. Value of this variable is a balance between, sensor SNR capability, ADCS capability and image quality of the lens. Lenses with lower F numbers are usually larger too.

3.2.2.3 Mechanical design

To not complicate testing and design lenses were selected to not have any electromechanics. Some lenses are designed to be used in manufacturing environments and machine vision; they are usually more vibration resistant than other lenses. They are more likely to include more fixing points too. Design of the lenses should also be acceptable for high vacuum of space, which means no sealing, and suitable materials, adhesive and grease selection. Their thermal performance should also be acceptable for orbital environment. Some of these simply requires testing, because of lack of detailed information from manufacturers. While selection of the lenses in progress, information from manufacturers and educated guesses were utilized.

It is important that lenses were also able to project large enough image on the selected sensors. Quality of the optics are important to have sharp images.

Price and availability were also played an important role on the selection.

After examining more than 50 different lenses; a selection were made.

First iteration of this system used Fujinon HF50HA-1S and Fujinon HF25HA-1S, both could be seen in Figure 3.7.

Figure 3.7 : Fujinon HF50HA-1S and HF25HA-1S.

Their calculated GSD, and swath values are in Table 3.1, and specifications of the lenses themselves could be seen in Table 3.2. They were selected, because they are small sized lenses with vibration resistant designs.

Table 3.1 : GSD, and swath values for 500km orbit.

Lens	Focal Length	GSD (meter/pixel) horizontal	GSD (meter/pixel) vertical	GSD average	Swath width horizontal	Swath width vertical
Fujinon HF50HA-1S	50	14.17	14.09	14.13	36736. m	27384. m
Fujinon HF25HA-1S	25	44.00	43.57	43.78	71800. m	53941. m

Table 3.2 : Technical specifications of HF50HA-1S, and HF25HA-1S [39].

Manufacturer	Lens	Focal Length	Aperture (F-stop)	Lens Length	Diameter ⌀	Mount	Mechanical Resistance	Mass
Fujinon	HF50HA-1S	50mm	2.3	29.5mm	29.5mm	C-mount	Vibration resistant	45g
Fujinon	HF25HA-1S	25mm	1.4	29.5mm	29.5	C-mount	Vibration resistant	40

They were also locally available, and manufacturer FUJIFILM Dış Ticaret A.Ş. also provided sample lenses to examine their suitability. A structural design for this lenses were ongoing while, suitability and performance tests were also conducted. They passed all environmental tests, but they were deemed to be not sharp enough. Since during this time, satellite design were more mature, and structural details of iXRD were more precise, more volume were allowed for imaging system. With this new space, Fujinon CF50ZA -1S, and Fujinon HF25HX-1S by Fujifilm in Figure 3.8, selected to be used on board. Technical specifications of the duo could be seen in more detail in preceding sections.

Figure 3.8 : CF50ZA-1S (on left) and HF25HX-1S (on right).

The sensor is coupled with CF50ZA-1S lens. It has no electronic components, motors, drivers, servos etc. Coupled with the sensor and CF50ZA-1S's high modulation transfer function gives a theoretical GSD of 14m/px. This sensor is coupled with HF25HX-1S lens. This combination gives a calculated GSD of 44 m/px. MTF for these lenses were provided by manufacturer under NDA, it is not possible to include it in this thesis, but performance deemed to be suitable. Ground testing that will be discussed in further sections confirmed this. Technical specifications of CF50ZA -1S, and HF25HX-1S are shown in Table 3.3.

Table 3.3: Technical specifications for CF50ZA-1S and HF25HX-1S [40, 41].

Manufacturer	Lens	Focal Length	Aperture (F-stop)	Lens Length	Diameter ⌀	Mount	Mechanical Resistance	Mass
Fujinon	CF50ZA -1S	50mm	2.4	68mm	39mm	C-mount	Vibration resistant	155g
Fujinon	HF25HX-1S	25mm	1.6	29.5mm	46.5	C-mount	Vibration resistant	72

According to information given in section 2.2.7 rough calculation for worst case scenario for integration time is calculated as follows: Assuming there are no significant tumbling movement from the satellite bus and with no motion compensation for a satellite that is taking an image of a point on the equator at 500km orbit. Assuming satellite velocity to be 7.6 km/s, and Earth velocity to be 0.5 km/s. Assuming satellite and Earth motion to be exact opposite of each other combine velocity of the ground point could be calculated as

$$V_{comb}=7.6+0.5=8.1 \text{ km/s} \tag{3.1}$$

If we use integration time of 1ms then min GSD resolution would be

$$GSD_{min} = V_{comb}*t_{int} \tag{3.2}$$

$$8.1 \text{ m/px} = 8.1 \text{ (km/s)}*1/1000 \text{ (s)} *1000 \tag{3.3}$$

Which is much lower GSD than what imagers are capable of. During testing it was observed that integration time of 1ms in daylight does not create any problems with SNR.

3.2.3 First design

This design were a stepping stone for the final system, work done on this design will be summarized here including tests done. During this phase many of the details of main payload, and some details of subsystems were not clear. This meant more limitations on physical size. Lenses and sensors were first tested in TVAC to see their combability. They are tested in high vacuum with a temperature range of -30C to +70C. In Figure 3.9 testing chamber, and setup could be seen. System is placed to be focused on the target. It was not possible to test the system at infinity focus (which is what is required for a spaceborne mission) because of lack of infinity collimator.

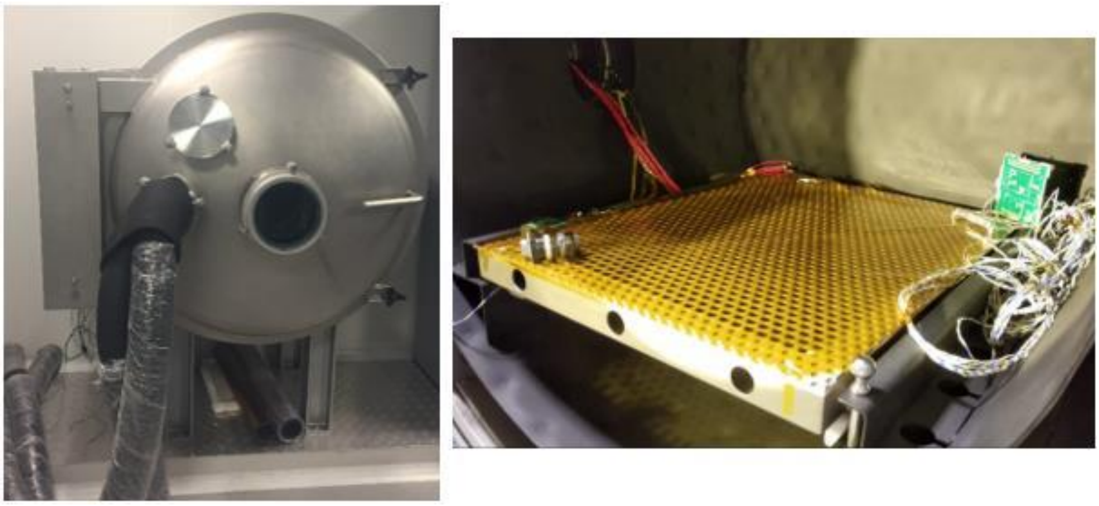

Figure 3.9 : Testing chamber, and testing setup for TVAC.

This test is only done on 50mm, and 5mp sensor. HF25HA-1S lens were not able to focus to target at available distance. During testing imager is mounted to a part of an old satellite structure, and various thermocouples were placed on structure, lens, and sensor as shown in Figure 3.10,Figure 3.11, andFigure 3.12. Target is a space grade PCB. Imager were focused on PCB as best as we could, since setup were not stable (ground testing PCB were not ready). Aim was to see if there were any changes in focus with vacuum, and temperature. Sharpness was not a priority.

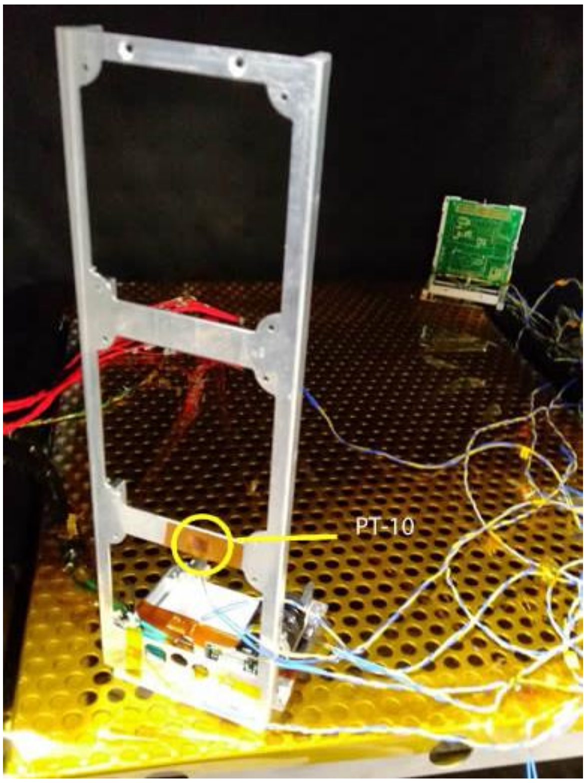

Figure 3.10 : Thermacouple placed on part of old satellite structure.

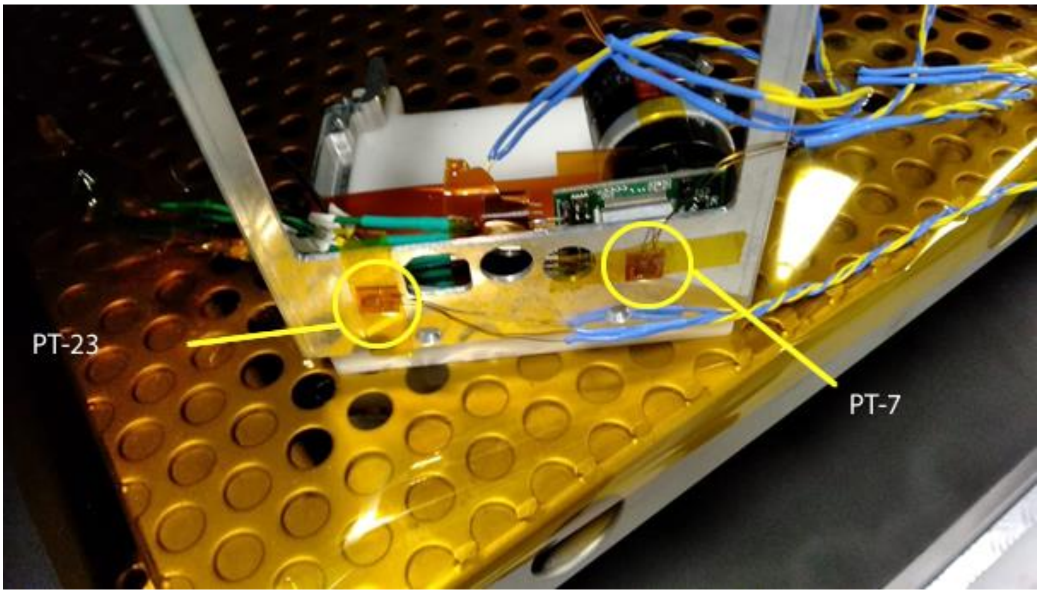

Figure 3.11 : Thermocouples placed near 5mp imager on satellite structure.

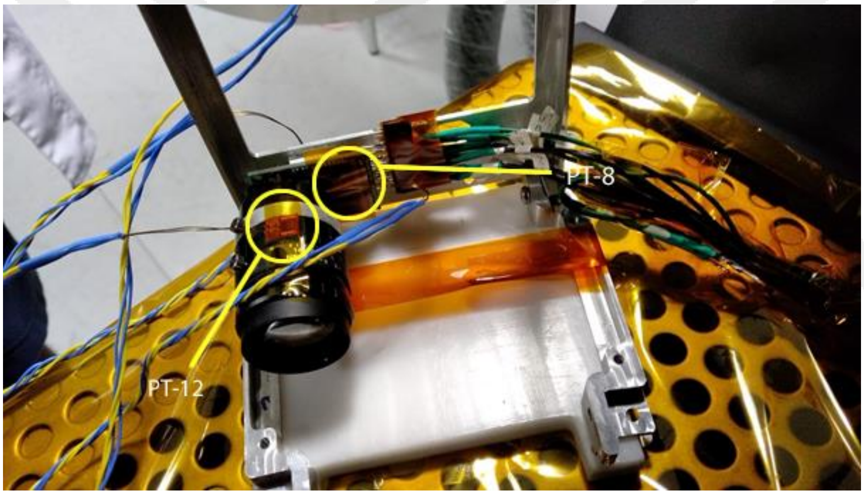

Figure 3.12 : Thermocouples placed on imaging board and lens.

This test were done as a proof of concept at early stage of SharjahSat-1 satellite design, where many of the details were not decided.

After end of the TVAC imager mass were measured, and no mass change were detected. Images taken during test were then examined. From Figure 3.13 it was decided vacuum did not effect image focus.

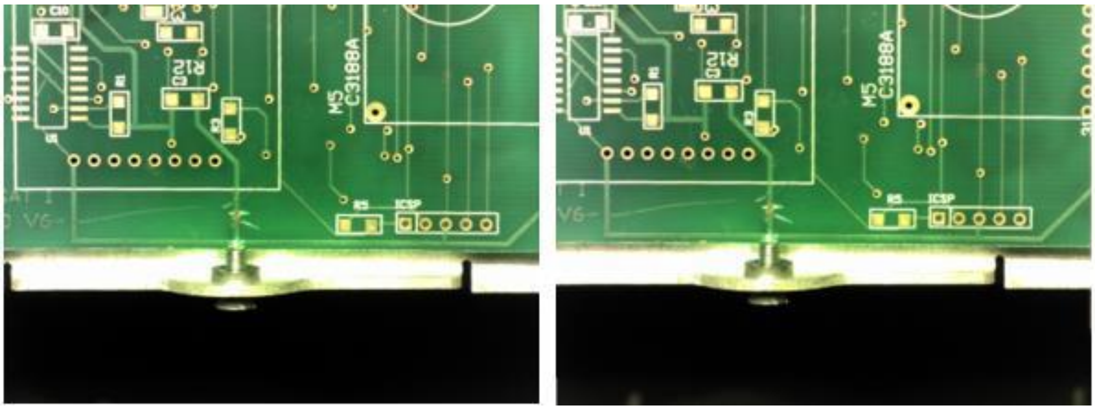

Figure 3.13 : Image captured prior to the test (left) Image taken in vacuum environment, at +20°C (right).

On images taken at 80C (imager temperature) clear sings of noise could be seen in Figure 3.14.

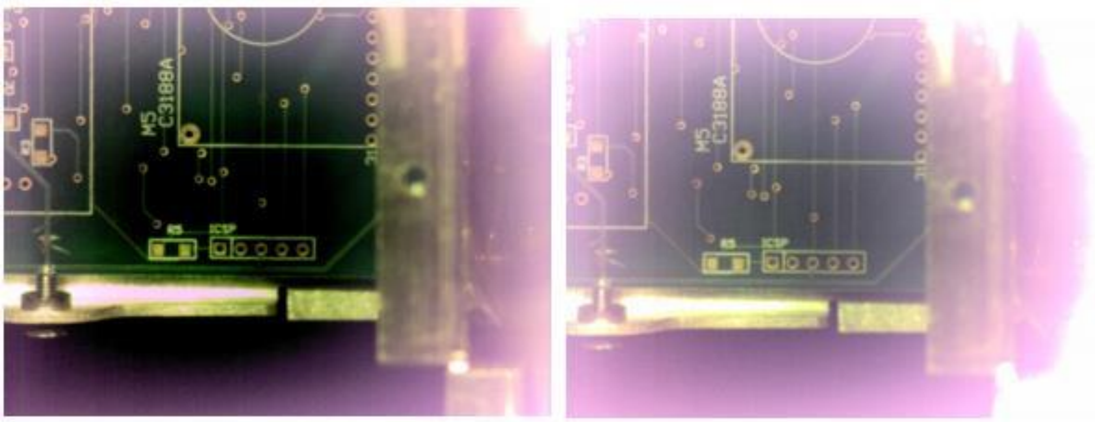

Figure 3.14 : Consecutive images at 80C.

At image taken at -50C change in imager focus could be seen Figure 3.15. Focus is shifted. Images taken later in room temperature, in or out of high vacuum showed no signs of reduction in focus.

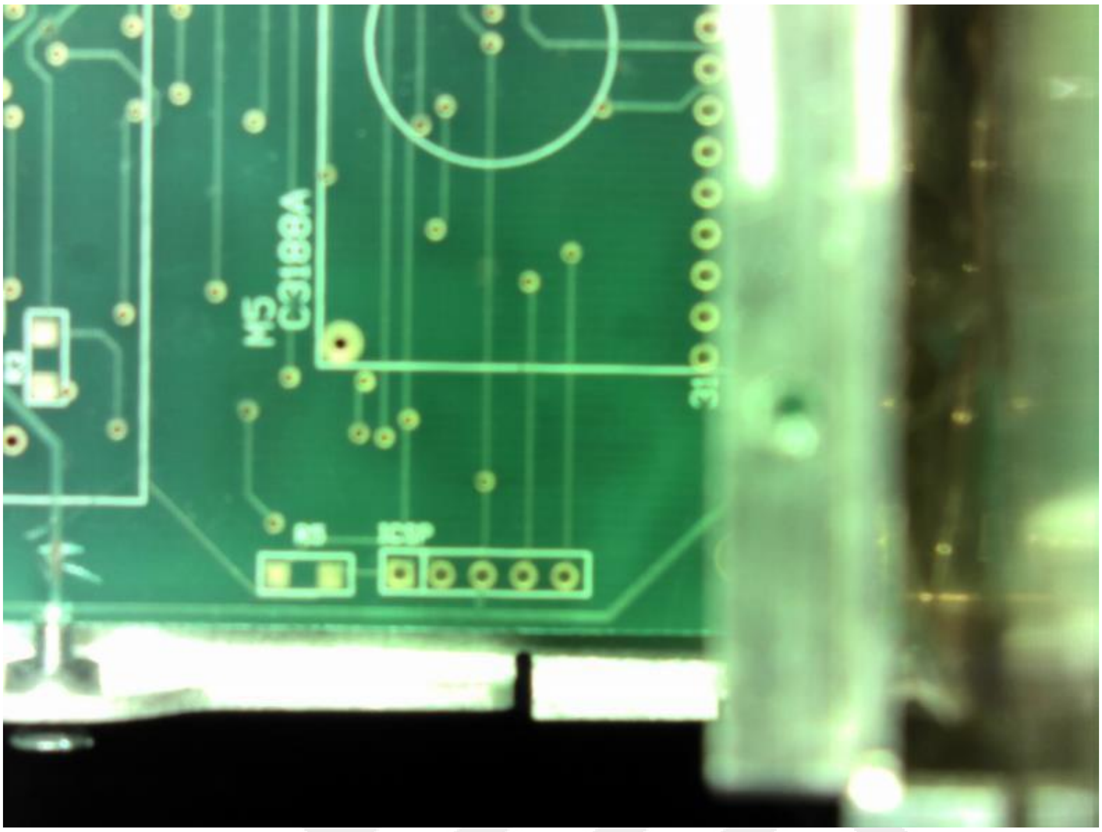

Figure 3.15 : One of the images taken at -50°C.

From results of TVAC test, it was assumed that lens, and sensor is suitable for continuing with design. While design were being done, further test for image quality were planned. Details of the design will be omitted, because it was never manufactured and tested for structural loads.

Some test images were taken from 41m with 25mm & 2mp sensor, to check for the image quality. From images shown in the Figure 3.16 and Figure 3.17 it was decided they were not enough. It was decided that lenses will be changed.

Figure 3.16 : Test targets at 41m.

Figure 3.17 : Close up of test targets.

3.2.4 Second design

Electro Optical Camera Subsystem is a self-contained unit, image parameters can be set from OBC, and the camera handles the rest of the picture taking, then presents the image data to OBC. This eases up satellite computer and, adds protection against failures by keeping image processing away from the main computer. Introduction of more space in to secondary payload meant, more liberty with volume. To improve image quality Fujinon CF50ZA -1S, and Fujinon HF25HX-1S is used in this iteration. This design is deemed to be acceptable after manufacturing and testing. 2 models have ever been manufactured as of date, one for engineering model of SharjahSat-1, and one for flight model. As of writing of this thesis system is in orbit, and system operation is under testing, but since satellite is still in LEOP there are no images of Earth taken.

CF50ZA -1S is connected to 5mp imaging sensor, and HF25HX-1S is connected to 2mp imaging sensor. Imager is designed to be:

- Compact as possible
- Resilient to vibrations as possible
- Stiff as possible
- Heavy as possible
- Flexible for adjustments
- Bus compatible

To make system as compact as possible while also keeping bus combability with pc104 boards, lens and sensor combos are placed next to each other as shown in Figure 3.18.

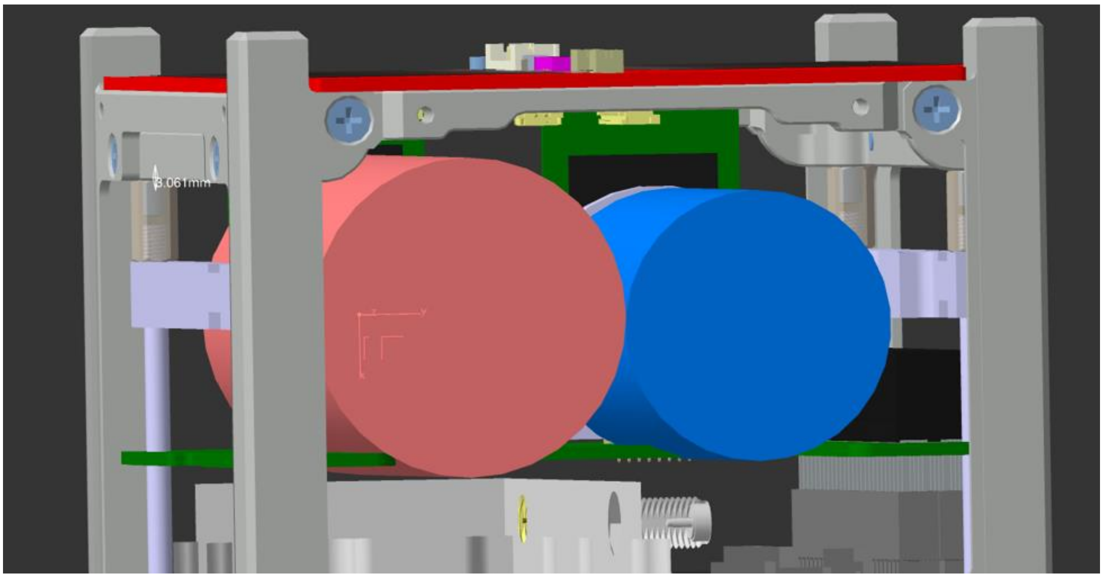

Figure 3.18 : Preliminary lens placement.

Sensors are rotated (as shown in Figure 3.19) 90 degrees in final design, as it used less Z height, and bus combability were still achieved.

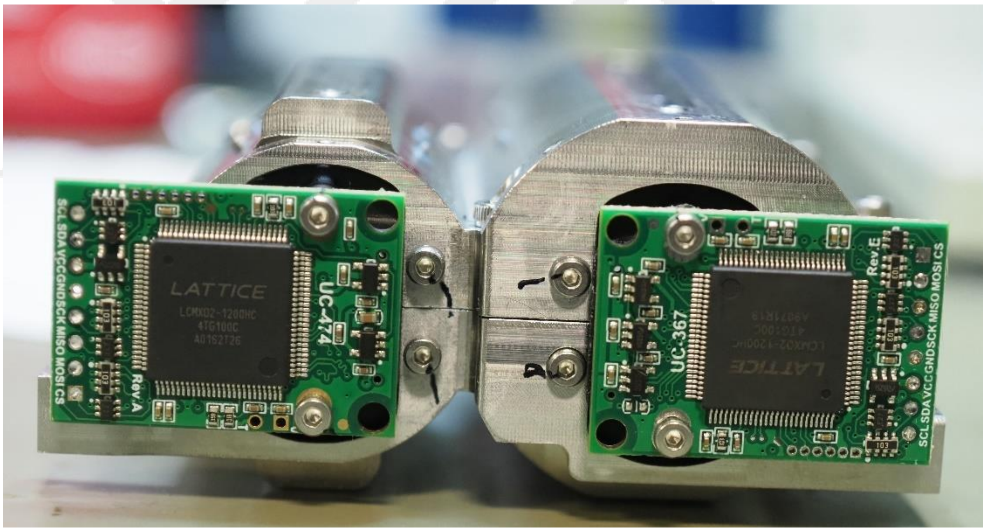

Figure 3.19 : Rotated sensors.

To make system vibration resilient, many precaution were taken. Selected lenses are vibration resistant by design, they all have 3 set screws per adjustment (aperture and focus). All of these set screws are utilized as shown in Figure 3.20.

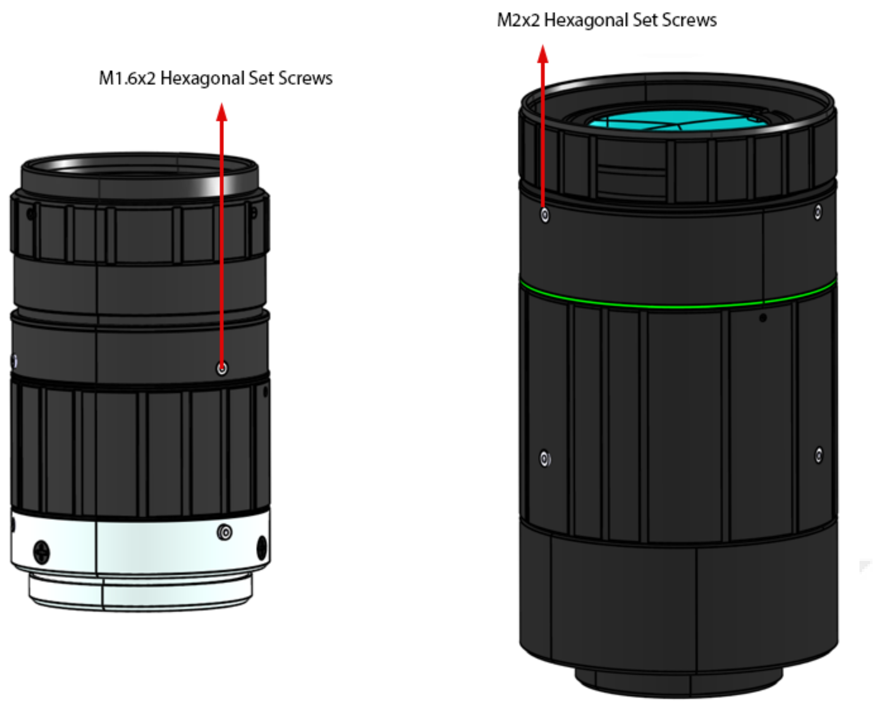

Figure 3.20 : Some of the set screws for lenses.

Each lens is connected to via Integrated Lens Mount Holder (ILMH) as shown in Figure 3.21, which has a C-mount (ASME 1-32UNF) that connects lenses to the ILMH. ILMH is responsible for connecting lenses to rest of the assembly, and sensor boards.

Figure 3.21 : Image of integrated lens mount holder.

ILMH connects each lens to low and high side structural pieces as show in Figure 3.22 and Figure 3.23 respectively.

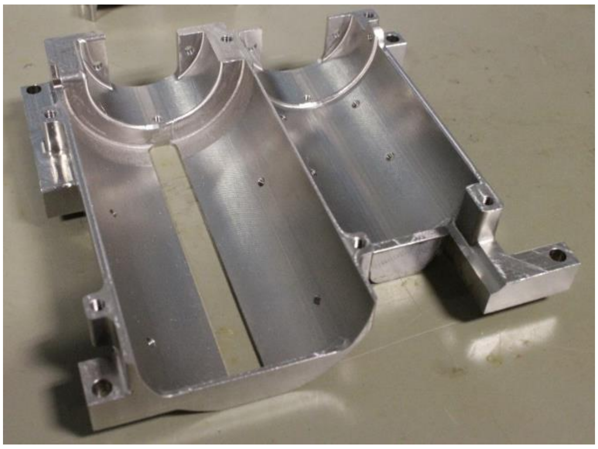

Figure 3.22 : Low side part.

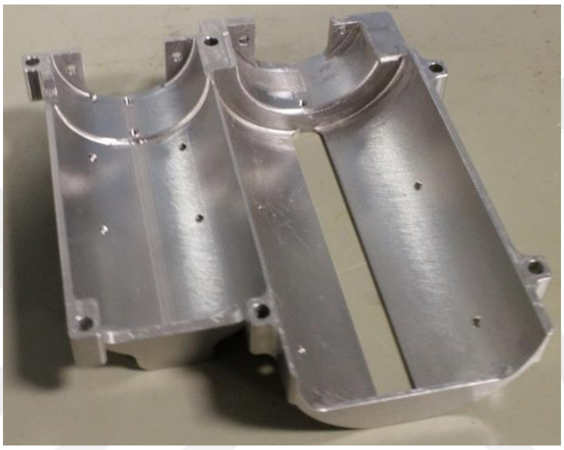

Figure 3.23 : High side part.

These parts have many set screw positions to hold lenes secure. They also provide balance mass for the SharjahSat-1.

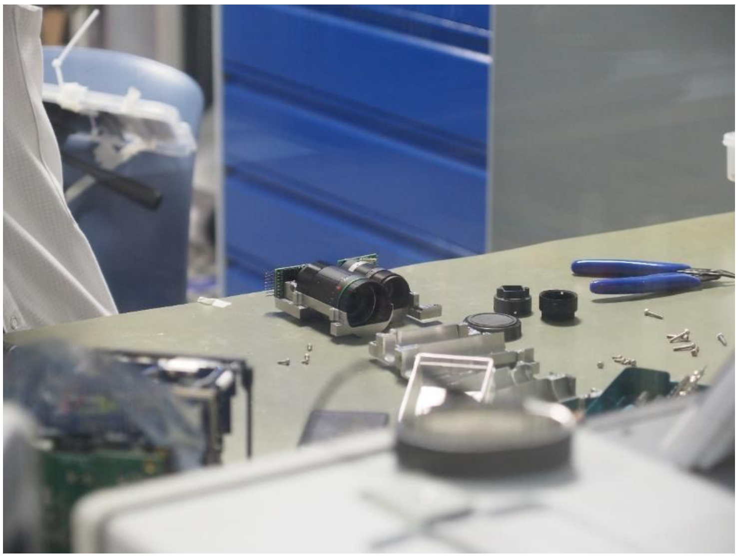

Figure 3.24 : Image from imager assembly.

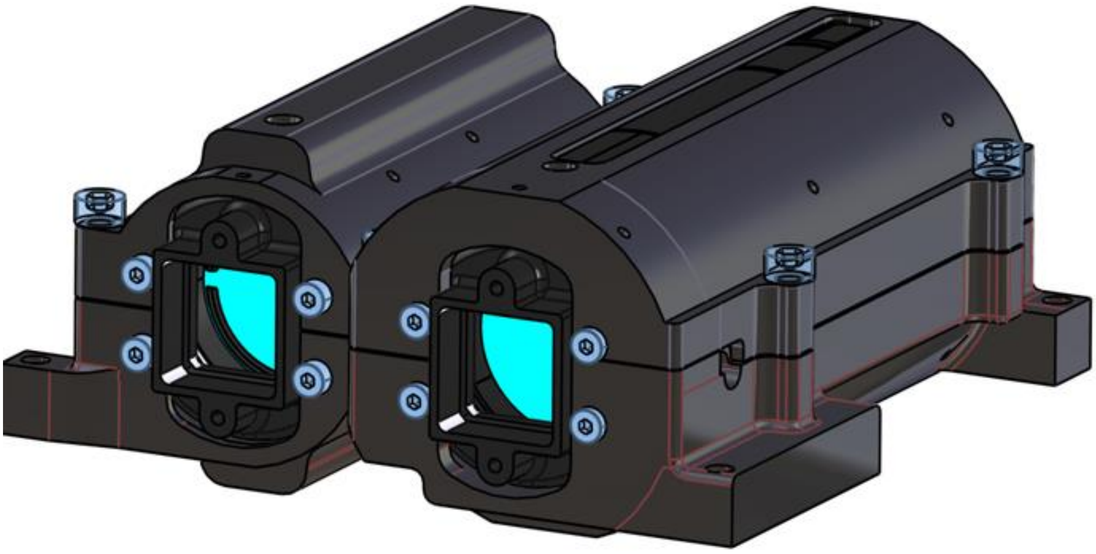

Figure 3.25 : Lenses + ILMH + low side + high side.

System is designed to protect optical alignment as best as possible. Partly assembled system could be seen in Figure 3.24 and Figure 3.25.

Satellite assembly is a complex and lengthy processes, and it is effected by many different subsystems and their interactions between them. Cabling is always a problem, and keeping tight tolerances on assembled parts adds additional steps, and extra time for assembly process. There are always unknowns with timeframes, testing, and possibility of unforeseen problems. Structural holders are designed to have excess space between optical front ends of lenses and holders to allow for lens tolerances as shown in Figure 3.26. Lenses have a general marking that shows infinity focusing setting, but these are usually a suggestion.

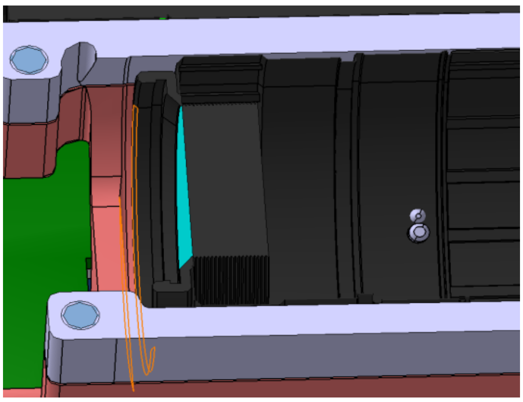

Figure 3.26 : Excess distance between 25mm lens optical front end and structural part.

Lenses have large tolerances for infinity focusing and they can focus past beyond infinity, this is done to compensate for temperature variances, and mounting tolerances. Lens is physically shortest when it is focused beyond infinity. This means when lens is focused to infinity it might be longer than before, and exact length could be found with testing, and could vary from lens to lens. For this design specificity it was possible to ground metal parts of the lenses to fit them. CF50ZA -1S did not need any modifications for flight and engineering model; whereas HF25HX-1S needed a millimeter of material to be shaved off which could be seen in Figure 3.27.

Figure 3.27 : 25mm lens with grounded surface.

Lens assembly is heavy, so it is designed to be only supported by satellite structure, there are no mechanical connections to PC104 board. Sensor boards needs to be connected to lenses, and they also needed to be connected to satellite bus, which is routed through PC104 connectors. To achieve this, a PCB designed. Purpose of this board is the bridge connections from cameras to the PC104 bus. To make heavy components decoupled from this PCB for structural integrity, the sensors are connected to this board via cables, which is something that adds thickness and additional complexity to satellite assembly, but it is considered to be necessary. In

Figure 3.28 assembled system with its white cables could be seen in part of the satellite structure.

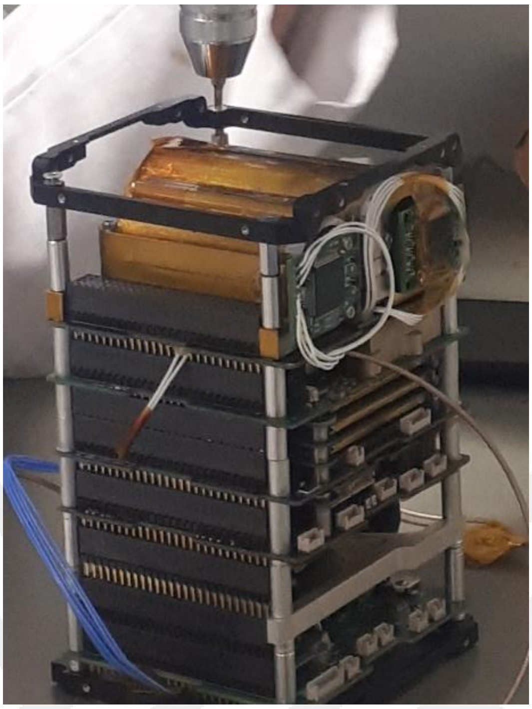

Figure 3.28 : Imaging system integrated to part of the satellite structure.

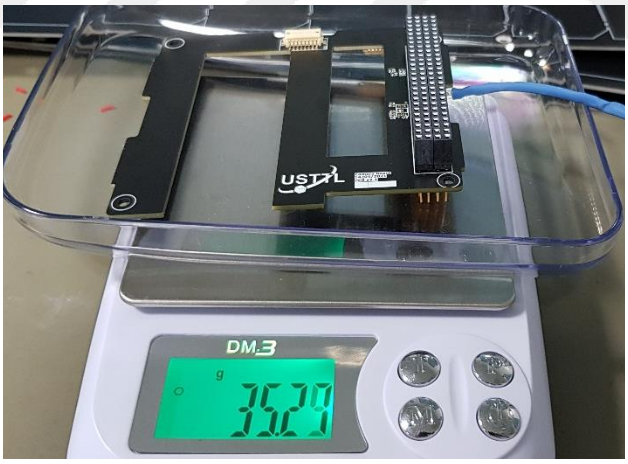

Figure 3.29 : Camera bus connection PCB.

This PCB could be seen in Figure 3.29 and Figure 3.27. PCB has cutouts for protruding sides of the lens assembly, lens assembly uses any space available to either fit large diameter of CF50ZA -1S or to add additional mass, for the mission. In Figure 3.30 Camera system and its neighbors could be seen. System is as close as possible to other system, given that assembly tolerances and cabling (not shown) allows.

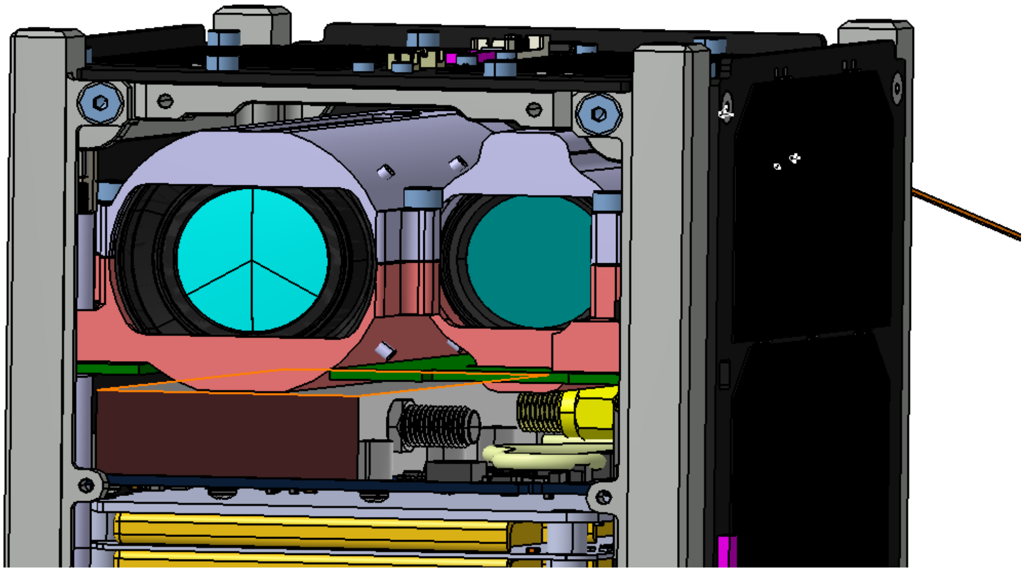

Figure 3.30 : Imaging system, and its neighboors (cables not shown).

Manufacturing, tolerances, and testing will be discussed in further sections. With this design of the system it is measured to be 663.9g, which is a considerable help for minimising distance between Center of Mass and geometric center of the satellite. In Figure 3.30 mass measurement excluding bus PCB could be seen.

Figure 3.31 : Lenses + sensor boards + structural parts of imaging system.

Metal parts of the assembly is covered with polyamide tape to insulate it against possible electrical shorts, and possible damage inflicted to bus PCB with launch vibrations.

Small baffle for protection against stary light rays is also added to the looking side solar panel. This is a simple Delrin piece in black, with rough inner surfaces. Size is limited by POD. Flight model of SharjahSat-1 with visible imager optics and baffle could be seen in Figure 3.32 and Figure 3.33.

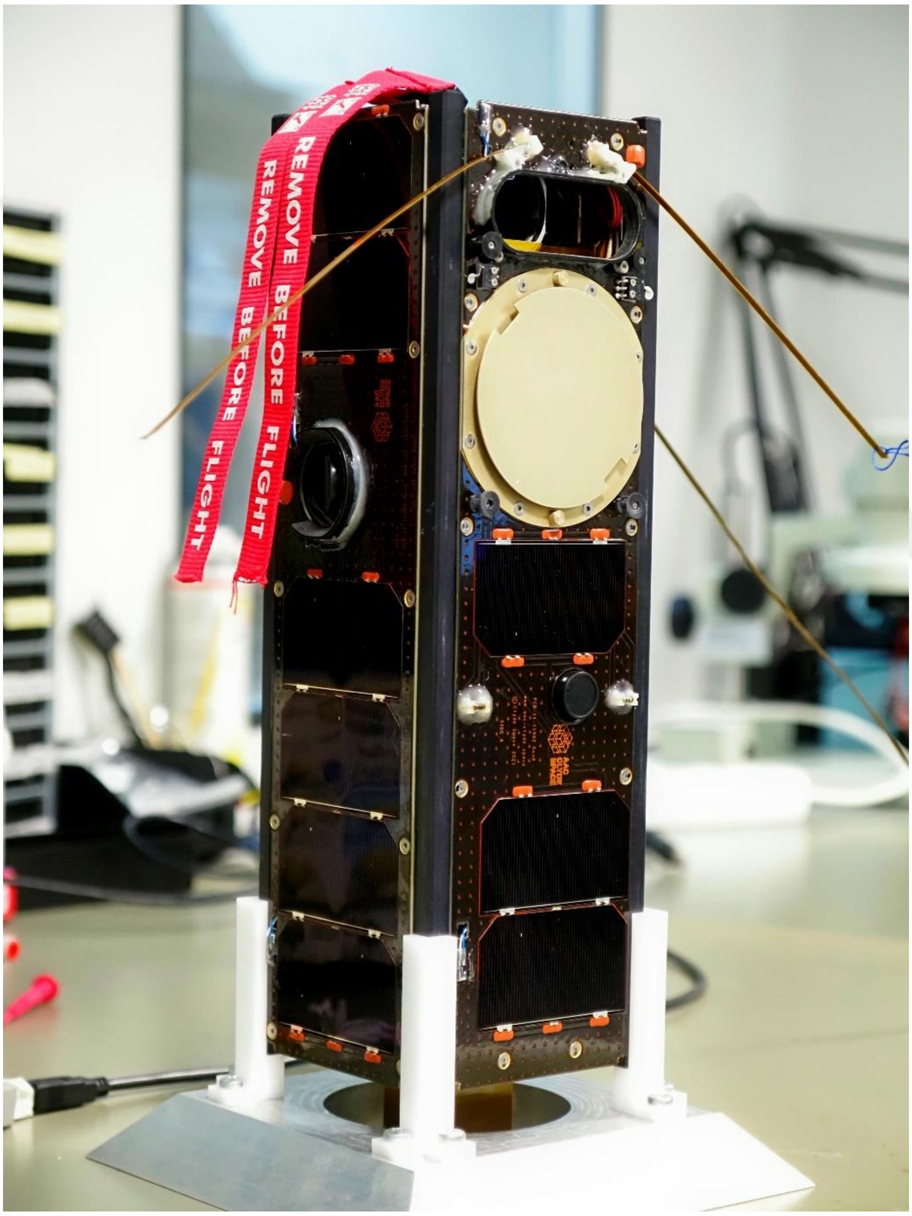

Figure 3.32 : SharjahSat-1 flight model.

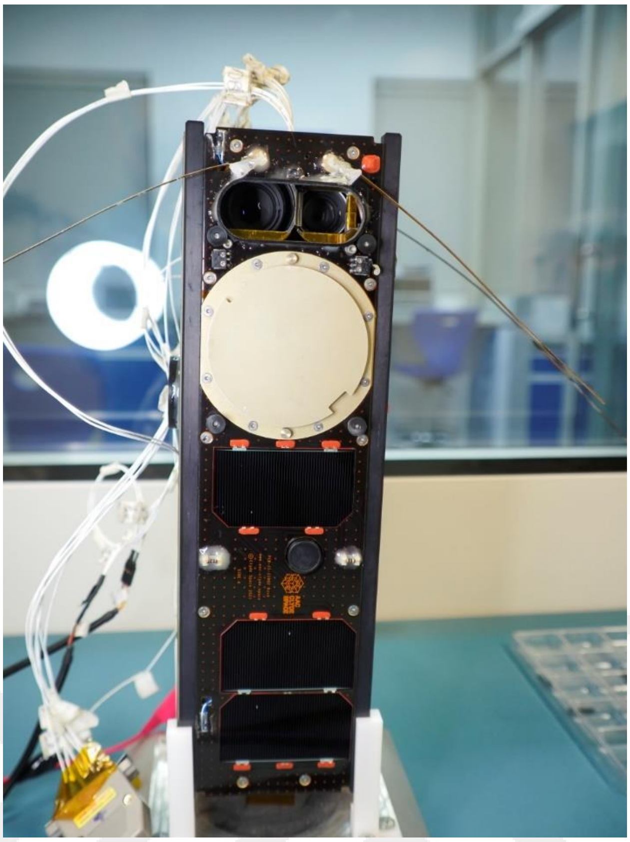

Figure 3.33 : SharjahSat-1 flight model with umblical connection.

3.2.5 Flight software design

A Real Time Operating System (RTOS) is used for obtaining a predictable (deterministic) execution pattern using a utility called the scheduler. The RTOS used in SharjahSat-1 is FreeRTOS, which is a small yet powerful RTOS that can run on a microcontroller. With the operating system on the satellite, it is possible to schedule missions, set imaging parameters, and geodesic position. Record these images, and its auxiliary data; and then download them whenever suitable for orbit. Satellite OBC communicates with imagers via I2C and SPI. I2C is used for setting image parameters, and SPI is used for data transmission. Image data could be stored on OBC in raw or JPEG format. Image parameters such as sensitivity, white balance, and integration time could be assigned as desired.

When imager is powered on, some default registry values are downloaded to imagining sensors. Then depending on the settings selected on the mission planning stage, different registry values are downloaded to imaging sensors from OBC memory. Some of the values entered from ground station (such as integration time, sensitivity) are uplinked to OBC as numeric values and then converted to registry data. Default registry values, and values for different configurations are provided by sensor board manufacturer.

3.2.6 Image metadata

During image capture following data is recorded on storage units of OBC with the captured image.

- All camera settings
- Time stamp
- Nadir panel temperature
- Attitude angles
- Angular rates
- ECI position
- Commanded angles
- Estimated fine angular rates
- Fine sun vector
- Coarse sun sensor raw data

This information is recorded for calibration of the imager, and for any possible future work from this data.

3.2.7 Ground software design

3.2.7.1 Ground testing software

During development of imaging payload, development of the payload were done outside of satellite OBC. With ground testing software shown in Figure 3.34. It was possible to test imager parameters, set lens focus, and test programming without the need for the whole satellite. Firmware developed with help of this software is then implemented on the satellite firmware.

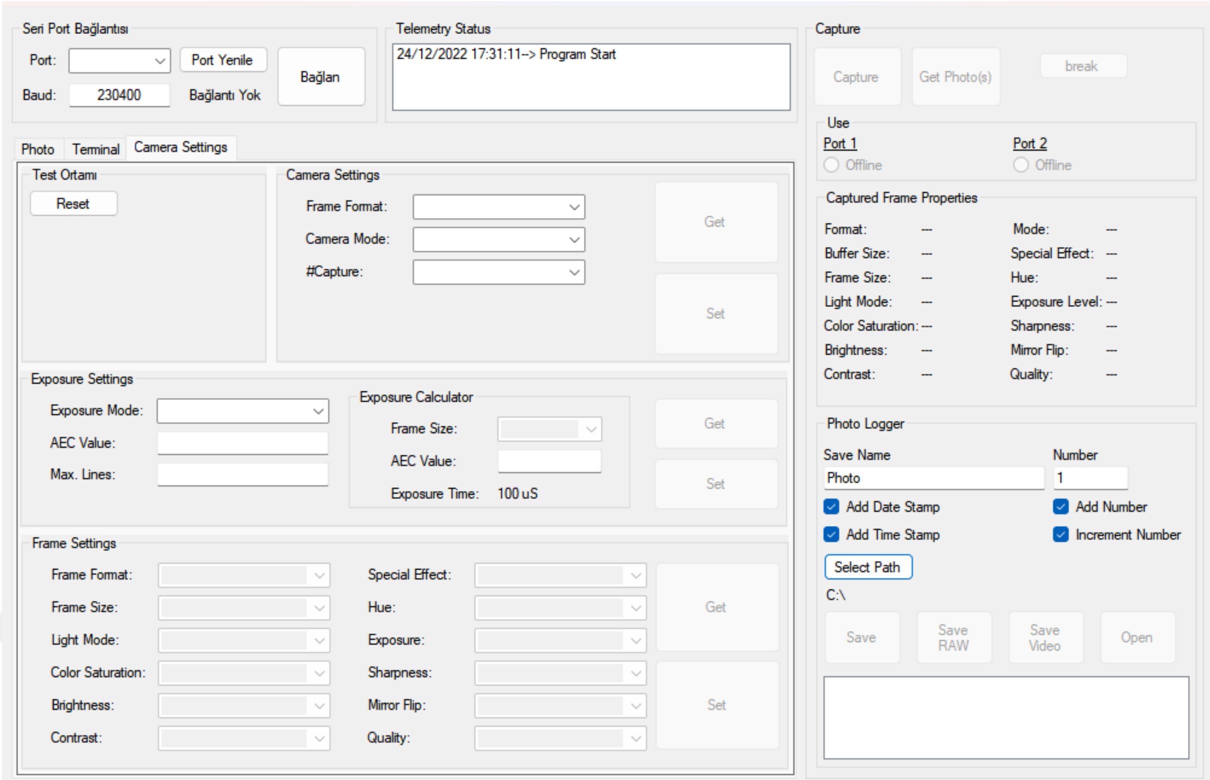

Figure 3.34 : Ground testing software for SharjahSat-1 secondary payload, development build.

This software works with camera testing PCB, which has connections to imaging sensors for 5mp and 2mp. This board provides connections for image downloading, and firmware debugging.

3.2.7.2 Ground station software

SharjahSat-1 satellite has its own custom ground station software to interact and assign missions. This software allows to uplink mission details for imaging payload by allowing selection of coordinates of the point of interest and various image settings as shown in Figure 3.35. Images could be requested from file system via S-band or VHF band during ground station passage.

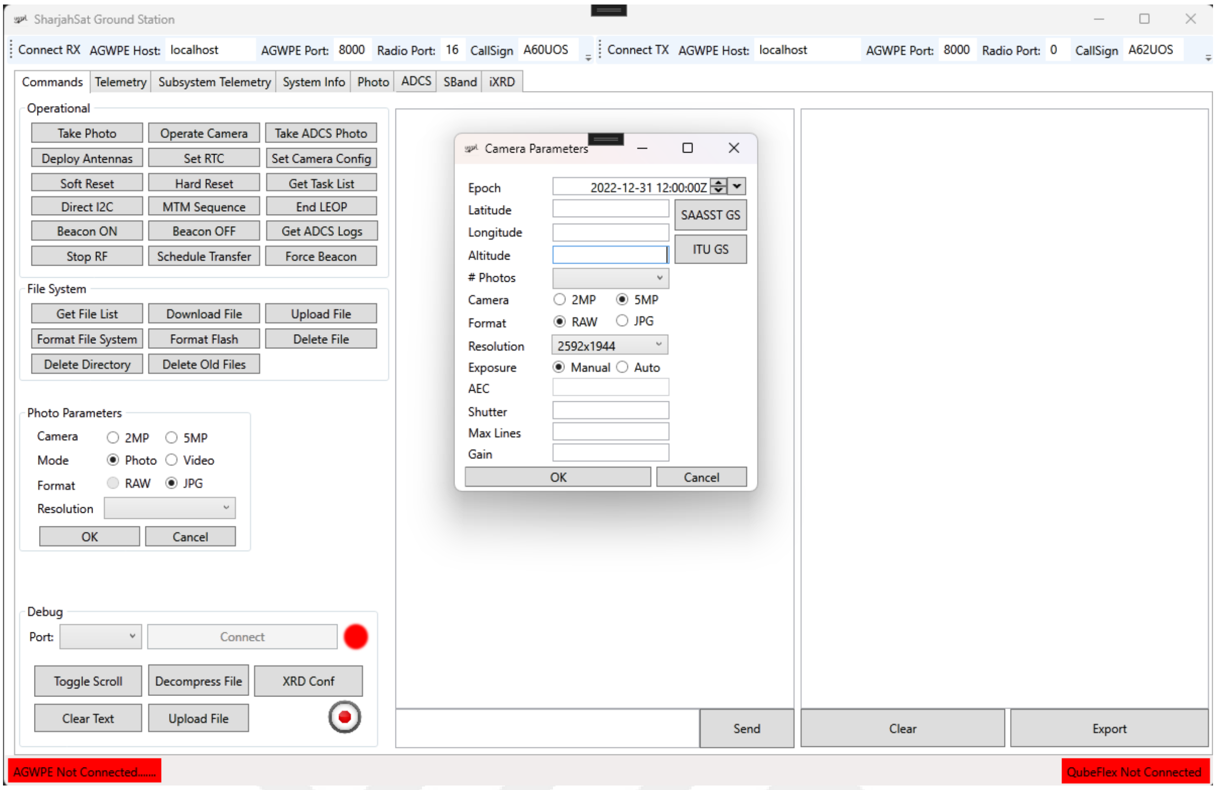

Figure 3.35 : SharjahSat-1 ground station software development build.

3.2.8 Manufacturing

Imaging payload of SharjahSat-1 required one of the most precise manufacturing on the satellite to keep parts secure and optically aligned. Design geometry of the lens holding pieces, and requirement for precise mounts, and selection of the materials proposed a challenge. Being a secondary payload meant budget to be modest. Modest budgets with precise requirements results in longer lead times. Possible supply chain issues, and time constraints of the SharjahSat-1 meant no rooms for mistakes, and manufacturing of both flight and engineering models at once. This helped with cost reduction. In Figure 3.36 manufactured metal parts could be seen, and in Figure 3.37 manufactured and assembled PCB could be seen.

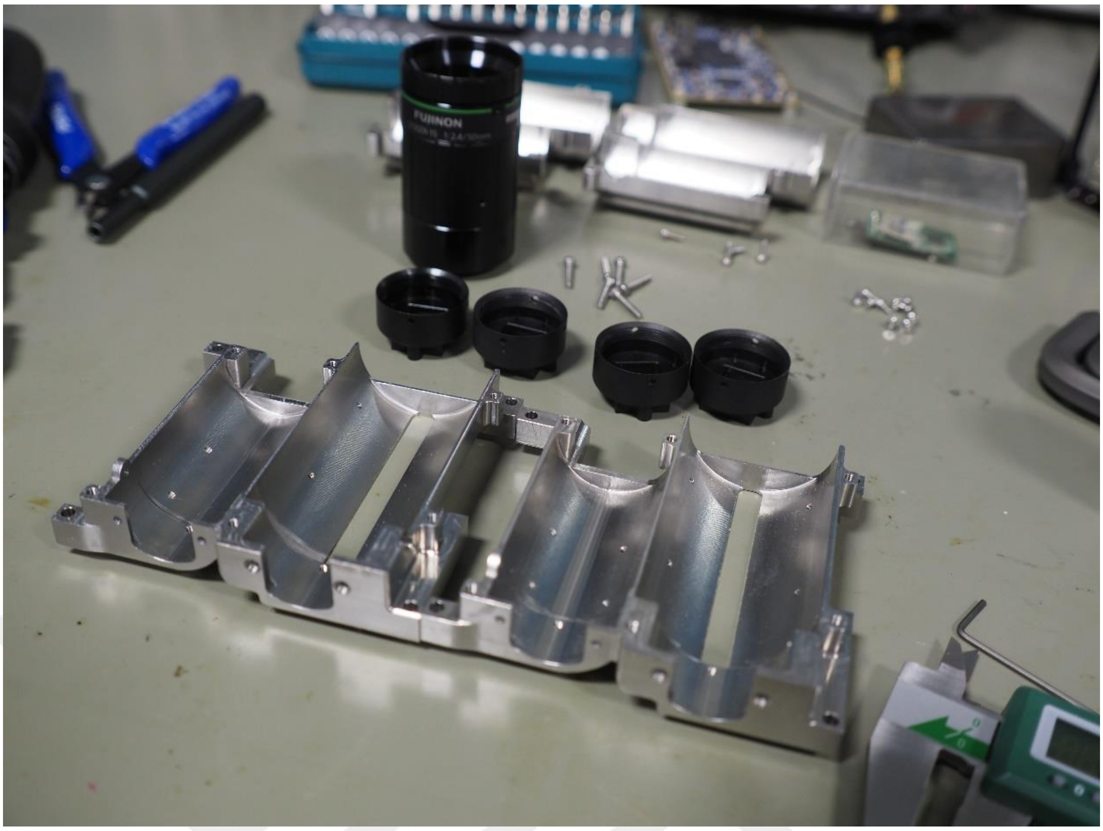

Figure 3.36 : Manufactured metal parts for the payload.

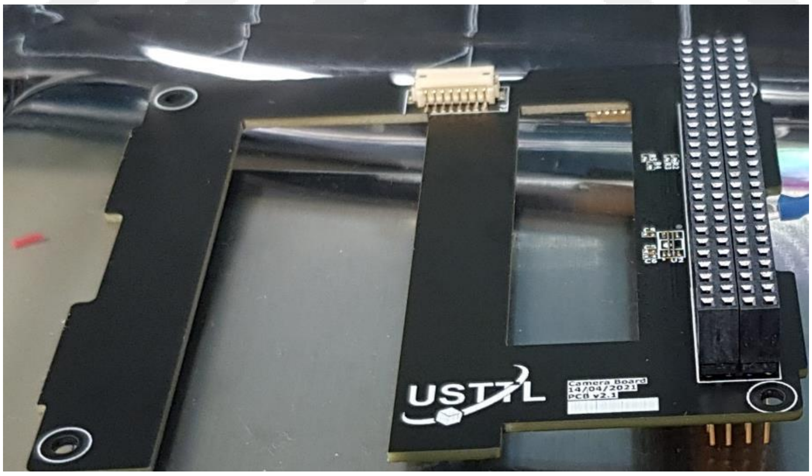

Figure 3.37 : PC104 board for imaging payload.

PCB manufacturing is a faster and a cheaper process it did not pose a threat. Delrin piece to be used as a baffle were easy and cheap to manufacture locally. Piece is shown in Figure 3.38.

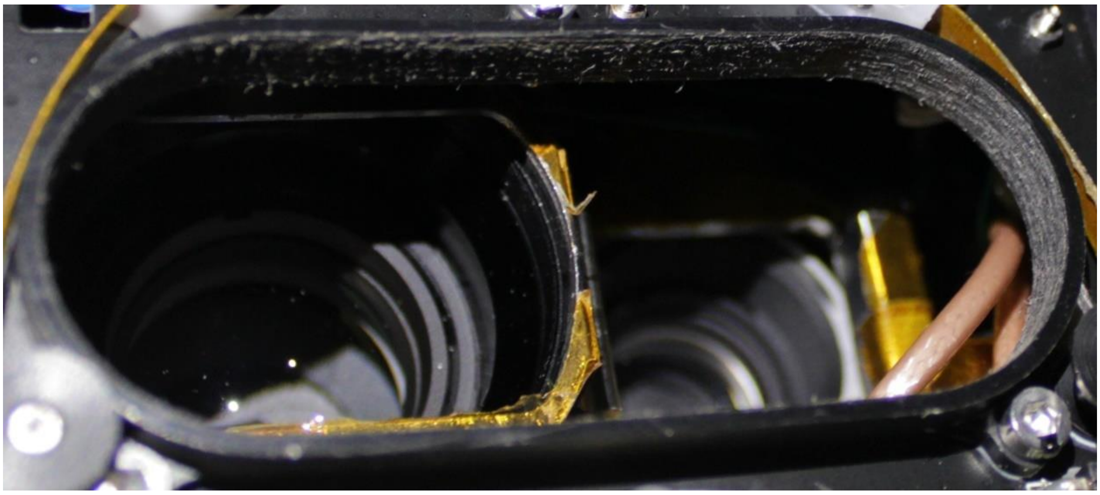

Figure 3.38 : Delrin baffle for the payload.

Manufacturing drawings, which show fastening points, tolerances etc. For metal pieces could be seen in Figure 3.39, Figure 3.40, Figure 3.41.

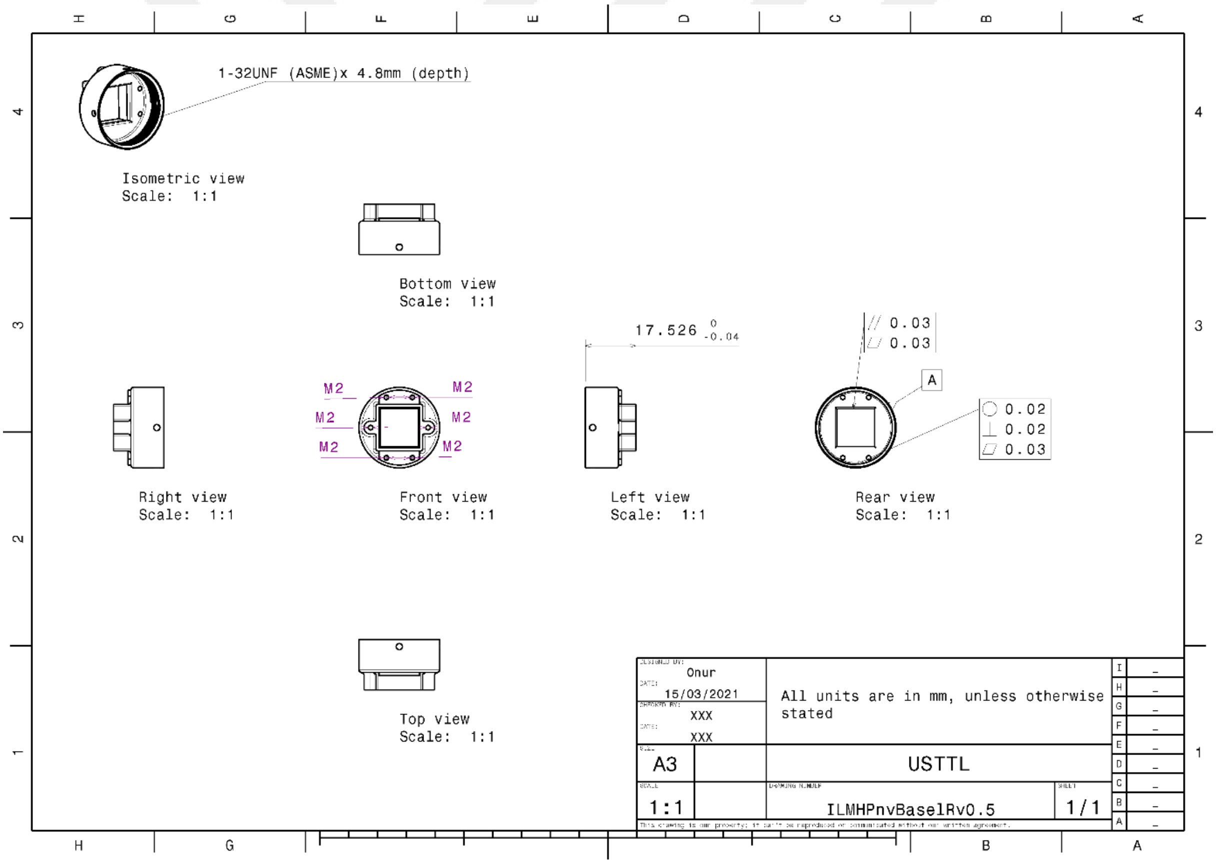

Figure 3.39 : Technical drawing showing threads, and out of standard tolerances for ILMH.

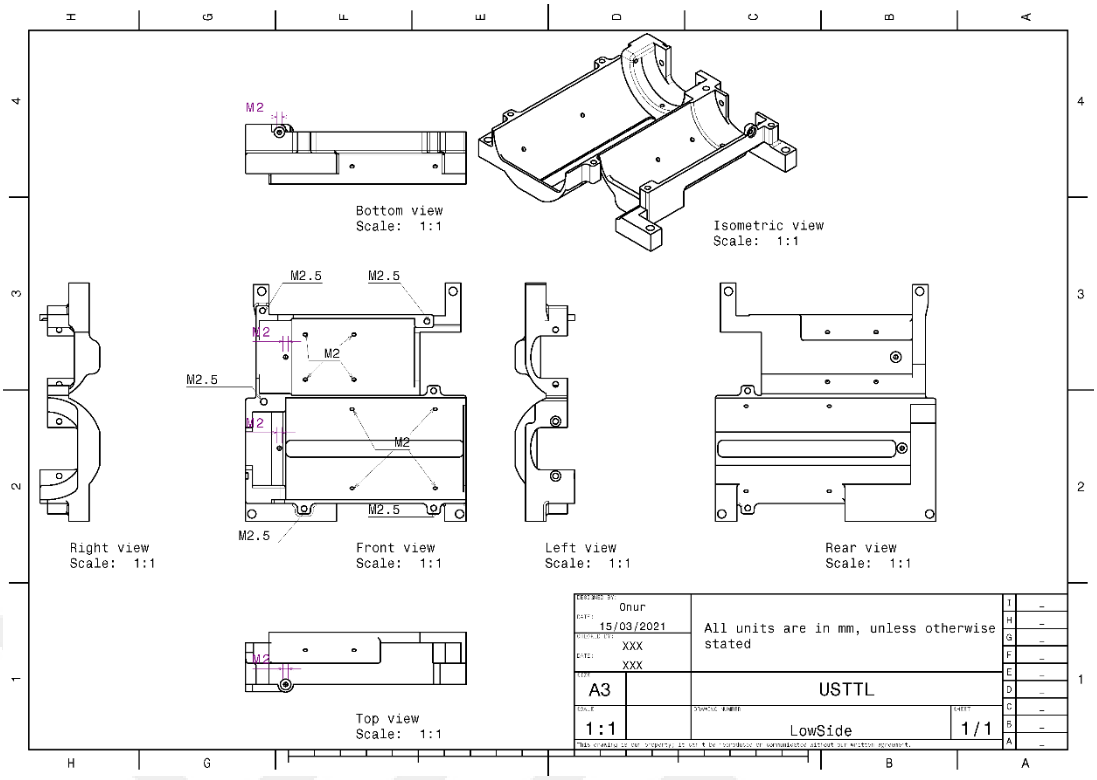

Figure 3.40 : Technical drawing showing threads for low side part.

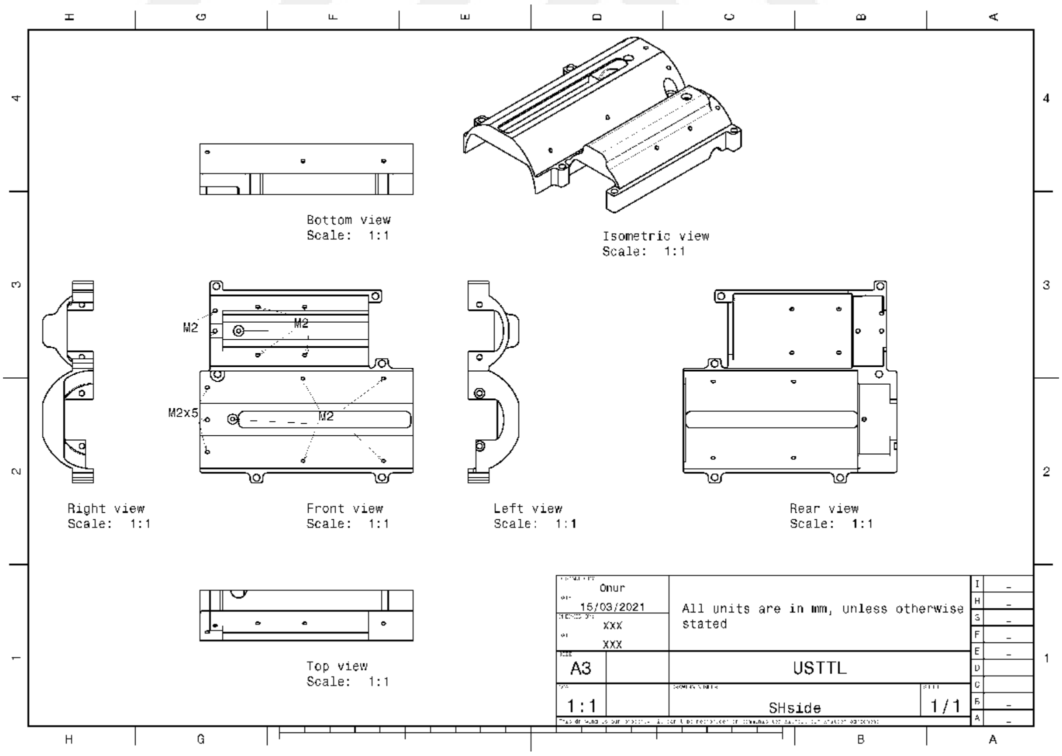

Figure 3.41 : Technical drawing showing threads for high side part.

3.2.9 Assembly

Assembly processes are explained under two sections. Assembly of the second payload at satellite level is designed to be as simple as possible to not disturb rest of the satellite assembly. Instrument assembly is done such in a way it requires setting of the focus during mid assembly, and then completing rest of the assembly. Changing focus after assembly is not possible. Details are explained in further sections.

3.2.9.1 Instrument assembly

Assembly of the imager is a delicate and long process. Length of this process comes from setting focus to infinity, modifications required on 25mm lens, and curing of adhesives used.

In Figure 3.42 technical drawings of all of the parts used on imager assembly could be seen.

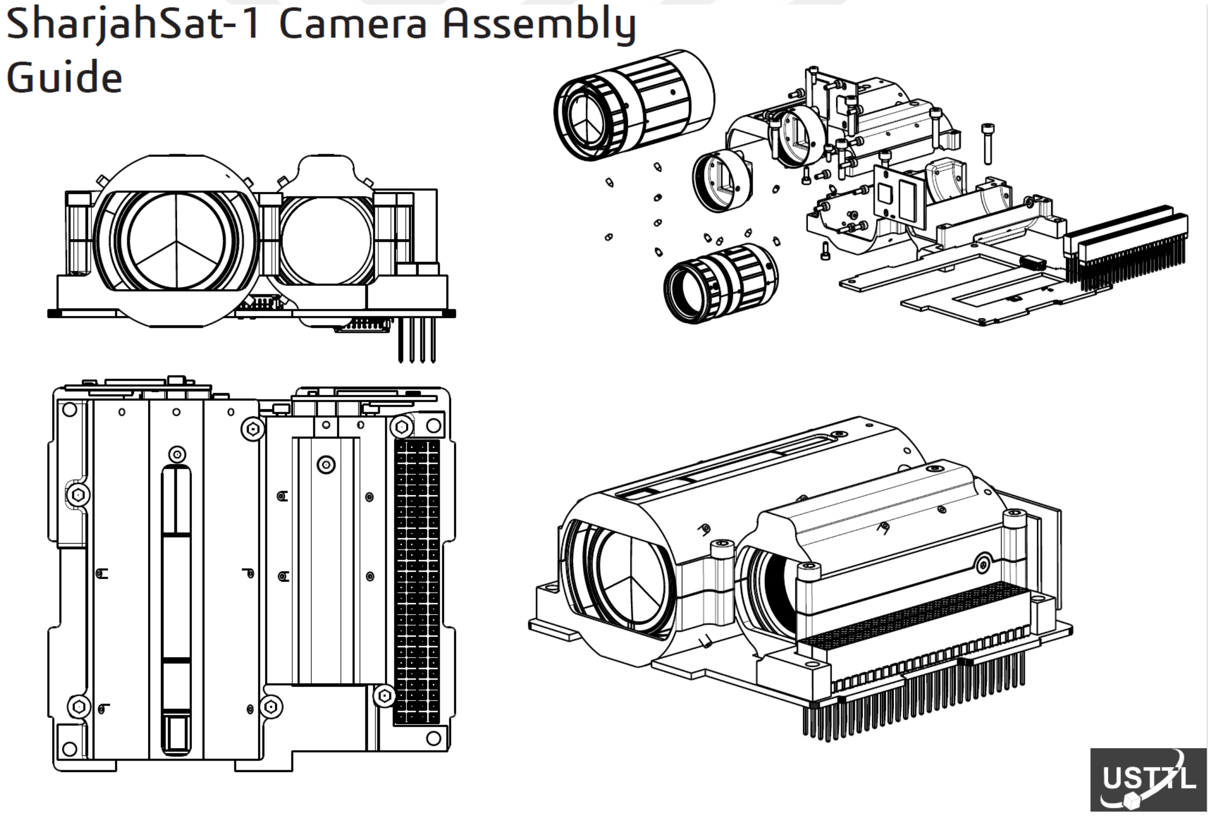

Figure 3.42 : All parts required for assembly of the imager paylaod.

Before starting assembly HF25HX-1S lens needs to be sanded down to fit when focused infinity. comparison between grounded and not grounded lens is shown in Figure 3.43.

Figure 3.43 : Sanding of HF25HX-1S.

After grounding lens is visually examined and cleaned with a binocular microscope as shown in Figure 3.44.

Figure 3.44 : Visual examination of HF25HX-1S.

At this stage system is half partially (shown in Figure 3.45) and send to an optical lab for infinity focus with an infinity collimator.

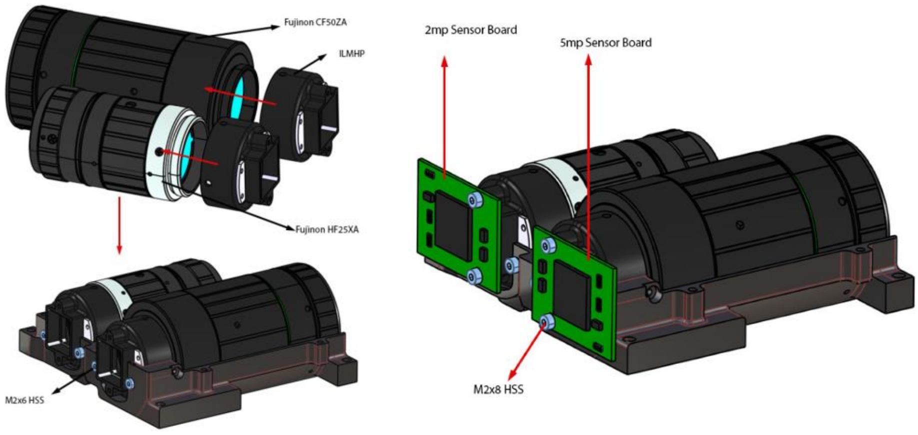

Figure 3.45 : Partially assembled system for focus setting.

After focus is set for the lenses they are set in to place by set screws with thread lock as shown in Figure 3.46, and Figure 3.47.

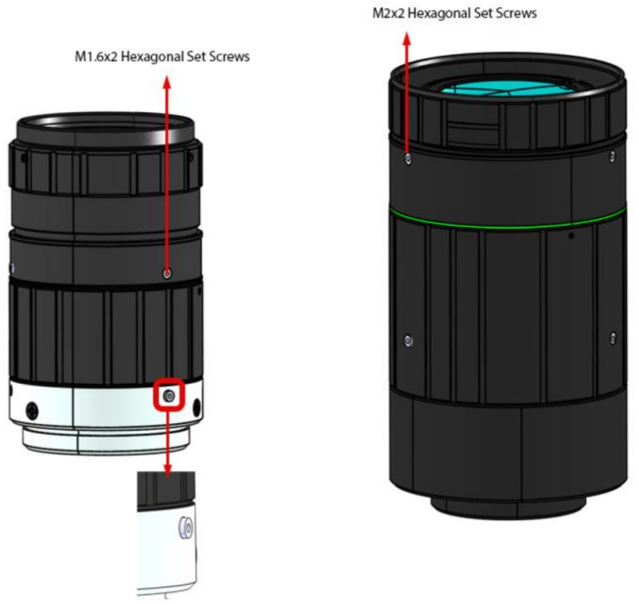

Figure 3.46 : Fastening points on the lens bodies.

Figure 3.47 : Thread lock applied to a set screw.

Assembly is then continued with steps similar to Figure 3.45, and continues with fastening process for the general system fasteners as shown in Figure 3.48, and fastening of set screws for the general structure precedes as being shown in Figure 3.49. During these steps, adhesives are applied to necessary parts of the structure and lenses.

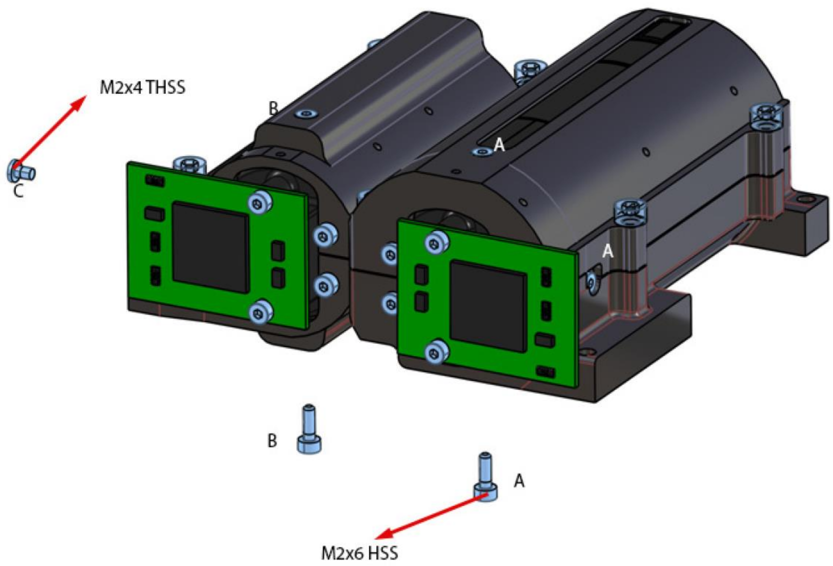

Figure 3.48 : General assembly of imager payload.

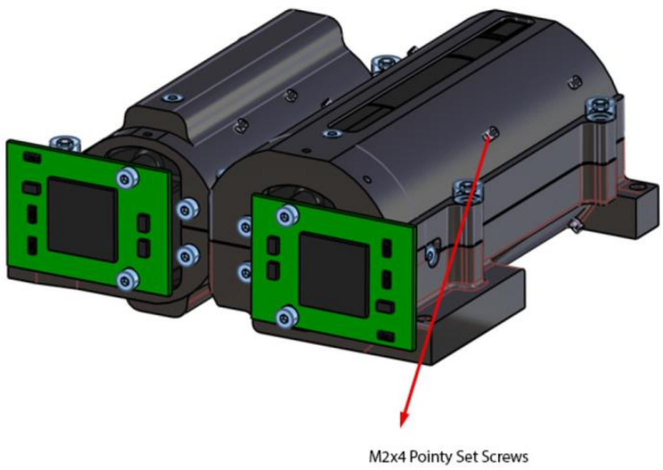

Figure 3.49 : Set screws for general structure of imager payload.

Figure 3.50 : Torquing of fasteners.

Every fastener on the system is torqued down to their designed values, which is determined according to general conventions, experience, stiffness, and fastener's material. Example of torquing could be seen in Figure 3.50. Most of the fasteners of the system are marked with a permanent marker that could be seen in the back side image of the system in Figure 3.52. This is done to aid visual examination of the system after vibration testing. These marks are only checked if there is a problem after vibration testing, or some other problem that requires disassembly of the satellite. Assembled images of the real system could be seen in Figure 3.51, Figure 3.53, and Figure 3.54.

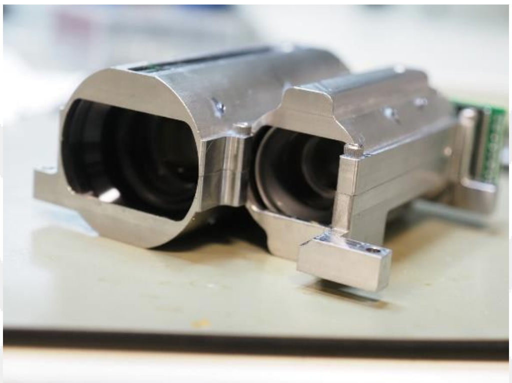

Figure 3.51 : Assembled imager without PC104 PCB.

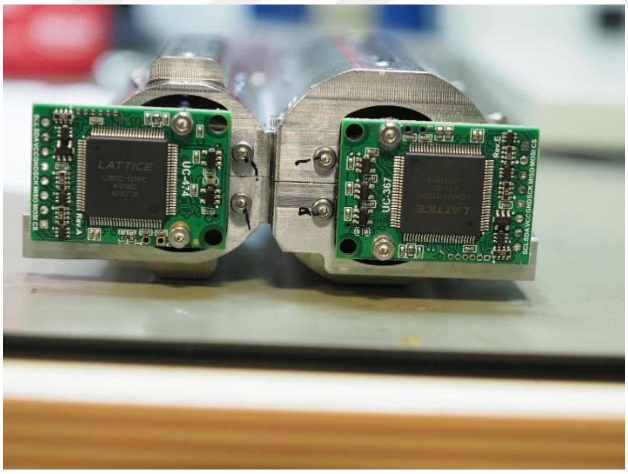

Figure 3.52 : Assembled system that shows sensor boards, and markings of the fasteners.

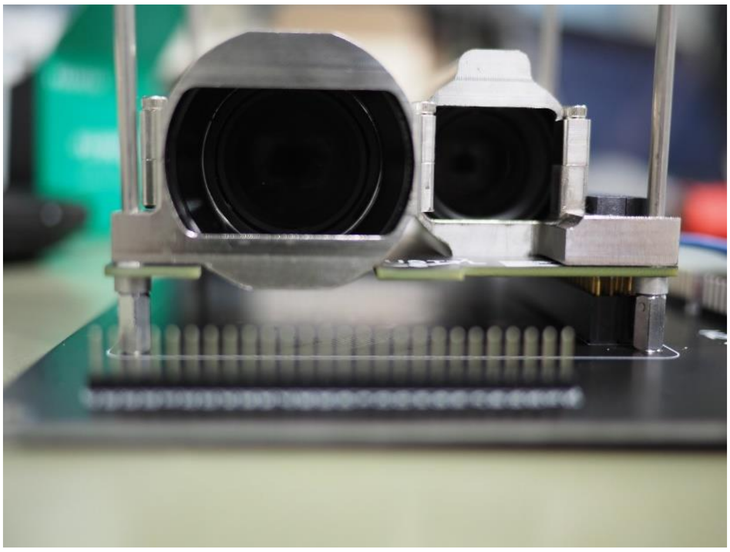

Figure 3.53 : Front image of the system with PC104 PCB on general development board.

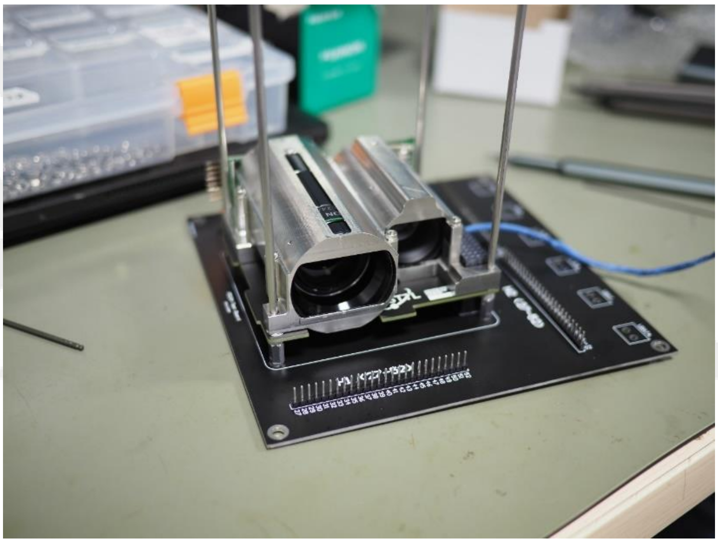

Figure 3.54 : General image of the fully assembled system, without polyamide tape applied.

After these steps, polyamide tape is applied to imager surfaces. Imager is then visually examined for dust contamination on optics, and dedusted and cleaned under binocular microscope while using clean room grade optical surface cleaners etc.

This prepares system for satellite assembly.

3.2.9.2 Satellite assembly

In this section only a small portion of the satellite assembly will be discussed as it is out of space for this work. SharjahSat-1 have 2 stacks. One on the Z side houses imager payload. Payload in Upper stack could be seen in Figure 3.55. Because of its position, there are many cables routing around this system (umbilical connector cables, RF cables) this is a limiting factor, and special care must be shown so that solar panels will not be harmed from any cable being stuck between satellite frame and the panels, or any cable getting harmed.

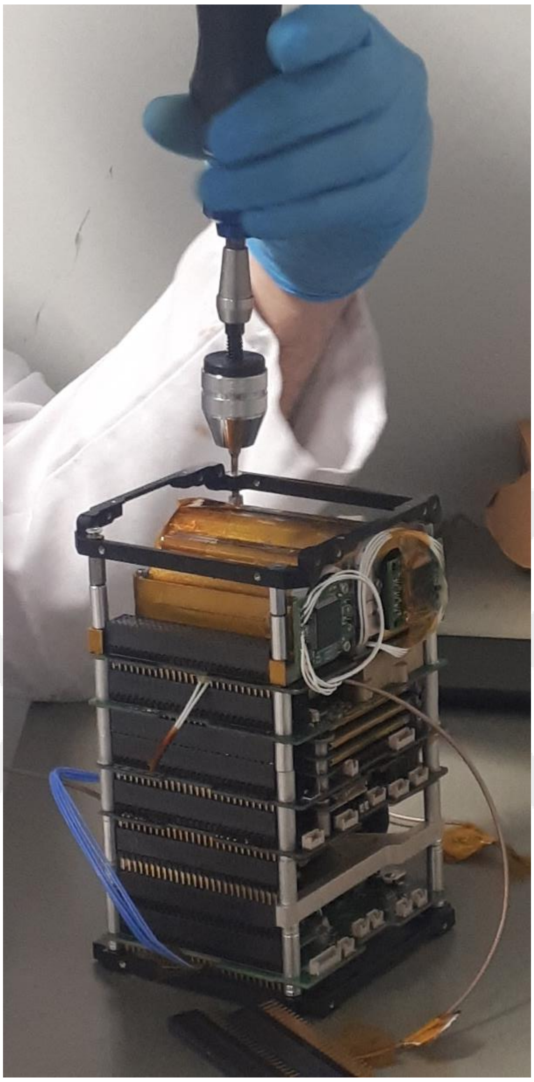

Figure 3.55 : Integration of Upper stack (imager stack) of SharjahSat-1.

3.3 Instrument Calibration

Any kind of imager requires some calibration. Some calibration could be done in software, and during satellite mission; but, some of them are required to be done in hardware. SharjahSat-1 imager does not include any capability to adjust its hardware after launch, reasons for this have been explained before. Payload is optics are focused during assembly processes, and details are explained in the next section.

Depending on the satellite bus capabilities, some image parameters are required to be calibrated this is explained in further sections under “Imaging parameter calibration”.

Payload has the ability to capture an image in JPEG format, which in self could provide many kinds of useful information for photogrammetry, and does not require any significant calibration. Additional value could be added to images by simple image manipulation. Alas, images captured as a raw data provides more data, but requires to be processed.

3.3.1 Focus calibration

To achieve a sharp image optics needs to be focused. This focusing needs to be done while in ground testing process. Due to the altitude SharjahSat-1 operates at, and pixel pitch size of the imager if optics of the payload are focused to infinity (parallel rays) it would give as sharp as an image possible for this system. This is done with an infinity collimator, during assembly step of the payload. For engineering model, payload partly assembled is calibrated in an optical lab. After environmental tests are done satellite is brought back to the same infinity collimator to test for any precipitable difference in focus of the system. For flight model payload, it is calibrated during assembly but system is not brought back for additional testing. In Figure 3.56 SharjahSat-1 engineering model could be seen in the optical lab pointed to an infinity collimator after environmental tests.

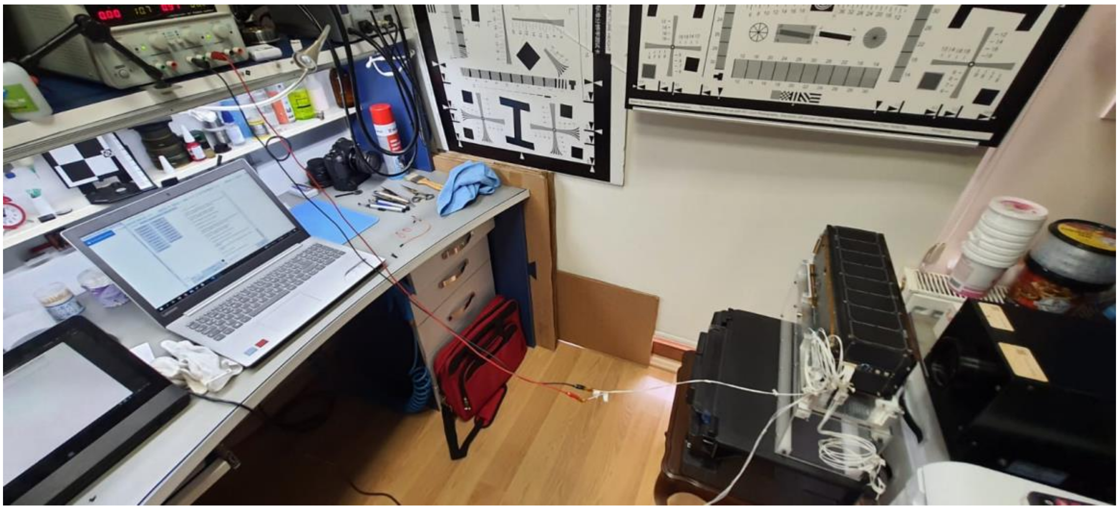

Figure 3.56 : SharjahSat-1 engineering model optical payload being tested with an infinity collimator.

3.3.2 Imaging parameter calibration

There are some critical parameters that needs in space adjustments to get the best quality image, these parameters could be calculated to some extend but it is best to calibrate them with operational data.

Depending on the capture image data’s format some settings such as white balance might need some adjustment, if imaging sensors would fail to atomically determine best possible white point. Even if white point is off by some small amount, it is something that could be solved without much image quality loss.

Integration time for a space borne imagers are critical; because, it affects many of the image parameters such as SNR. It is important to select this value as long as possible while not showing satellite bus movement on the captured image. Rough value for this is calculated to be 1ms for SharjahSat-1 5mp imager, but it will need adjustments from data incoming. Since this value does not include satellite bus perturbations.

Imager gain is also a variable that requires calibration. This value should be kept as low as possible while providing a suitable image exposure.

3.3.3 Raw image conversion calibration

Raw images require processing to get meaningful data. Raw image data from payload image sensors have 8-byte header. Other bytes are direct pixel values read from sensor (after black level calibration done) in 8bit format. This means data is in bayer pixel format. This requires de-bayering to get accurate colour information. There are many different kinds of algorithms to achieve highest quality images with most data. Some of these algorithms are open source while some are specific to raw editing software. While it is also possible to directly use this data with a remote sensing software, it requires different kinds of processing. Sensor board is claimed to be support 10bit raw image capture as well, but later on manufacturer revealed it was not supported.

4. TESTING AND EXPERIMENTS

During development of the imaging system some tests and experiments are done to validate the system. Some of these tests are done on payload level while others are done on the whole satellite. Further sections will explain these experiments, and tests. Results will be shown as well.

4.1 Test Setups

4.1.1 Test setup

After the first design's lenses were not able to provide enough quality, selected lenses Fujinon HF50HA-1S and Fujinon HF25HA-1S required testing. This time testing were done inside USTTL facilities to have more controllable environment. Lens and sensor combos are placed on a development prototype board as shown in Figure 4.1, and images at full resolution were taken at different F numbers with sufficient integration time. Target images are at 1443cm distance. This images are then examined on computer to determine if the image quality would prove itself to be sufficient enough.

Figure 4.1 : Test setup for lens testing.

During testing it was found that best quality images (sharpest) were at F4 (which was also stated in the lens manuals [40, 41]). In Figure 4.2 image taken with 50mm lens at 5mp resolution is shown. In this figure close up from original image for the upper region of the secondary page of the target is shown.

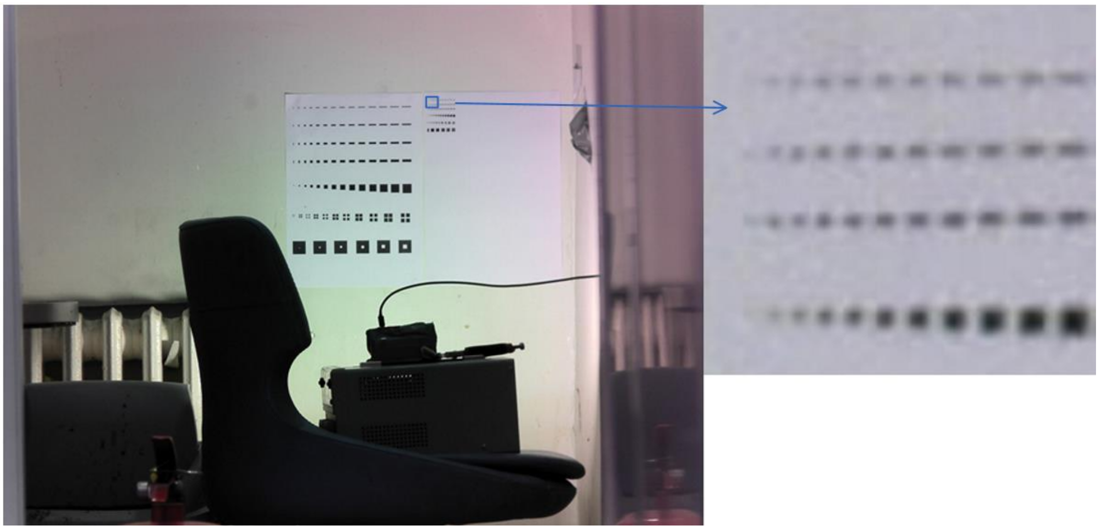

Figure 4.2 : 50mm F4 1443cm – CF50ZA-1S @ 2592x1944px.

In Figure 4.3 sizes for squares on the target is shown. From Figure 4.3 and Figure 4.2 it could be determined that 0.5mm details could be seen while 0.25mm details could not. From these results it was conducted that, Fujinon HF50HA-1S and OV5642 imaging sensor are a good match for their purpose.

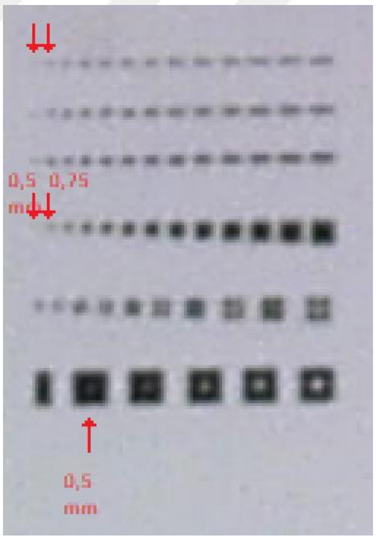

Figure 4.3 : Target sizing.

From Figure 4.4 and Figure 4.5 it could be determined that although not as sharp as Fujinon HF50HA-1S and OV5642 combo, Fujinon HF25HA-1S and OV2640 imaging sensor combo are good match for their purpose.

Figure 4.4 : 25mm F4 1443cm – HF25XA-1S @ 1600x1200px.

Figure 4.5 : 25mm F4 1443cm – HF25XA-1S @ 1600x1200px – larger detail close up -.

With these results, lens and sensor combos are tested under TVAC as explained in next section.

4.1.2 TVAC

Multiple TVAC tests have been conducted. First one is done on the lens and sensor combos to validate their high vacuum suitability. Then environmental tests for EQM and FM were conducted. For EQM and FM after TVAC test, vibration testing is done which is explained in the next section.

4.1.2.1 System validation TVAC test

In Table 4.1 requirments that are needed to be verified for accepting ststem to be suitable for high vacuum enviroment is shown.

Table 4.1: Requirement to be verified.

Thermal Vacuum Cycling test characteristics	
Min temperature	-20 ± 1 °C
Max temperature	+80 ± 2 °C
Temperature variation rate	≥ 1 °C/min
Dwell time	1 hour at extreme temperatures
Vacuum	< 1.3 x 10^{-3} Pa
Cycles	2

During this test images are taken at room conditions as reference, and then under high vacuum at room temperature, 80 °C and -20 °C. Images are compared with reference to see how much image degradation is present if any. In Figure 4.6, and Figure 4.7 general test setup could be seen. Where imagers are focused to a PCB target (target is high vacuum tested). Lenses and sensors are connected to a microcontroller board inside TVAC unit. Microcontroller is connected to a ground testing software outside of TVAC. Images are taken at maximum resolution of the sensors 2592x1944px and 1600x1200px for 5mp and 2mp sensors respectively.

Images are set up to be jpeg with automatic settings for shutter time, and ISO.

External light source (30 watt led) is deployed when capturing images. All systems were in working continuously for the duration of the test. In Figure 4.8 connected temperature sensors could be seen.

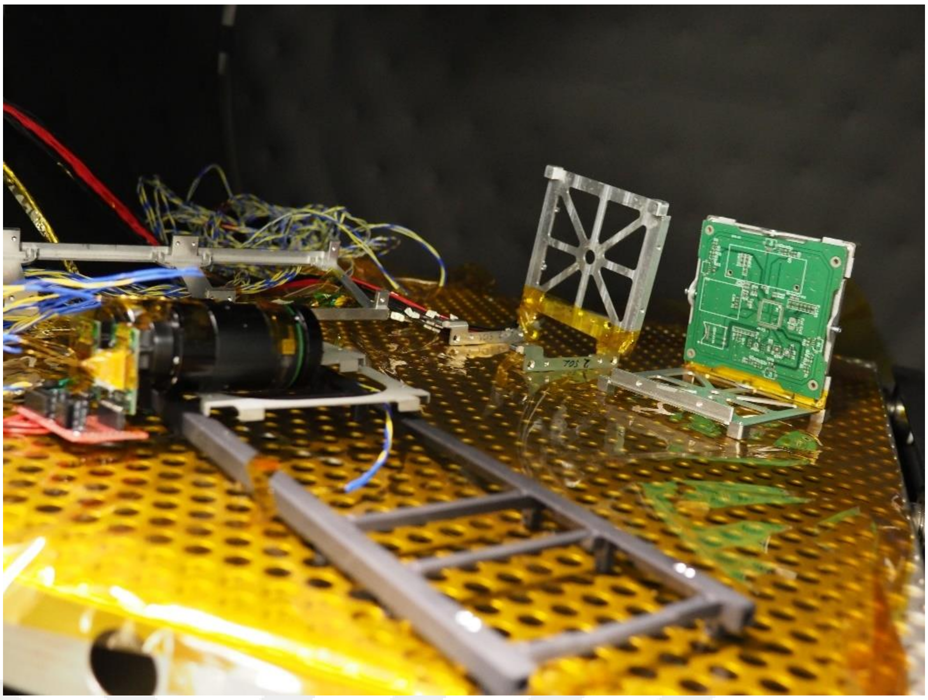

Figure 4.6 : TVAC test setup, with focus targets.

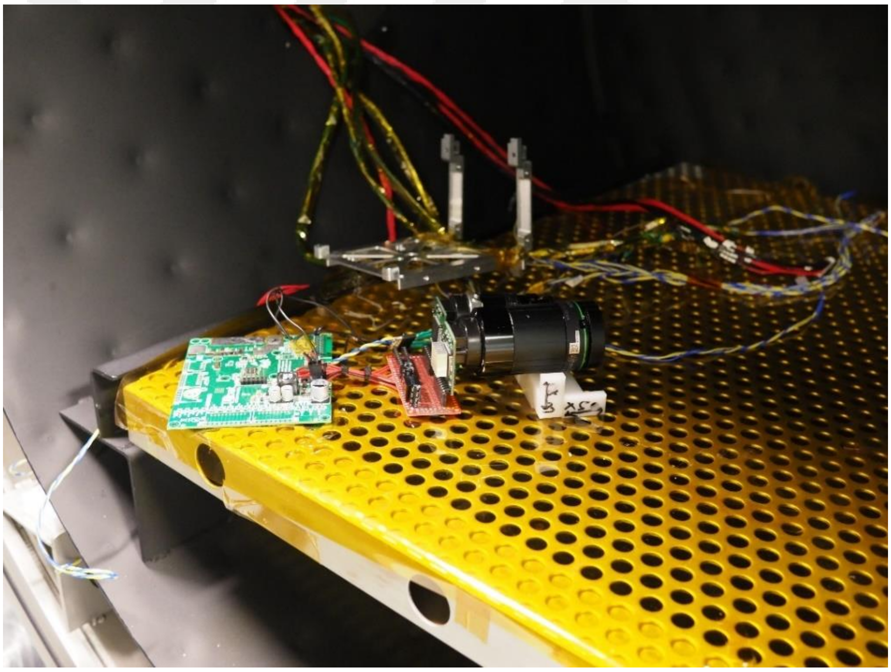

Figure 4.7 : Support PCB for imagers.

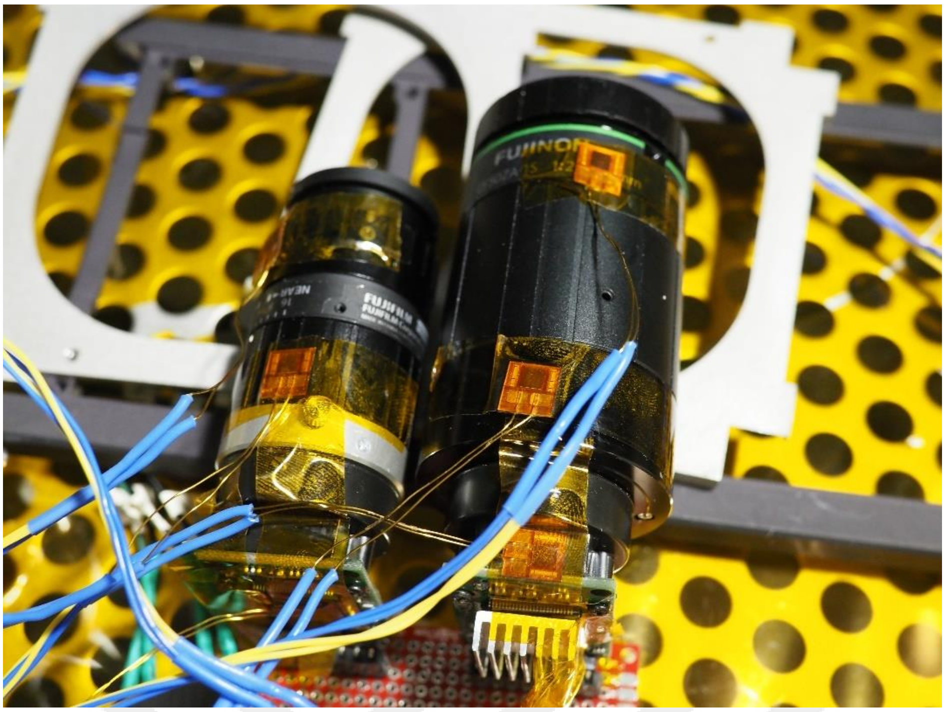

Figure 4.8 : Thermocouples connected on lens bodies, lens mounts, and imaging sensor boards.

Result for the 2mp imager could be seen in Figure 4.9. These results have shown that imagers are OK to use in vacuum, while too high temperatures result low SNR levels, temperatures under 0 could cause focus shifting. System returns to its initial state when temperature nears standard room temperature. Same results have been achieved for 5mp imager, but they are not shared in this document.

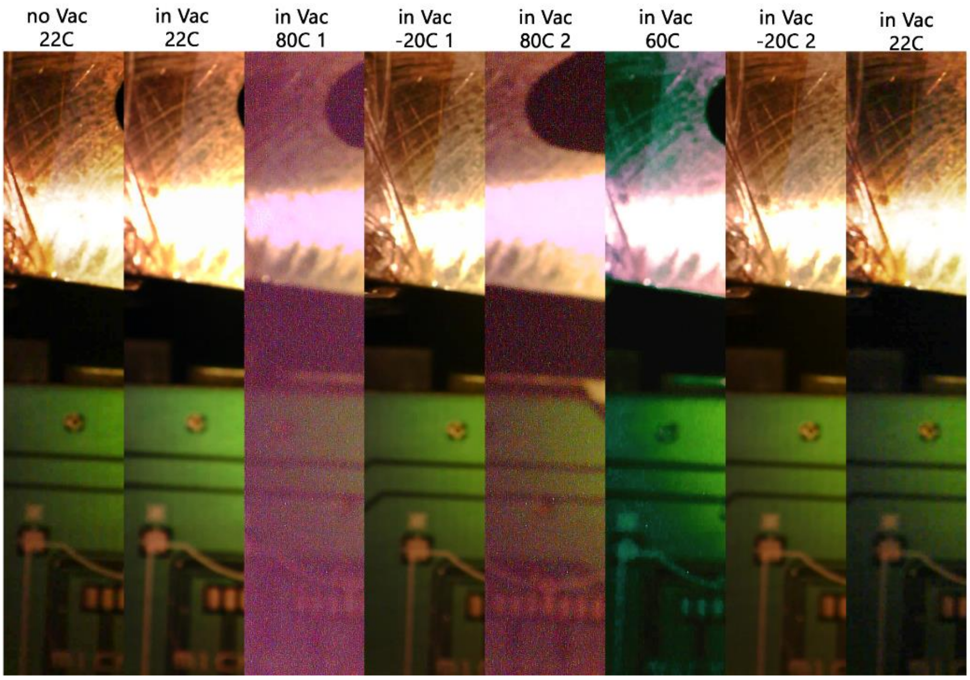

Figure 4.9 : 2mp imager comparison images, slight brightness and hue adjustments are done on captured images.

4.1.2.2 EQM TVAC test

EQM level tests are designed to be more challenging for the satellite. This is to test the satellite to its limits at worst possible scenario. EQM level TVAC test is done to whole satellite, while satellite is running a special branch of the firmware that is designed for this testing only. This software operates camera every 10 minutes. In Figure 4.10 SharjahSat-1 could be seen inside TVAC with its connections to outside of the chamber. In Figure 4.11 some thermocouple wiring could be seen.

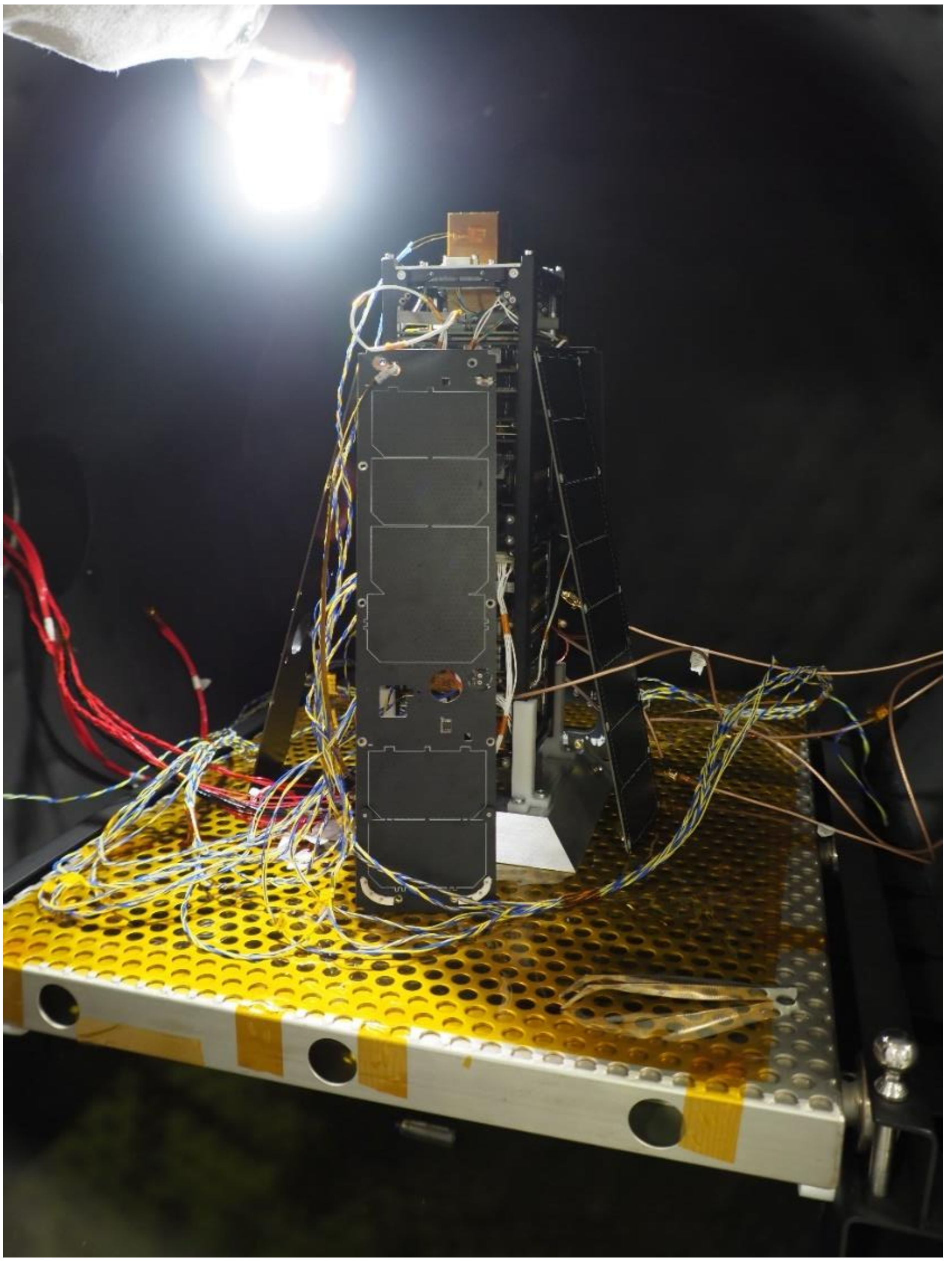

Figure 4.10 : SharjahSat-1 EQM before TVAC.

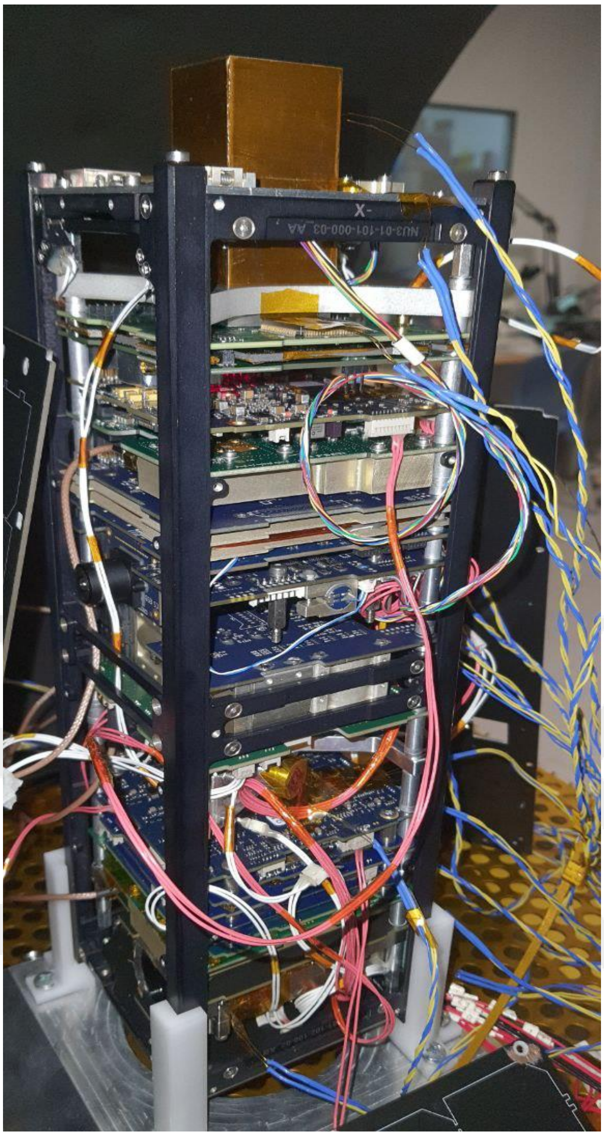

Figure 4.11 : EQM thermocouple connections.

In Table 4.2 testing characteristics could be seen.

Table 4.2: Thermal Vacuum Cycling test characteristics.

Min temperature	-30 ± 1 °C
Max temperature	+70 ± 2 °C
Temperature variation rate	≥ 1 °C/min
Dwell time	120 minutes at extreme temperatures
Vacuum	< 1.3 x 10^{-3} Pa
Cycles	8

After this test camera is confirmed operating, this test does not confirm if the lens focus is still same. Focus check is done after vibration testing of the EQM.

4.1.2.3 FM TVAC test

FM TVAC test is shorter and less stressful than what is done to EQM model. Where EQM model is tested for worst possible case, and to see failure FM test is done with less stress. There are no focus checks after FM environmental tests.

4.1.3 Vibration

Vibration testing for EQM model is at more harsh levels. It is assumed if satellite passes this test, FM testing will be without any issues. After EQM vibration test shown in Figure 4.12 satellite is brought to an optical lab to test for focus shift. This is not done after FM test to not contaminate FM model.

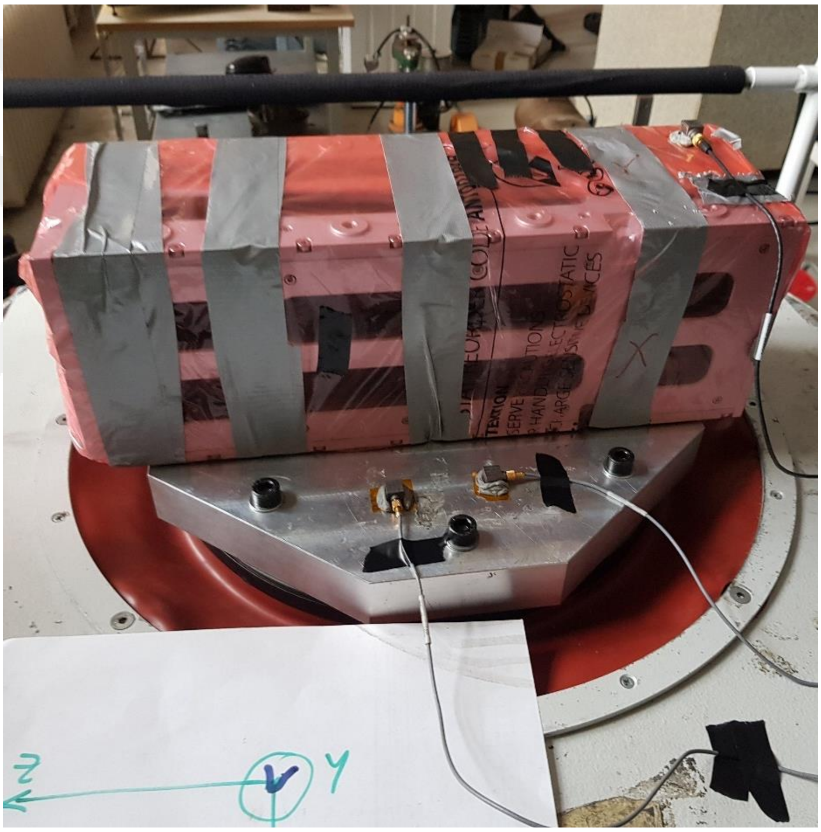

Figure 4.12 : SharjahSat-1 EQM model in vibration testing table.

4.1.4 Focus check

After environmental tests for EQM model is completed, system is tested for any shifts in focus, which would prove to be detrimental for the imaging quality of the payload. To test this SharjahSat-1 EQM model is brought into an optical lab, and pointed at an infinity collimator. Images taken with both 2mp and 5mp imager is then compared with the images taken before any environmental testing. Figure 4.13 and Figure 4.14 explains this further.

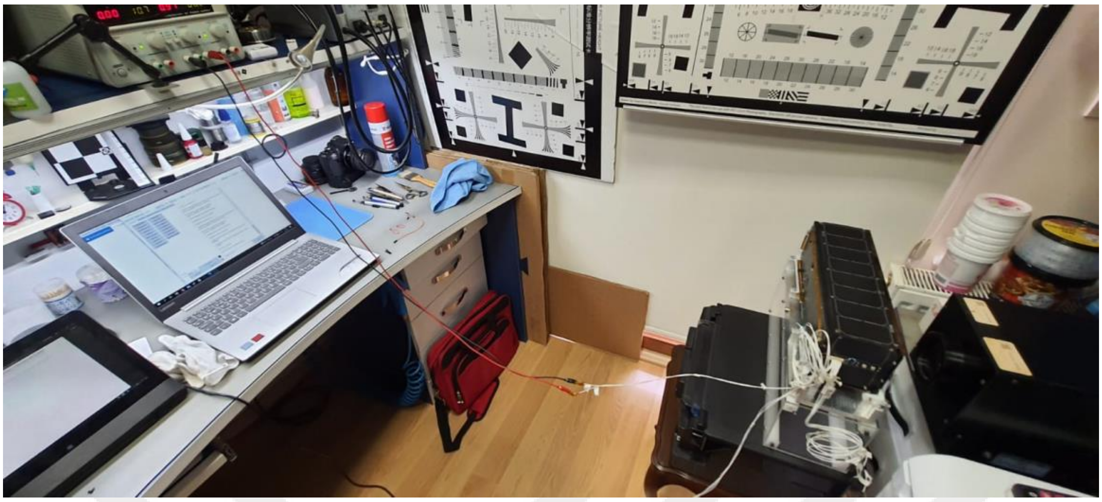

Figure 4.13 : SharjahSat-1 EQM inside an optical lab.

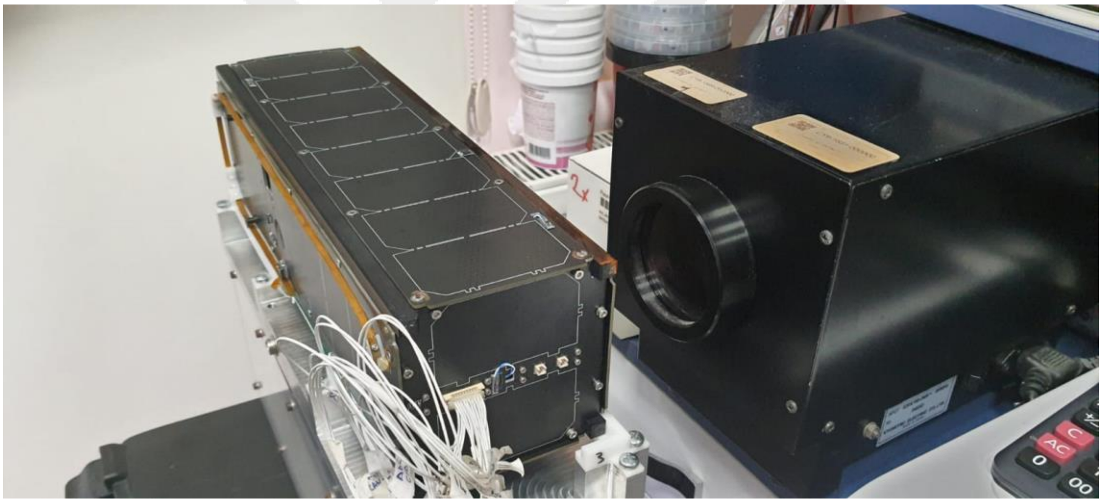

Figure 4.14 : SharjahSat-1 pointed at an infinity collimator.

In Figure 4.15 before and after results 5mp imager is shown. Examining images shows that there are no appreciable reduction in image quality from either TVAC or vibration testing.

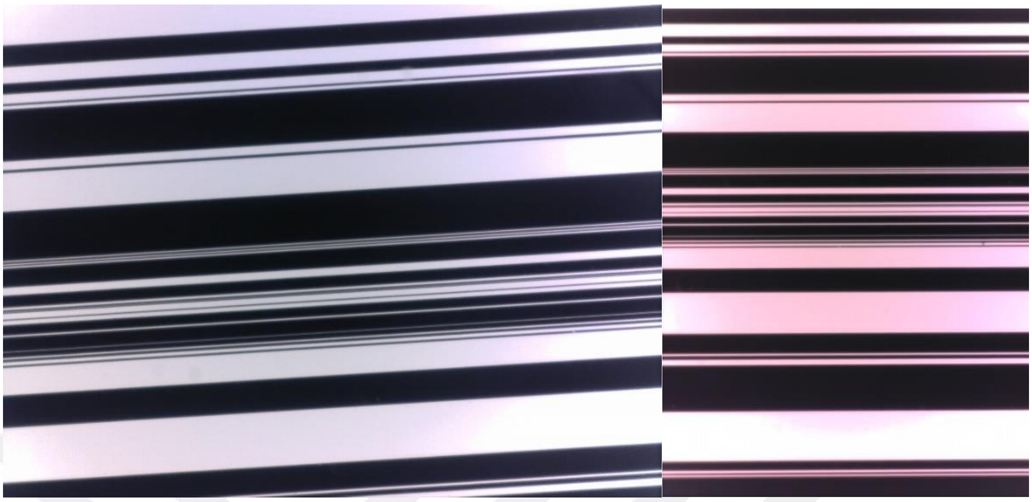

Figure 4.15 : Before and after (respectively) environmental testing for 5mp 50mm imager.

In Figure 4.16 before and after results 5mp imager is shown. Examining images shows that there are no appreciable reduction in image quality from either TVAC or vibration testing.

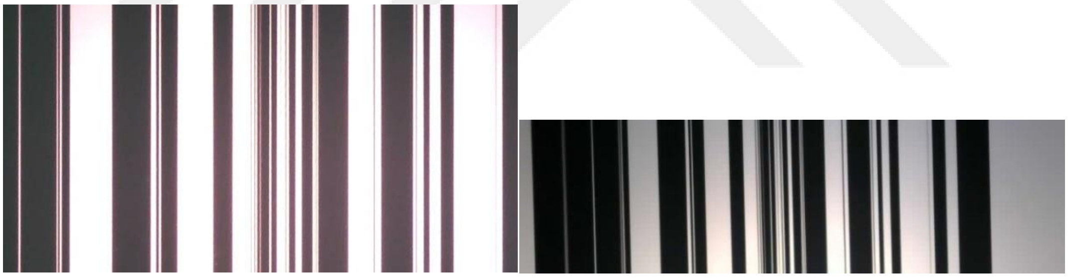

Figure 4.16 : Before and after (respectively) environmental testing for 2mp 25mm imager.

From these tests we can conclude that system is suitable for spacecraft usage. It can be assumed that same performance could be expected from FM payload as well.

4.1.5 Before flight checks

After all environmental testing, Imaging payload is covered to prevent contamination. Before launch satellite is tested for various tasks, these vary from setting of antennas, testing software, formatting memory structure etc. For imager, polyamide cover is removed. Optics are cleaned, and camera operations are tested with other subsystems, such as scheduling operation, image downloading via UHF, or S-band etc.

5. CONCLUSIONS AND RECOMMENDATIONS

Work presented in this thesis could be used to conclude that, it is possible to manufacture an imaging payload with dual cameras using CoTS parts while achieving satisfying image results. Tests done such as TVAC, vibration, imaging, after environmental testing etc. could be accepted to prove system's capabilities. It would be recommendable to conduct an MTF test to further detail imagers' capabilities. In orbit tests for MTF and calibration efforts are important future work as well. In Figure 4.17 an image taken in orbit with 5mp, 50mm imager at 1204x768px resolution is be seen. This image is taken for testing purposes, and downloaded in UHF band, there are some missing data packages, that could be seen as artifacts at lower right edge of the image.

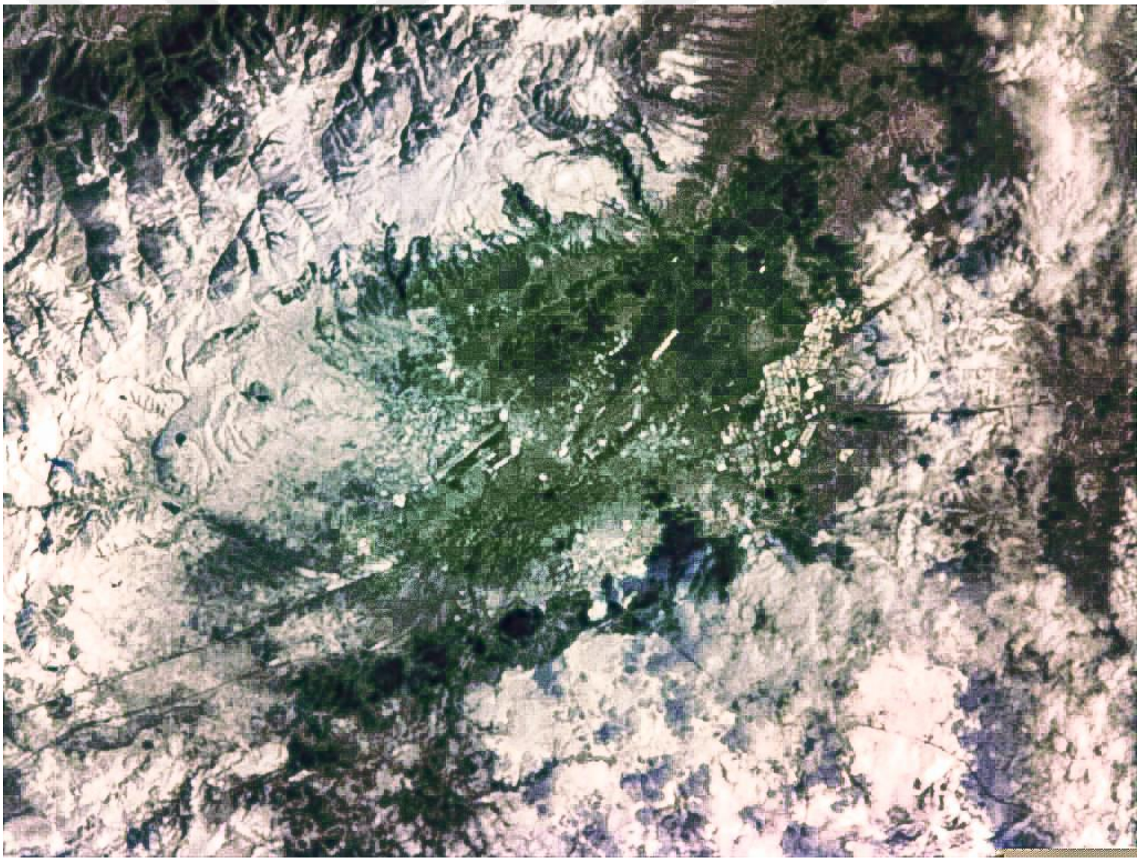

Figure 4.17 : In orbit image from system.

5.1 Practical Application of This Study

Results presented here could be used for development of other imaging payloads as well.

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