ISTANBUL TECHNICAL UNIVERSITY ★ GRADUATE SCHOOL OF SCIENCE ENGINEERING AND TECHNOLOGY

MISSION ANALYSES OF A DOUBLE UNIT CUBESAT BEEAGLESAT

M.Sc. THESIS

Çağrı KILIÇ

Department of Aeronautics and Astronautics Engineering

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JANUARY 2015

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Thesis Advisor: Prof. Dr. Alim Rüstem ASLAN

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İSTANBUL TEKNİK ÜNİVERSİTESİ ★ FEN BİLİMLERİ ENSTİTÜSÜ

BEEAGLESAT KÜP UYDUSU GÖREV ANALİZLERİ

YÜKSEK LİSANS TEZİ

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OCAK 2015

Çağrı KILIÇ, a M.Sc. student of ITU Graduate School of Science Engineering and Technology 511121201 successfully defended the thesis entitled **"MISSION ANALYSES OF A DOUBLE UNIT CUBESAT BEEAGLESAT"**, which he prepared after fulfilling the requirements specified in the associated legislations, before the jury whose signatures are below.

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		....

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To my respected future colleagues

FOREWORD

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Finally, the personal note for myself: *"Non est ad astra mollis e terris via!"*

January 2015 Çağrı KILIÇ

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ABBREVIATIONS

ACS	: Alcantra Cyclone Space
ADCS	: Attitude determination and control system
Bps	: Bits per second
CD	: Coefficient of drag
COTS	: Commercial off the Shelf
CSS	: Coarse sun sensor
EGM	: Earth Gravitational Model
EKF	: Extended Kalman Filter
EPFL	: Ecole Polytechnique Federale de Lausanne
EPS	: Electrical Power Subsystem
ESL	: Electronic Systems Laboratory
FIPEX	: Flux Probe Experiment
FSK	: Frequency Shift Keying
GPS	: Global Positioning System
GTO	: Geostationary Transfer Orbit
HPOP	: High Precision Orbit Propagator
I2C	: Inter Integrated Circuit
INMS	: Ion Neutral Mass Spectrometer
IOD	: In Orbit Demonstration
Kbit	: Kilobit
LEO	: Low Earth Orbit
LEOP	: Launch and Early Orbit Phase
LTAN	: Local Time of Ascending Node
Mbps	: Megabit per second
m-NLP	: Multi Needle Langmuir Probe
MSISE	: Mass spectrometer and incoherent scatter radar exosphere
NRL	: Naval Research Laboratory
OBC	: On-Board Computer
PCB	: Printed Circuit Board
RAAN	: Right Ascension of the Ascending Node
RFIC	: Radio Frequency Integrated Circuit
SD	: Secure Digital
SSDTL	: Space System Design and Test Laboratory
SSO	: Sun Synchronous Orbit
STK	: Systems Tool Kit
TRL	: Technology Readiness Level
TurAFA	: Turkish Air Force Academy
UHF	: Ultra High Frequency
UTC	: Universal Time Coordinated
VHF	: Very High Frequency
VKI	: The von Karman Institute
WGS	: World Geodetic System

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MISSION ANALYSES OF A DOUBLE UNIT CUBESAT BEEAGLESAT

SUMMARY

Nowadays, humankind have the technology of landing to a comet and develop the capabilities needed to organize a manned mission to Mars. Space missions in the past were focused on traditional larger high-cost satellites, but are now transitioning to smaller satellites. These satellites include nano-satellites that are becoming one of the most exciting, diverse and fast paced satellites of today. CubeSat is one of a type of nano satellites. 1U Cubesat is traditionally 10x10x10 cm cubic satellite that weighs no more than 1.33 kg. They are currently used in many countries especially in educational institutions. The ability to access space is a vital strategic capability for all countries but there is not an easy way to do it with large-scale satellites. The high cost of reaching orbit is the major factor preventing the exploration and utilization of space. CubeSat technology is an affordable way to access to space. The QB50 project is an initiative of the von Karman Institute to operate a network of 50 CubeSats to conduct in-situ, multi-point and long duration measurements in the lower thermosphere between 90 and 380 km.

In this study, one of the participants of QB50 project, double unit CubeSat of Istanbul Technical University and Turkish Air Force Academy, namely BeEagleSat was investigated, focusing on orbital mission analysis. Starting with the known values and restrictions of the project, simulations have been carried out in order to obtain the orbital analysis of the satellite. Additionally, perturbations in the orbit, comparisons with other models and simulations with different parameters were studied. There are also some uncertainties for orbit perturbations such as atmospheric drag, Earth gravity and solar radiation pressure were briefly clarified.

Before launch the first thing to do would be the orbital analyses of the satellite. In this way, the STK (Systems Tool Kit) software is used to perform these analyses. Deployment altitude, velocity and direction effects were explored with taking ballistic coefficient, atmospheric drag, and gravity effects into account. By using STK Lifetime Tool, lifetime for the CubeSat was calculated. Solar activity effects are also considered and explained briefly.

Definition of CubeSat, QB50 project and mission characteristics of BeEagleSat is the introduction part of this work. Afterwards, the architecture of BeEagleSat with the payloads and individual subsystems are briefly explained. Then, mission phases of the CubeSat which are mostly related to the ADCS subsystem are shown. In order to give an information of surveyed analyses, orbital elements and orbital perturbations are shown with necessary calculations. Finally, simulations are done by predicting lifetime, calculating sunlight and coverage ability of BeEagleSat. This thesis proposes knowledge of the orbital limits and forecasted results of BeEagleSat.

BEEAGLESAT KÜP UYDUSU GÖREV ANALİZLERİ

ÖZET

İlk insan yapımı uydu olan Sputnik uydusunun fırlatılması uzay araştırmaları alanında bir kilometre taşı olmuştur. 1950'li yılların sonlarında ulasılan bu teknolojik üstünlük ile birlikte ileri bilim ve teknoloji konularında muazzam çalışmalar elde edilmiştir. Sputnik fırlatıldıktan sonra çok sayıda insanlı ve insansız uzay aracı Dünya'dan fırlatılmış ve fırlatılmaya da devam edilmektir. Bu fırlatmalar insanlığa teknoloji, bilim ve eğitim alanında tahmin edilebileceğinden fazla şey katmıştır. Günümüzde uzay teknolojisi ile bir kuyruklu yıldıza iniş yapılabilmekte, gezegenler arası yörüngeler kullanılarak Mars'a insansız iniş gerçekleştirilebilmekte, insanlı inişler için yeni uzay araçları denenmektedir. Uzay görevleri önceleri büyük ve pahalı uydular ile yapılırken günümüzde bu boyut küçültülmüş ve boyutun küçültülmesi ile bu uydulara harcanan miktarlar da azalmıştır. Bu yeni küçük uydular kütlelerine göre çeşitli isimler almaktadır. Nano boyutta uydular da bunlara bir örnektir. Dünya'da bu olgu 2000'li yıllarda oluşmaya başlamış ve ilk denemeler de aynı senelerde gerçekleşmiştir. Günümüzde genellikle Uluslararası Uzay İstasyonu'ndan yörüngeye bırakılan küçük uydular olduğu gibi birden fazla küçük uydu bir fırlatma aracı ile istenilen yörüngeye bırakılabilmektedir. Küp uydular nano uydu ailesinin bir üyesi olarak ortaya şıkmış ve bu aile içerisinde oldukça popüler olan bir uydu türüdür. Genel mantığı ile eğitimsel, teknolojik gösterim ve teknoloji hazırlık seviyesini artırma amaçlı alt sistemlerin ve yüksek teknoloji ürünü görev yüklerinin denendiği küp uydular düşük maliyetlilerdir. Ülkemiz küp uydu yapımı ve fırlatılması durumuna entegre olmuş ve İstanbul Teknik Üniversitesi ile birlikte 2014 yılı itibarı ile tasarlamış ve fırlatılmış 2 küp uydusu ile küp uydu teknolojisine ülkemizde öncülük etmektedir. Avrupa 7.Çerçeve programı kapsamında QB50 isimli uluslararası küp uydu ağı projesine İstanbul Teknik Üniversitesi, Hava Harp Okulu ile birlikte geliştirmekte oldukları BeEagleSat küp uydusu ile katılacaklardır. Toplamda 40'a yakın iki birimlik ve 10 adet üç birimlik küp uydular ile projenin gerçekleştirilmesi planlanmaktadır. Projenin amacı alçak yörüngede atmosferik ölçümlerin yerinde yapılması ve yapılan ölçümlerin tüm dünya genelindeki yer istasyonları tarafından toplanmasıdır. Genel olarak QB50 projesi 50 küp uydunun aynı anda ve birlikte 380 km'lik, dairesel, 98 derece eğikliğe sahip bir yörüngeye bırakılarak eş zamanlı termosferin alt tabakalarında ölçümlerin yapılmasını amaçlamaktadır. BeEagleSat küp uydusu içerisinde Langmuir cihazı ve Sabancı Üniversitesi ve İstanbul Teknik Üniversitesi iş birliğinde geliştirilen X-Ray algılayıcı görev yükleri ile projede yer alacaktır. Küp uydu içerisinde bu görev yükleri ile birlikte ülkenin uydu projeleri ve alt sistem tasarım kabiliyetinin geliştirilmesi açısından kabiliyet ve tecrübe kazanması amaçlanarak bazı alt sistem parçaları ülke içerisinde tasarlanıp geliştirilecektir. Uydunun bazı alt sistemleri ise daha önce uzayda denenmiş, uydunun güvenilirliğini artıracak olan ticari ürünler ile tamamlanacaktır. Uydu, yönelim belirleme ve kontrol sistemi, uçuş bilgisayarı, haberleşme alt sistemi,

elektrik güç, yapı ve ısıl kontrol alt sistemlerini barındırmaktadır. Uydunun fırlatılması olayı ele alınırken bir çok belirsizlik de beraberinde gelmektedir. Bu belirsizliklerin nedenleri birbirine bağlı birçok parametre ile anlaşılabilir, fakat modellenmesi ve ön kestirim yapılması hesaplama yükü bakımından ağırdır. Yapılan varsayımlar ile belirsizliklerin bazıları kısmen azaltılarak ya da yokedilerek çeşitli analizler gerçekleştirilebilir ve uyduların yörüngedeki durumları gözlemlenebilir. Yapılan analizler ile önerilen belirsizliklere bağlı olarak sonuçlar çeşitlilik gösterebilmektedir. Temel amaç mertebe türünden uygun ve güvenilir sonuçların elde edilmesidir. Bu görevleri yerine getirebilmek için BeEagleSat uydusunun özellikleri verilmiş ve QB50 projesi dahilinde önerilen başlangıç koşulları ile birlikte yörünge elemanları açıklanmıştır. Literatür çalışmaları ve önerilen başlangıç koşulları ile yapılan tahminler ile edilen parametreler sayesinde BeEagleSat uydusu için önemli ve kritik görülen yörünge analizleri yapılarak uydunun yörüngesel kısıtları belirlenmiştir. Bu analizler uydunun ömrü, gerçekleştirebileceği haberleşme süreleri, kütle ve alanının yörünge ömrüne etkileri, ne kadar süre Güneş göreceği ve buna benzer durumlar için yapılmıştır. Bu analizlerin sonuçlarını doğrudan etkileyen faktörler ise irtifa etkileri, fırlatma zamanı etkileri, fırlatma yönleri ve uydunun kontrol edilebilmesidir. İrtifa etkileri ve fırlatma zamanı etkileri küp uyduların ömürlerini incelemek için yapılmıştır. Balistik katsayı, atmosferik sürüklenme ve yerçekimsel etkiler ışığında ele alınarak incelenmiş ve analizleri gerçekleştirilmiştir. Küp uyduların tanımı, etkileri ve günümüzdeki yeri ile birlikte QB50 projesinin ve BeEagleSat uydusunun görev tanımı bu çalışmanın giriş kısmını oluşturmaktadır. Ardından BeEagleSat uydusunun mimarisi tanımlanmıştır. Bu mimari içerisinde görev yükler olan Langmuir cihazı ve X-Ray algılayıcısı anlatılmıştır. Alt sistemler ise uydu yapısı, yönelim belirleme ve kontrol sistemi, elektrik güç sistemi, uçuş bilgisayarı ve veri kotarma alt sistemleri ile açıklanmıştır. Uydular Cyclone-4 fırlatma aracı yörüngeye yerleştirilecektir. Fırlatma aracı içerisinde bulunan fırlatma mekanizması ile roketin üçüncü kademesinden ayrılması ile birlikte başlayan görev fazları ise fırlatma ve ön yörünge analizleri adı altında açıklanmıştır. Uydunun fırlatma mekanizmasından çıkması ile oluşacak açısal hız bozuntuları ile birlikte uydu, önceden ön görülemeyen bir şekilde kendi etrafında dönmeye ve takla atmaya başlayacaktır. Bu fazı açıklayan takladan kurtulma durumu, 1960'lı yıllarda ortaya koyduğu öngörülebilen takla modeli ile bilinen Thompson'ın kendi adını taşıyan Y-Thompson arafazının detaylı açıklanması ile görev fazlarına devam edilmiştir. Uydunun öngörülen takla modunun ardından geçerli ön şartları sağlaması ve yörünge hız vektörüne paralel bir şekilde devam etmesini öngören Y-momentum fazı akabinde detaylandırılmaya çalışılmış ve en son olarak uydunun operasyonel fazında neler yapması gerektiğine değinilmiştir. Bir sonraki yörünge dinamiği-mekaniği bölümünde okuyucuya yapılan analizlerin daha iyi anlaşılabilmesi için yörünge elemanları bilgisi ve görselleri, yörünge bozuntuları ve hesaplamalarının yanı sıra bu bozuntuların bazıları hesaplamalı olarak verilmiştir. Ön bilgilerin devamında ise yapılan modelleme ve analiz sonuçlarından bahsedilmiştir. Uydunun yörünge ömrü boyunca ne kadar gün ışığından yararlanabileceği, yörünge ömrü tahmin hesaplamaları ve uydunun İstanbul Teknik Üniversitesi yer istasyonu ile ne kadar süre haberleşebileceği tartışılmış ve detaylıca açıklanmıştır. Göreceli olarak daha karmaşık ve diğer analizlere göre hesap yükü ve süresi daha fazla olacak şekilde BeEagleSat uydusunun fırlatma anından, atmosfere giriş yaparak yanma aşamasına kadar kullanılan programın izin verdiği sınırlar dahilinde olabildiğince gerçeğe yakın hesaplamalar içeren bir görev analizi de gerçekleştirilmiş ve sonuçları sunulmuştur. Son olarak ise elde edilen sonuçlar ve öneriler sunulmuştur. Bu tezde sözü geçen analizler 2012 yılına kadar ismi Satellite Tool Kit olan ancak onuncu versiyonunda Systems Tool Kit olarak değiştirilen STK programı ve MATLAB programıkullanılarak gerçekleştirilmiştir. STK programı içerisinde çeşitli yörünge denklemleri çözücüleri bulunmakta ayrıca yörünge ömür tayini yapabilmektedir. Bunlara ek olarak bu program sayesinde uyduların birbirleri ile çarpışma riskleri de incelenebilmektedir. Daha bir çok özelliği olan bu programın, bir takım araçları sayesinde yapılan bu senaryolar ve bazı çizimler öğrencilere sunduğu eğitim lisansı ile gerçekleştirilmiştir. Bu senaryolar yörünge dinamiği senaryoları adı başlığında tanıtılan çözücüler ve ara programlar ile gerçekleştirilmiştir. Genel olarak hata payı en aza indirgenmiş modüller kullanılmıştır. Bunun yanı sıra kullanılan modüllerde bulunan bilgi dosyaları en güncel formatta kullanılmıştır. Tüm senaryolar için yönler ve 3 boyutlu görüntüler mümkün olduğu kadar verilmiş ve bu sonuçlara bağlı olarak detaylı tablolar, grafikler ve sonuçlar her senaryonun sonunda verilmiştir. Bununla beraber, STK isimli program kullanılarak küp uydular, başlangıç şartları ile oluşturulan çeşitli senaryolar içerisinde analizler gerçekleştirilmiştir. Fırlatma zamanı etkileri de aynı şekilde küp uyduların ömürlerine fırlatma zamanlarının nasıl etkiyeceğini araştırmak için yapılmıştır. Küp uydu ömür tayini için STK isimli programın Lifetime adlı alt programı kullanılmıştır.

1. INTRODUCTION

1.1 Definition of CubeSats

Thanks to advanced investigations in the aerospace area, mankind bear witness to significant events. Especially the launch of Sputnik in 1957 [1] was the dawn of the space explorations and a milestone in the advances in science and technology. Since the Sputnik was launched, plentiful manned and unmanned spacecraft have been launched and continue to be launched for scientific, educational and other technological purposes. Traditional satellite missions are expensive to design, build, launch and operate. Consequently, both the aerospace industry and the research community have started directing their attention to missions involving many, small, distributed and inexpensive satellites. Furthermore, many space projects in university laboratories are focused on the development of small satellites for both scientific and educational purposes. This trend is the result of the potential cost reduction and improvements in technology, which allows the reduction of size. New approaches emerge as small satellite field demands itself as a special field. Therefore, constellation, cluster, and swarm concepts became popular because of their capacity of protracted mission.

This field includes CubeSats, becoming one of the most interesting, diverse and fast paced satellites of today. The concept of CubeSats was publicly proposed in 2000 [2] and the first six CubeSats launched in 2003 [3]. A CubeSat is a nano satellite, 10 cm cube, has a mass of no more than 1.33 kg which includes the most standard functions of a big scale satellite [4]. Also, they may be produced with deployable solar panels, antennas or booms. Orbit control can be used by micro propulsion; however, it is generally limited. It takes about six month to two years to develop a CubeSat from scratch to launch. The cost of a CubeSat design is normally in the range $75-125k but the cost depends on the complexity of payloads [5]. Current technology is growing; however, it is the fact that single CubeSat may not have enough in size. It has simply circumscribed power budget to support sensors for critical scientific research.

Certainly, every scale of satellite has its own benefits and limitations.
CubeSats are small, cost-effective and educationally satisfying satellites; however, single CubeSat is small to carry sensors for important scientific investigation. On the other hand, it is possible to make tremendous and vast deductive space operations by using many CubeSats in orbit. Considering all these points, it would not be a drastic change from larger satellites to smaller satellites in the near future, but trends and popularity of miniaturized satellites gives a point of view that growing potential of the CubeSat technology is not a doubtful or unsubstantiated reality.

1.2 QB50 Project

QB50 is a network of 50 CubeSats (string-of-pearls configuration) built by mostly universities and some companies all over the world. The CubeSats will be launched in the early January of 2016 by a single launch vehicle into circular orbit at 380 km. The goal of the QB50 mission is to achieve a maintained and economical access to space and measure scientific data in the largely unexplored lower thermosphere region [6]. The project claims that this important research can be accessible in a small satellite network program. The 50 CubeSats include 10 triple unit (3U) CubeSats for technology demonstration and 10 atmospheric 2U CubeSats. The QB50 project is an initiative of The von Karman Institute (VKI) to coordinate universities, institutes and companies from all around the world, to design, build and operate a network of CubeSats [7].
At the altitude of deployment, atmospheric drag is the most influential source of orbital decay. Due to atmospheric drag, orbits of the CubeSats will decay and lower layers of thermosphere region from 380 km to 200 km altitude will be explored without the need of controlling the decay by using a propulsion system. The other benefit of QB50 mission may be given as studying re-entry process of a CubeSat by measuring CubeSat on-board temperature and deceleration parameters [6]. The initial total network size in orbit may be determined by the deployment sequence, initial deployment velocity and initial direction of deployment with respect to deployment mechanism. Orbital dynamics modelling has shown that due to density variations along the orbit and small distinctions in the CubeSat ballistic coefficients, the separation range will be altered

significantly, eventually leading to a non-uniform distribution of CubeSats all the way around the Earth [8]. An example illustration is given in Figure 1.1.

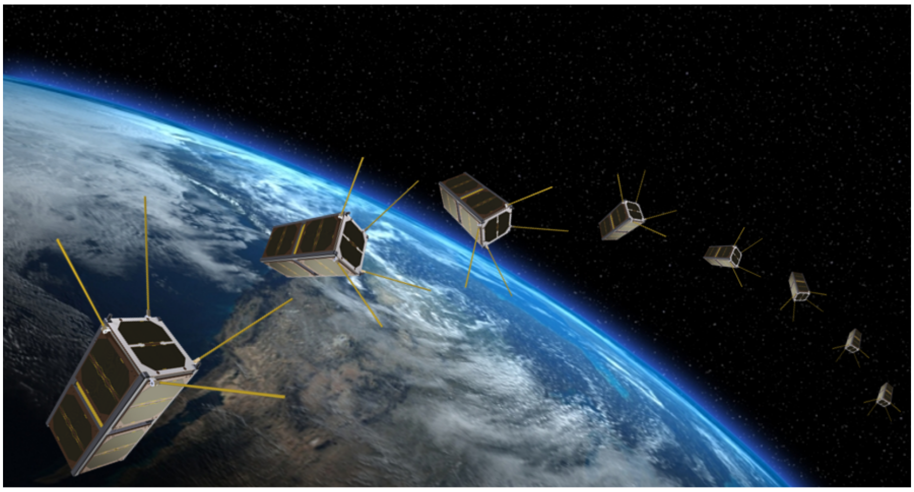

Figure 1.1: Illustration of CubeSat string of pearl configuration in orbit.

As mentioned before, a single CubeSat is too small to carry sensors for significant scientific research. Even though the main aim of developing and operating a CubeSat is educational for universities, combining a large number of CubeSats by using identical sensors into a network raise the benefits of educational purpose for answering fundamental scientific questions by using affordable sources. According to the QB50 mission, single or a few CubeSat reliability is not a major concern because the network can still achieve its mission objectives even if a few CubeSats fail [6]; however, it should be fully investigated how to perform a faultless mission.

1.3 Mission Definition

The mission definition of the QB50 project is to carry out atmospheric research within the lower thermosphere. Investigation of this region was established by satellites which using elliptical orbits; however, due to lack of time spent in the region would give insufficient data. Sounding rocket flights may be another way to perform in situ measurements. Nonetheless, the time spent of these rockets in lower thermosphere is relatively short and they cannot offer measurements with a completely coverage of this region. In addition, using a single satellite provides single point measurement [9]. On the other hand, using multiple satellites would offer multi-point and in-situ measurements and putting them into a circular orbit provides the opportunity to perform atmospheric measurement in higher time resolution. The multi-point, in-situ

measurements of QB50 will be beneficial and close the gaps of unanswered questions about the lower thermosphere [10]. There are three different types of science units in QB50 mission. These science sensors include the Ion-Neutral Mass Spectrometer (INMS) as part of set 1, the Flux- Φ -Probe Experiment (FIPEX) as part of set 2, multi-Needle Langmuir Probe (m-NLP) as part of set 3. In addition, QB50 mission includes in orbit demonstration CubeSats which will serve as technology demonstration satellites [6].
The QB50 satellites will be launched by the Cyclone-4 launch vehicle. The launch service is provided by Alcantara Cyclone Space (ACS). The launch parameters of the orbit are nearly 98 degrees inclination and a 380 km altitude and proposed launch date is January 2016 with a 8 am-to-2 pm LTAN [6]. It is known that CubeSat missions are inherently risky. This is due to the use of low cost components and the educational nature of such projects. The instruments, which will be used for QB50 satellites, must be tested in orbit even though it had been tested on ground. Hence, in order to reduce the level of risk and ensure that the new instruments will operate correctly, the QB50 management team launched QB50p1 and QB50p2 in June 2014 as precursor satellites, which are currently in orbit [11].
QB50p1 carries an INMS payload and QB50p2 carries a FIPEX payload. Both satellites carry thermocouple instruments provided by the VKI and amateur radio payloads provided by AMSAT Netherlands and AMSAT Francophone. The satellites are tested for their attitude determination and control subsystems (ADCS) which is designed for QB50 mission. This precursor mission has made it possible to test the developed payloads, ADCS performance in orbit. Also, the operational capability of QB50 satellites were tested [12]. Although the precursor flight is a milestone of the project, the orbit of the precursor satellites is different from the QB50 mission. The main aim of this flight is to test payloads and subsystems. The precursor QB50 CubeSats were launched by a Russian Dnepr rocket into a near Sun Synchronous Orbit (SSO) at 624 km altitude [13].

Up to now, the proposed QB50 CubeSats are listed in Table 1.1 with the payload determined. There are other QB50 CubeSats but their payload proposal has not yet been officially announced. The full list without their payloads may be found at [6].

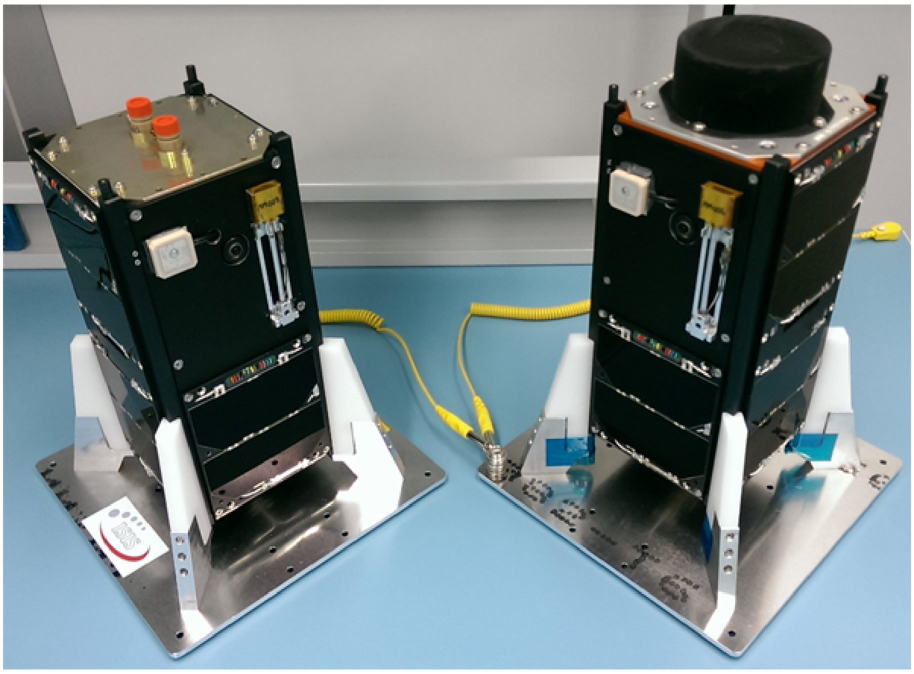

Figure 1.2: Precursor CubeSats of QB50 mission [11].

Table 1.1: Proposed QB50 CubeSats with known payloads [6].

INMS	FIPEX	m-NLP	IOD
UNSW-EC0	YUsend-QB50	RIOSAT	QARMAN
BUSAT 1	VZLUsat 1	ExAlta 1	EntrySat
LilacSat 1	SOMP 2	BeEagleSat	DELFFI 1
QBITO	SNUSAT 1	HAVELSAT	DELFFI 2
ANUSAT 1	NUSQB50	-	InflateSail
LINK 1	-	-	-
PHOENIX	-	-	-

2. ARCHITECTURE OF BEEAGLESAT

BeEagleSat is a 2U CubeSat which is designed by Istanbul Technical University (ITU) and Turkish Air Force Academy (TurAFA) for the QB50 project. Sabanci University (SU), HAVELSAN Co. and some small sized enterprises are also contributing to the project by providing a novel X-Ray detector system and support on the satellite software and subsystems. BeEagleSat is using the Sensor Set 3 consisting of Multi Needle Langmuir Probe, corner cube laser retro-reflectors, thermistors and thermocouples, to be provided by the QB50 management. The BeEagleSat will use commercial off-the-shelf (COTS) systems such as Surrey Space Centre's ADCS unit with magnetorquers and a momentum wheel, electrical power system of Clyde Space, as well as in house developed systems such as on-board computer, modem and the secondary payload, the X-Ray detector. All these subsystems are being developed to meet mass, volume, link and pointing requirements of the QB50, within allowed margins. Ground station software developed by Swiss Federal Institute of Technology Lausanne (EPFL) is also adopted [14]. System diagram employing the subsystem contents of the BeEagleSat is shown in Figure 2.1. Also, a CAD of the BeEagleSat is given in Figure 2.2.

Instruments and spacecraft sub-systems are classified according to a Technology Readiness Level (TRL) on a scale of 1 to 9. Levels 1 to 4 relate to creative, innovative technologies before or during mission assessment phase. Levels 5 to 9 relate to existing technologies and to missions in definition phase [15]. The TRL values for the most subsystems are level of 9, as pointed out in Table 2.1. By this means of using TRL-9 subsystems or parts, the risk of having the CubeSat ready in time is minimized. In case of a problem with Gumush Aerospace structure, the 2U structure by Innovative Solutions in Space will be employed. The Pumpkin OBC and the Astrodev Helium-100 are the backup plans for the in-house developed OBC and communication systems, respectively.

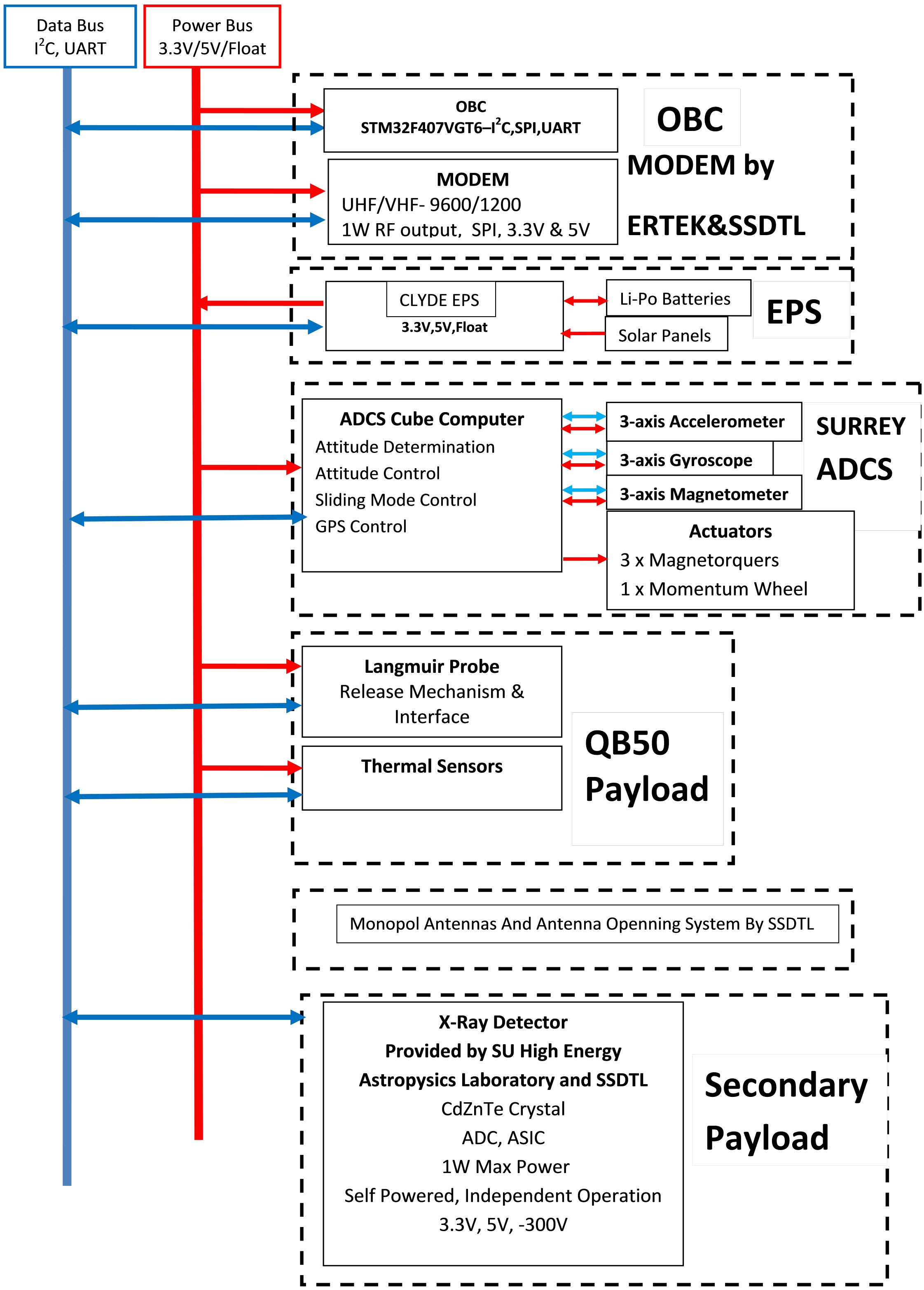

Figure 2.1: BeEagleSat system diagram [14].

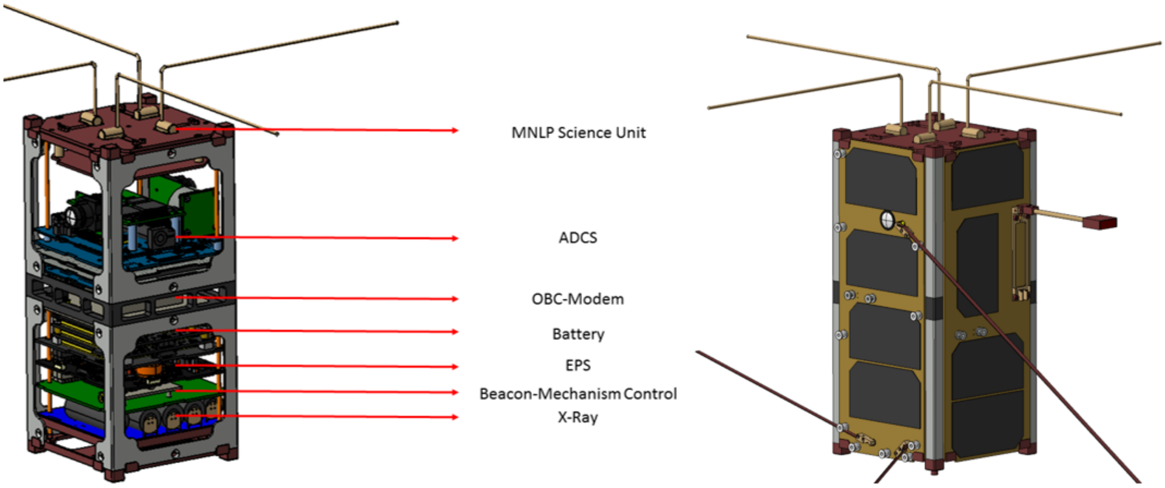

Figure 2.2: BeEagleSat CAD model [14].

Table 2.1: TRL level of BeEagleSat subsystems [14].

Subsystem	Developer	TRL
Structure	Gumush Space	5
Solar Panels	Clyde Space	9
X-Ray Detector	SU-SSDTL	5
EPS	Clyde Space	9
Battery	Clyde Space	9
ADCS	SSC	9
Magnetometer	Honeywell	9
Antenna Switch	SSTDL	9
OBC-Modem	Ertek-SSDTL	5
Memory Card	Toshiba	9
m-NLP	QB50 Science Unit	8

Table 2.2: Mass budget analysis of BeEagleSat [14].

Subsystem	Mass (gr)	Proposed Mass (gr)
Structure	271	271
Solar Panels	345	326
X-Ray Detector	297	297
EPS+Battery	310	300
ADCS	400	400
OBC-Modem	240	220
m-NLP	300	225
Others	220	220
Total	2383	2259

The current total mass of system with non-tailored design of subsystems is 2383 gr. The planned total mass is 2259 gr which is shown in Table 2.2.

2.1 Payload

BeEagleSat carries two payloads. The primary payload is Multi Needle Langmuir Probe (m-NLP) which is provided by QB50 Management Team. The secondary payload is X-Ray Detector which is developed by Istanbul Technical University Space Systems Design and Test Laboratory (SSDTL) and Sabanci University.

2.1.1 Multi needle langmuir probe

The primary payload of BeEagleSat is Multi Needle Langmuir probe (m-NLP) which is developed at the University of Oslo. The system makes use of four needle probes biased with a different positive potential well above the platform potential, and a data acquisition unit. The m-NLP system gives high time resolution measurements of absolute electron density and satellite floating potential. The instrument works by measuring the current collected individually from four needle probes, placed in front of the rocket/satellite's ram surface [16]. The collected current is converted to voltage, filtered, digitized and then sent to the central telemetry system for down link to the ground station. By using data from four fixed-bias Langmuir needle probes, sampled at the same time, the plasma electron density can be derived with high time resolution without the need to know the electron temperature and the spacecraft potential. Corresponding CAD model of m-NLP can be seen in Figure 2.3 which was taken from [9]. According to the QB50 Science Sensor providers m-NLP sensor has

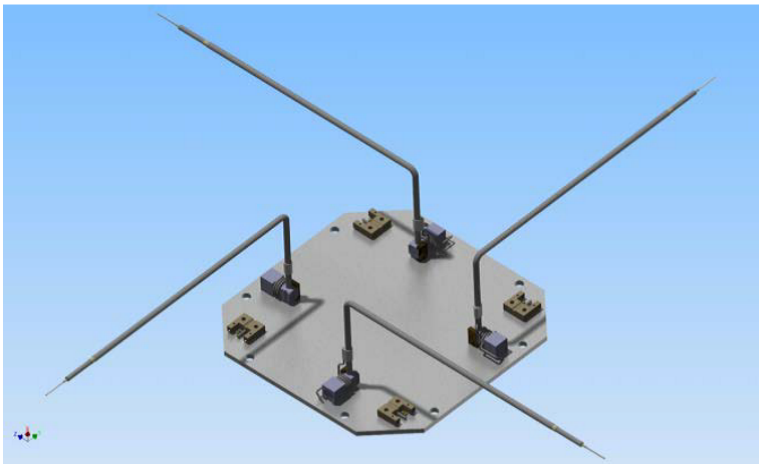

Figure 2.3: m-NLP payload of BeEagleSat [9].

been planned to use in a CubeSat called CubeStar, whose launch scheduled for late 2014 but is still under development [9].

2.1.2 X-Ray detector

The BeEagleSat will carry an X-ray detector system as a secondary payload. The X-ray detector system includes orthogonal strip CdZnTe crystal. A CdZnTe based semiconductor X-ray detector and its combined readout electronics is developed by the project team. The detector system on the BeEagleSat has orthogonal strips for position resolution. Such orthogonal strip detectors have been proposed as detector material for X-ray observatories before, but they have never been realized in space. For small satellite systems, having low number of readout channels have assets regarding to power and using in space. This overcomes issues with the minimum energy and the weakened resolution of energy which fits the standard of CubeSats. The main purpose of the detector is technology demonstration. The secondary purpose is to measure the 20-200 keV electromagnetic resolution background in Low Earth Orbit [17]. The detector system that will be placed on the BeEagleSAT has orthogonal strips for the position resolution on the detector.

2.2 Subsystems

Due to the size restriction of CubeSats, it is hard to design and optimize the inner hardware configuration. However, advances in technology helps to configure the hardware of CubeSats. They have subsystems like large scale satellites but they are relatively small. BeEagleSat contains one of the most advanced CubeSat subsystems. Subsystems of BeEagleSat are given in the following subsections with their specifications.

2.2.1 Structure

BeEagleSat 2U structure is developed by a micro scale enterprise Gumush Aerospace and Defense Ltd., which has been established by ITU students during a CubeSat project at ITU. Gumush Aerospace and Defense Ltd. designed and developed a CubeSat structure for BeEagleSat. This structure is a modular nano satellite structure and it is based on the CubeSat standards. The structure is made up of 1U sized cube blocks that can be integrated to each other. One unit obtains a metal frame and two caps components for top and bottom fields of the main frame [18]. In case of a problem with the proposed structure system, Innovative Solutions in Space (ISIS) double unit structure system will be used which has been successfully used several times [14].

2.2.2 Attitude determination and control subsystem

QB50 mission requires attitude control to minimize the influences of drag. Attitude control is also necessary to make it certain that the payloads are in the ram surface of the satellite. As the attitude determination and control system (ADCS), the QB50 ADCS by Surrey Space Center of University of Surrey and Electronic Systems Laboratory (ESL) of Stellenbosch University will be used. QB50 ADCS estimates the attitude with a combination of an external magnetometer, sun and nadir sensor measurements and a micro-electro-mechanical systems (MEMS) rate sensor. Two magnetorquers and a momentum wheel are used to stabilize and control the attitude of the satellite. Highly detailed information on the hardware components of the system and the attitude determination and control modes are available in [19]. Additionally, the specification of the certain elements of the ADCS unit are given in Table 2.3.

The body axes system of the satellite has its origin at the center of mass of the satellite. In order to be in line with other QB50 teams, the axes directions are selected as +X-axis is directed along the CubeSat axis and in the ram surface surface of the satellite, +Z-axis is in the nadir direction, and +Y-axis is perpendicular to the orbital plane and

Table 2.3: Specifications of ADCS hardware [19].

Specification	Value
Physical Dimensions	
CSS	3 x 10 x 3 mm^3
External Magnetometer	15 16 x 6 mm^3
GPS Patch Antenna	10 x 10 x 6 mm^3
Performance	
Attitude Update Rate	1 Hz
Attitude Measurement Accuracy	>200 km
Pitch	$<0.50°(1\sigma)$
Roll and Yaw	$<2.00°(1\sigma)$
Sun and Nadir Sensor	
Sun Sensor Measurement Accuracy	<2.00°
Nadir Sensor Measurement Accuracy	$<0.18°(1\sigma)$
Field of View	$180°(1\sigma)$
Coarse Sun Sensor	
Measurement Accuracy	$<10.0°(1\sigma)$
Magnetometer Measurement Accuracy	<50nT (1σ per X/Y/Z channel)
Rate Sensor Measurement Accuracy	<0.015°/s (1σ)

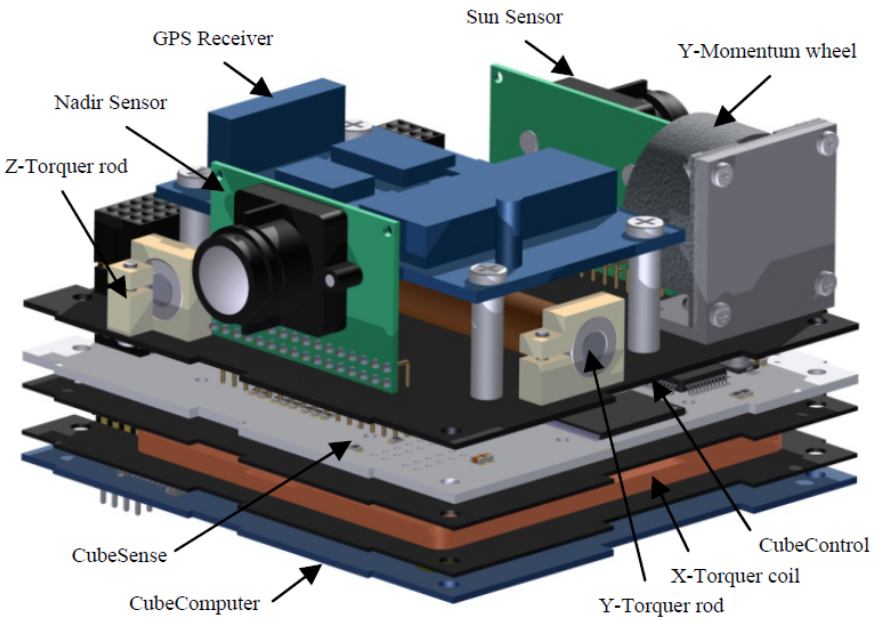

Figure 2.4: ADCS CAD model [23].

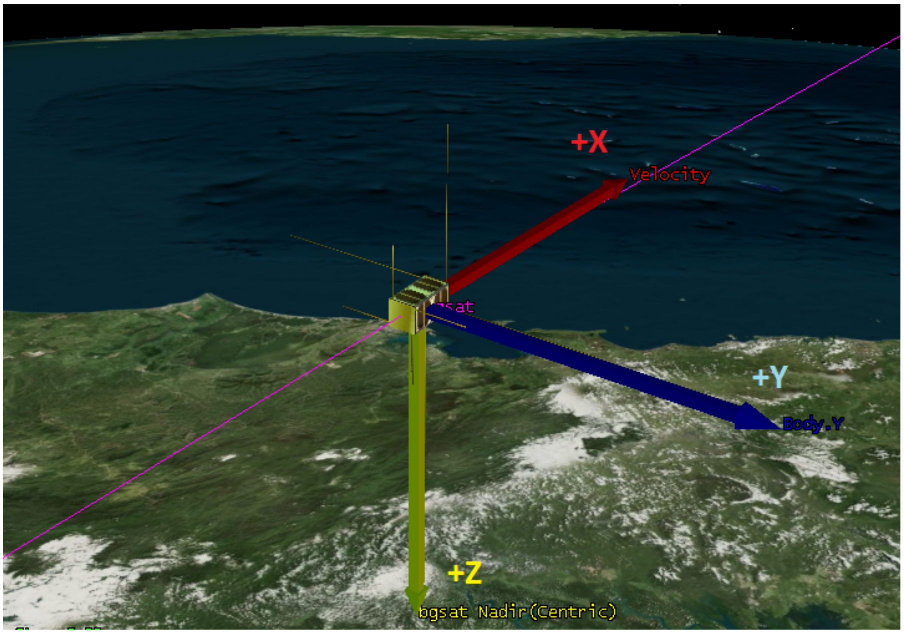

Figure 2.5: Axes and directions of QB50 CubeSats.

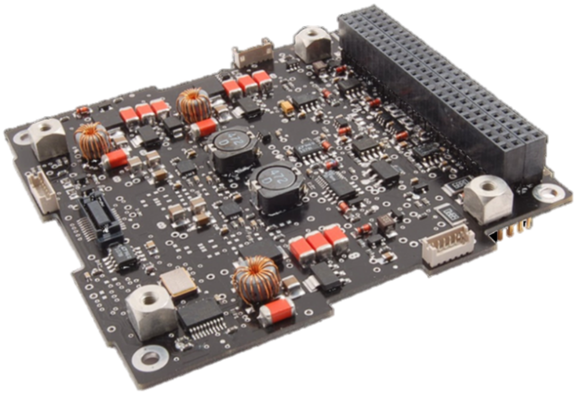

Figure 2.6: Clyde Space - electrical power subsystem [20].

directed in the opposite direction with the orbital angular velocity vector [19]. The body axes system can be seen in Figure 2.5.

2.2.3 Electrical power subsystem

BeEagleSat uses Clyde Space Electrical Power System (EPS) to regulate the voltage levels of all subsystems from 3.3 V to 5 V. The EPS has battery charge management for lithium ion and lithium polymer batteries, diode and separation switch, and I2C interface to telemetry and telecommand [20]. Besides, this system has 3.3 V, 5 V and raw battery buses with over-current protection and battery under voltage protection. The flight model of EPS can be seen in Figure 2.6. Clyde Space Lithium Polymer battery is used as an energy storage unit. The battery can be seen in Figure 2.7.

Figure 2.7: Clyde Space - battery [20].

The specifications of the battery [20] are itemized as follows:

- High energy density 150 Wh/kg
- Maximum 30 Wh energy
- 8.2 V terminal voltage at end of charge
- Suitable for hard vacuum conditions
- Integrated battery heater with thermostat to maintain battery temperature above 0°C
- Battery over-current protection
- Battery temperature, voltage and current telemetry via integrated digital interface
- Suitable for engineering model and flight model

The main power source of the satellite is produced by solar panels. BeEagleSat uses a tailored design of double unit solar panels from Clyde Space. The solar panels are located on the sides of the structure except for top and bottom sides because scientific unit (m-NLP) and secondary payload (X-Ray) are placed on these sides, respectively.

2.2.4 On board computer and data handling subsystem

Main objectives of On-Board Computer are to operate the various subsystems and to gather data required for the mission. On-board Computer of BeEagleSat is being designed and developed by ERTEK Space System CO. in collaboration with Istanbul Technical University, Space System Design and Test Laboratory (SSDTL). The OBC and communications subsystems share the same PCB to reduce required space and mass. Micro-controller unit consists of an ARM Cortex M4 as the microprocessor and its peripheral components. Storage unit includes two SD Card for redundancy. The communication subsystem includes Si4463 RFIC transceiver. The uplink frequency is in VHF band and the downlink frequency is in UHF band. As the modulation type, FSK modulation will be used. The beacon communicates with the OBC through the PC104 bus via I2C [21]. Antennas are tied to the solar panels using a fishing line on a resistor which will burn the line by applying a current through the resistor with OBC command. A prototype of OBC-Modem can be seen in Figure 2.8.

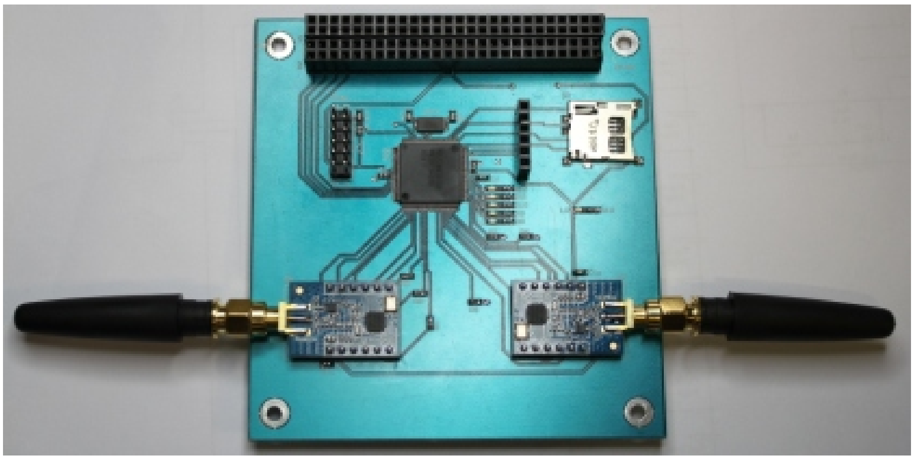

Figure 2.8: Prototype of OBC-Modem [21].

3. MISSION PHASES OF BEEAGLESAT

BeEagleSat mission modes include launch and early orbit phase, de-tumbling phase, Y-Momentum phase and operational phase. After deployment, the power of CubeSat will be switched on. After that, antenna opening mechanism will be turned on and beacon will start to transmit the identification data of BeEagleSat. Then, magnetometer boom will be deployed. After the deployment of magnetometer boom, ADCS on-board computer, CubeComputer, will move the next step which is de-tumbling mode. ADCS requirements for the mission are to keep a pointing accuracy within $\pm 10^\circ$ of the ram direction and have a pointing knowledge accuracy within $\pm 2^\circ$. ADCS always starts with an idle conditions. Orbit, attitude estimation and control modes can be changed via telecommand. The corresponding attitude estimation and control sequence systematic scheme can be seen in Figure 3.1. Planned time period of the

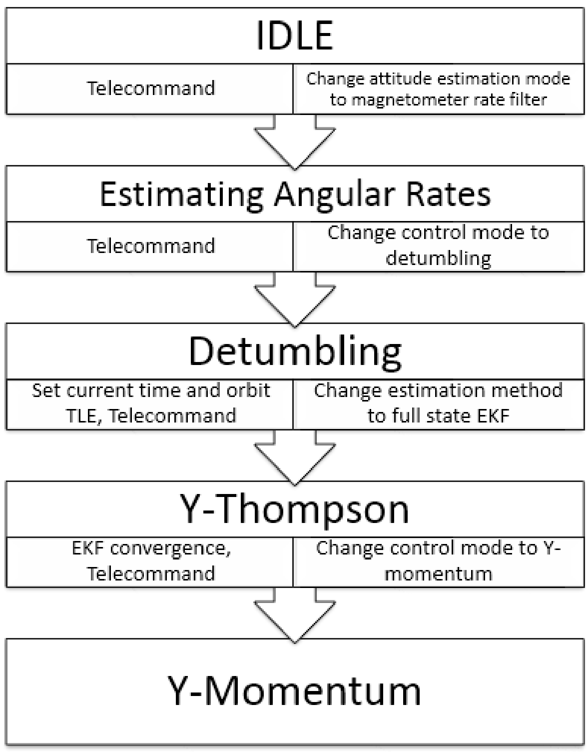

Figure 3.1: Mission modes of attitude determination and control system [19].

stabilization of BeEagleSat is maximum of 2 days [19]. GPS and ADCS data will be received by ground stations to monitor BeEagleSat stabilization status. This will make it available to estimate the BeEagleSat orbital state vectors with respect to time. Y-Momentum mode has specific conditions to achieve. Transition decisions will be made by monitoring the received states in telemetry. The CubeSat's orbital and body axes rate of change of angle will play the main role to decide transitions. If the necessary state telemetry are received after Y-Momentum mode, the CubeSat will start operational (normal) mode. The following stabilization will be enabled to release m-NLP probes which are estimated as 1 week after deployment. The data will be written onto the SD Card. The CubeSat shall be able to transmit at least 2 Mbit data per day. Finally, the received data will be uploaded to central server.

3.1 Launch and Early Orbit

Launch and early orbit phase (LEOP) includes launch and orbit insertion sub-phases. The power of CubeSat will be switched off during launch. QB50 satellites will be in QuadPack deployment mechanism which will be mounted on Cyclone-4 upper stage [22]. After releasing of BeEagleSat from QuadPack, the power will be switched on. The CubeSats will have different initial positions and velocity vectors due to deployment. Most of them may have different masses and change rate of reference areas. Hence, these CubeSats have different ballistic coefficients and duration of LEOP. This difference will be the main reason of along track separation of each CubeSat. Previous initial simulations by using CubeSats which have different ballistic coefficients showed that the range between them might be hundred of kilometres after 4 days from deployment of the CubeSats [8]. Approximately 30 minutes after deployment, antenna opening mechanism will be on and beacon will start functioning. Using provided two-line elements data, beacon modem data will be received. It is the starting point of beacon transmission and after this point, beacon will transmit the identification data of BeEagleSat at every 30 seconds. After 10 minutes from antenna deployment, the magnetometer boom will be deployed from BeEagleSat. Launch and early orbit phase will take nearly 40 minutes for BeEagleSat after deployment from QuadPack [14].

3.2 De-tumbling Mode

After deployment from QB50 dispenser, BeEagleSat may tumble at a random rate. Approximately, it will have a random body spin of less than 10 °/s in all axes [23]. However, measured tumbling rate of QB50p2 which is a precursor QB50 CubeSat is 30°/s for Y-axis [12]. Sensors of 3-Axis magnetoresistive will be attached on a deployable arm (magnetometer boom) to BeEagleSat. The deployment of the magnetometer boom will occur after ejection from the QB50 dispenser. Then, magnetic Bdot and Y-Thompson [24] controller will be enabled in order to eliminate X-axis and Z-axis angular rates and control Y-axis body rate to a reference balance rate. B-dot rate damping controller needs only a Y-axis magnetic moment. There is not any unequivocal attitude or rate information required for this [19]. After deployment of magnetometer boom from BeEagleSat, on-board computer of ADCS (CubeComputer) will pass to the next step which is the de-tumbling mode. The controller is applied by measuring the magnetic field vector at ordered intervals. Furthermore, the next step will be instantaneously controlling Y-axis rate of the BeEagleSat body bring to a controllable specific reference value using telecommand. This will take BeEagleSat to Y-Thompson mode of stabilization (head-over-heels tumbling). This can be done by using directly with rate sensor which is a part of CubeSense. Determination can be done by this method. In order to provide the control, ADCS unit needs to know current attitude with angles and/or rates by using sensor measurements. This will make it available to estimate the BeEagleSat orbital state vectors with respect to time. Transition decisions will be made by monitoring the received states in telemetry. X, Y and Z axis angular rates, the angle between the CubeSat body Y-axis and orbit Y-axis, and the pitch angle change rate of the CubeSat will play the main role to decide transitions. Y-Thompson spin state can be achieved by ADCS approximately in one day [23].

3.2.1 Y-Thompson mode

Aim of this mode is aligning the BeEagleSat Y-axis with the orbit normal axis. This makes the CubeSat a stable condition which is called as Y-Thompson spin (head-over-heels tumbling). In this mode, CubeSat is supposed to rotate only around

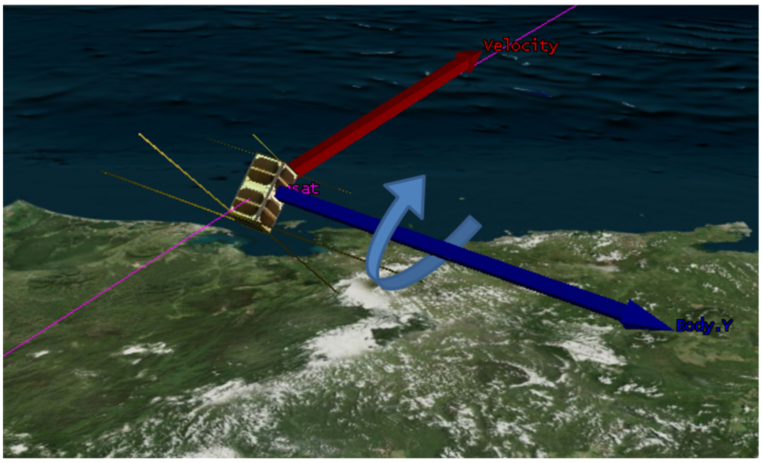

Figure 3.2: Illustration of Y-Thompson spin alignment.

ADCS Y-axis. The first control mode of the ADCS will try to recover the BeEagleSat from the initial unknown tumbling condition to a stable known tumbling motion. In order to meet the stabilization requirements of Y-Thompson spin mode, X and Z angular rates should be smaller than 0.5 °/s and Y axis angular rate should be within 0.5 °/s of the reference spin rate [19]. This reference spin rate can be changed by telecommand. Also, the angle between the satellite body Y-axis and orbit Y-axis should be smaller than 20°. If the CubeSat provides these conditions, it will be in the Y-Thompson mode.

3.3 Y-Momentum Mode

After the QB50 ADCS places the satellite in a stable and known tumbling motion, a Y-axis momentum wheel is used to absorb the body momentum and to stabilize the satellite body Y-axis into a fixed nadir pointing attitude. In another words, BeEagleSat is supposed to stop spinning and stabilize to the nominal orientation which are zero roll, pitch and yaw angles in this mode. This mode can only be activated once the Y-Thompson condition is satisfied. Also, the pitch angle can be controlled to a specific reference value via telecommand in Y-momentum mode. The nominal mode for BeEagleSat can be considered as Y-momentum mode because this mode satisfies the pointing requirements. On the contrary of Y-Thompson spin requirements, Y-momentum mode needs the full attitude knowledge with angles and rates. This can be estimated using an orbit propagator. Updated TLE and current time may be provided via telecommand [19].

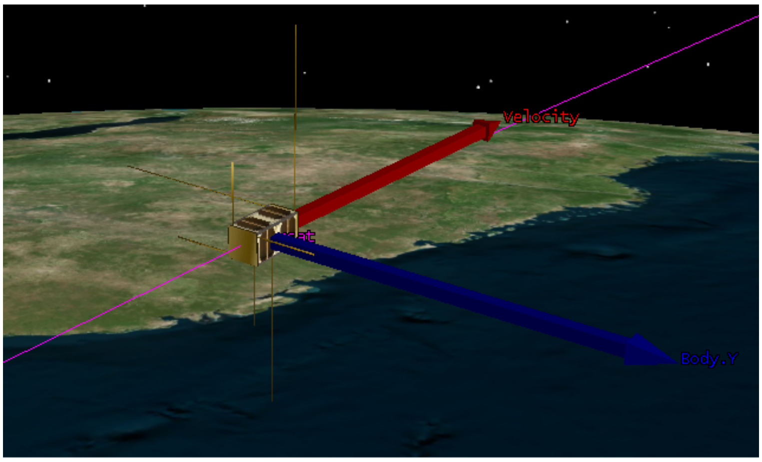

Figure 3.3: Illustration of Y-Momentum alignment.

3.4 Operational Mode

If the necessary state telemetry is received after Y-momentum mode, the CubeSat can start operational mode. m-NLP will be released after BeEagleSat stabilized. The payload releasing time is estimated as 1 week after deployment. The measured data will be written onto SD card. The CubeSat should transmit 2 Mbits data per day to ground stations [19]. After that the received data will be uploaded to central server.

4. ORBITAL MECHANICS

Orbital mechanics describes the motions of objects in space moving under the influence of different celestial objects, e.g. stars, planets and moons. Ascent or descent trajectories, landing, re-entry, lunar and interplanetary trajectories are also included in engineering applications of orbital dynamics [25], [26].

4.1 Orbital Elements

Orbital elements are required to identify a specific orbit. These elements are parameters that describe the size, shape, orientation (three angles) of an orbit and the location of an object in that orbit. All parameters are depending on a specified point in time (epoch), when they are first specified. In combination with epoch, six parameters are enough to specify an unperturbed motion of an object in space. The classical orbital elements are shown in Figure 4.1 and briefly explained below:

Semi-major Axis length, a: One half the distance across the long axis of an ellipse defines semi-major axis. It specifies the size of the orbit and relates to mechanical energy of the orbit. The orbital period and angular momentum of a satellite can be computed from the semi-major axis length.

Eccentricity, e: This parameter specifies the shape of an orbit and determines which kind of conic section it is. This is the ratio between distance from the center of the ellipse to the focus of the ellipse and semi-major axis length.

Inclination, i: The orientation of an orbital plane with respect to the equator of the central body can be characterized by inclination. It describes the orbit ellipse plane's tilt angle with respect to the plane of equator. Inclinations of 0° and 180° are called as equatorial orbit. An inclination of 90° is characterized that the orbit is polar. In addition, when the inclination value is between 0° and 90° the orbit is called as prograde orbit which means that the rotation of object is the same as the the rotation of Earth. On the other hand, if an object has inclinations above 90° and below 270°, the orbit of the object is called as retrograde orbit which means that moving in the orbit in the direction opposite to that of the Earth in its revolution around the Sun.

Right Ascension of the Ascending Node, Ω: The orientation or swivel of an orbital plane with respect to the reference direction is specified by right ascension of the ascending node (RAAN). It is also called as longitude of ascending node. It aligns the orbit ellipse in space. The intersection of the orbit plane and the equatorial plane is called the lines of nodes.

Argument of Periapsis, ω: The orientation of an elliptic orbit within the orbital plane indicated by argument of periapsis. It is the angle in the orbital plane between the ascending node and the periapsis direction. It ranges from 0° to 360°.

True Anomaly,ν: It specifies the location of spacecraft along its orbital path. True anomaly is the angle measured in the direction of motion from the perigee to the satellite's position at the given epoch time. If the satellite is at 180° true anomaly, the satellite is at the farthest point from the Earth, namely at the apogee. If it is at the 0° true anomaly, it is at the perigee.

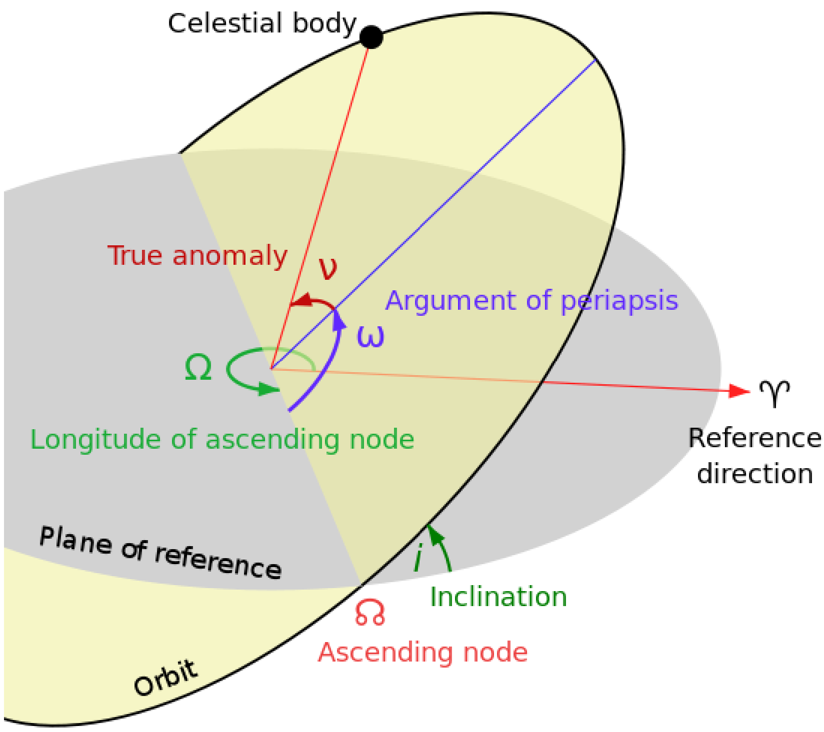

Figure 4.1: Orbital elements of a body [26].

Proposed initial orbital elements of the QB50 CubeSats could be roughly simulated when they are deployed. The proposed initial values are 380 km altitude and 98 degree inclination in a circular orbit. Certain LTAN value of the QB50 mission has not yet been confirmed by the QB50 Management Team. However, it has been announced that the possible LTAN values will be between 08:00 and 14:00. Hence, four of the possible LTAN values from 08:00 to 14:00 is illustrated to give an insight of initial deployment position in Figure 4.2.

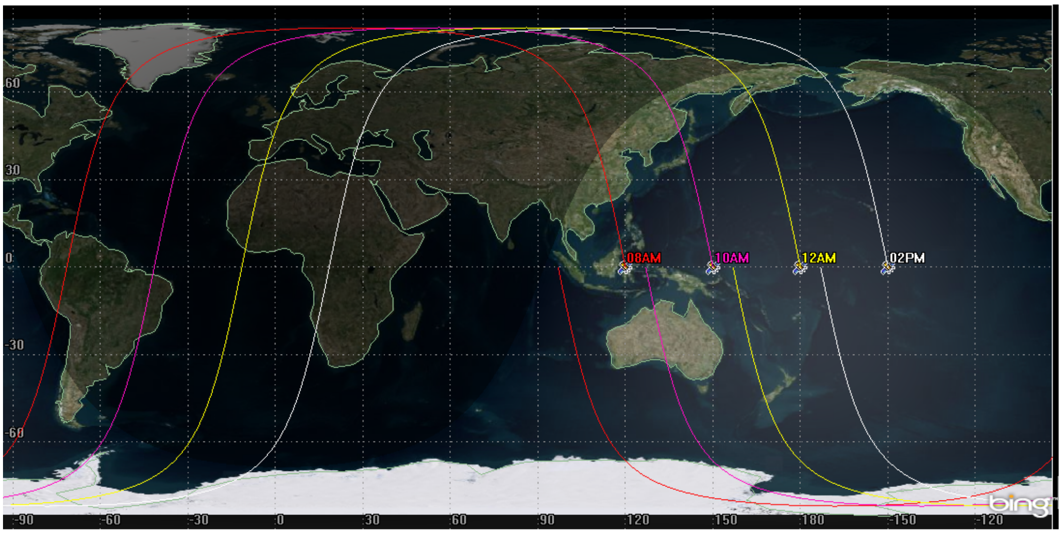

Figure 4.2: Possible initial orbital position of QB50 with respect to LTAN.

4.2 Orbital Perturbations

The satellite trajectory is described by its orbital parameters. A satellite in an ideal situation, without disturbances, the orbit would prevail infinitely. However, the satellite is affected by a number of disturbances in reality. Keplerian orbits have to be investigated to understand the behaviour of a satellite. Keplerian orbits are the closed form solutions of the two body equation.

$$\ddot{\mathbf{r}} = -\mu \frac{\mathbf{r}}{r^3} \tag{4.1}$$

where $\mathbf{r}$ is the CubeSat position in an Earth Centered Inertial frame, $\mu = 3.986 * 10^{14} \frac{m^3}{s^2}$ is the Earth gravitational constant. The equation 4.1 can be written as an assumption that there are two objects in space, symmetric gravitational fields are the only source of interaction between them [25]. An orbit of a satellite in an ideal two body system describes a conic section. There are several effects that cause the conic section to continuously change. These deviations from the ideal Kepler's orbit are called perturbations. Some examples of these perturbations are atmospheric drag, non-spherical earth effects, solar radiation pressure and third body effects. If the perturbation acceleration is added to the right hand side of equation 4.1, the dynamics of Earth orbiting spacecraft is governed by the equation:

$$\ddot{\mathbf{r}} = -\mu \frac{\mathbf{r}}{r^3} + \mathbf{P_a} \quad \textbf{(4.2)}$$

The magnitude of $\mathbf{P_a}$ is smaller than the primary gravitational acceleration [27]. The relative importance of orbit perturbation with respect to spacecraft altitude could be seen in Figure 4.3. For each effect the logarithm of the disturbing acceleration are

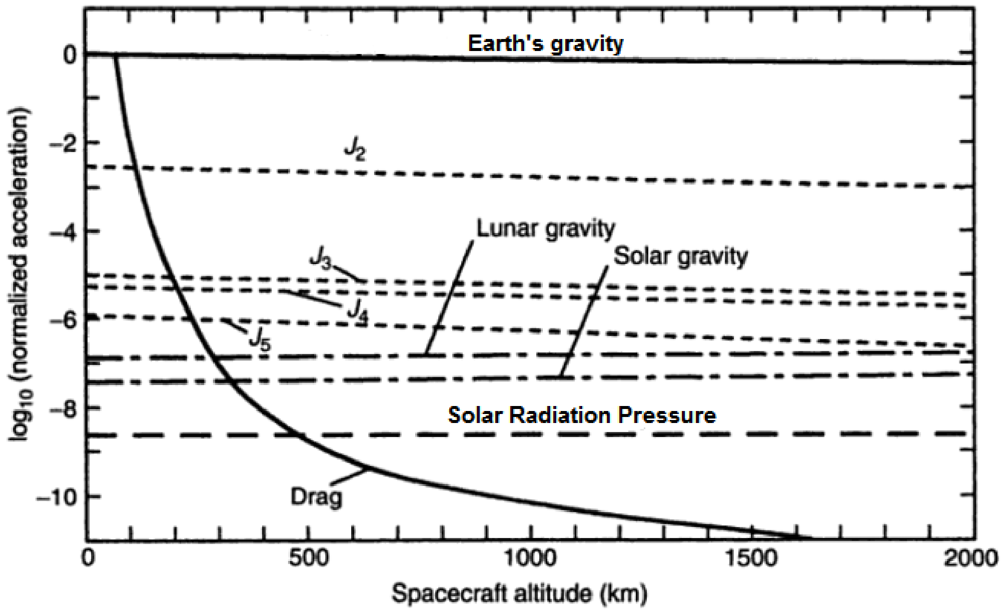

Figure 4.3: Orbit perturbations with respect to spacecraft altitude [27].

normalized to 1 g. The dominant perturbations for low earth orbit are second and higher-order harmonics of the gravity field of earth and atmospheric drag [28]. In order to investigate the further effects of other perturbations to BeEagleSat, from 380 km altitude to end-of-life altitude, a simulation is performed in Section 5.

The disturbing (perturbing) forces make the Keplerian orbital elements time dependent. The perturbation techniques are special and general perturbation techniques and they can be used to predict the time dependent orbit. The underlying approach is the Variation of Parameters for these approximations [25]. Special perturbation techniques use the accelerations which are integrated numerically to find orbital state vectors. This method requires initial conditions and specific force models and greater computational time comparing with the general perturbation method. The Cowell and Encke methods could be given as examples of specific perturbation techniques. Cowell method uses the equation of motion of satellite with all perturbations and integrates numerically step by step [29]. Encke method uses sum of accelerations and integrates with difference from primary acceleration and all perturbation accelerations [30].

4.2.1 Atmospheric drag

Orbital lifetime is mainly driven by atmospheric drag, in the case of a Low Earth Orbit [31]. Drag removes energy from the orbit which causes orbital altitude to reduce and the satellite is affected by more drag. The drag depends on satellite area to mass ratio, satellite velocity relative to the atmosphere and the atmospheric density. The inertial velocity of the satellite is $\mathbf{v}$ and the velocity of the atmosphere at the point is $\mathbf{v_{atm}}$ then the satellite velocity relative to the atmosphere can be written as;

$$\mathbf{v_{rel}} = \mathbf{v} - \mathbf{v_{atm}} \tag{4.3}$$

It is known that the atmosphere rotates with the Earth. The atmospheric velocity relative to the origin of the geocentric equatorial frame with respect to satellite position vector can be written as $\mathbf{v_{atm}} = \omega_\mathbf{E} \times \mathbf{r}$ where $\omega_\mathbf{E}$ is angular velocity vector of the Earth and $\mathbf{r}$ is the satellite position vector. Then, relative satellite velocity is expressed as;

$$\mathbf{v_{rel}} = \mathbf{v} - \omega_\mathbf{E} \times \mathbf{r} \tag{4.4}$$

The drag force on an object acts in the opposite direction to the velocity vector and the $\hat{\mathbf{v}}_{rel} = \mathbf{v}_{rel}/v_{rel}$ is the unit vector in the direction of relative velocity, drag can be written as;

$$D = \frac{1}{2}\rho v_{rel}^2 C_D A_{ref} \tag{4.5}$$

where ρ is atmospheric density, A_{ref} is the frontal area of satellite which is normal to velocity vector, C_D is the drag coefficient. If the drag force divided by the mass of satellite, the perturbation acceleration due to drag force can be found.

$$\mathbf{P_a}(h) = \frac{1}{2m}\rho(h) v_{rel} C_D A_{ref} \mathbf{v_{rel}} \tag{4.6}$$

The ballistic coefficient might be written as $B = \frac{C_D A_{ref}}{m}$ which shows that larger mass with the same reference area results to lower the drag acceleration. The effects of mass distribution is shown at Chapter 5. Atmospheric density is the main cause of drag perturbation [28] and solar weather, geomagnetic activity, satellite altitude, and the time of the year affects the atmospheric density. Also, selection of density model would affect the drag calculation. Due to uncertainties of the mission, an accurate prediction could be difficult but the approximations may help to predict as accurate as possible. Changing with atmospheric density model and drag area with mass of CubeSat, the lifetime prediction also varies. A corresponding simulations are analysed and the results are presented in Chapter 5.

4.2.2 Gravitational osculating parameters

The Earth is not a perfect sphere. The equatorial bulge exerts a force that pulls the satellite back to the equatorial plane and enforce to align the orbital plane with the equator. Due to its angular momentum, the orbit behaves like a spinning top and reacts with a precession motion of the orbital plane which means that the orbital plane of the satellite rotates in inertial space [25].

$$\frac{dh}{dt} = -\frac{3}{2}\frac{J_2 \mu R^2}{r^3}\sin^2 i \sin 2u \tag{4.7}$$

$$\frac{de}{dt} = \frac{3}{2}\frac{J_2 \mu R^2}{hr^3}A_1 \tag{4.8}$$

$$\frac{d\theta}{dt} = \frac{h}{r^2} + \frac{3}{2}\frac{J_2 \mu R^2}{ehr^3}A_2 \tag{4.9}$$

$$\frac{d\Omega}{dt} = -3\frac{J_2 \mu R^2}{hr^3}\sin^2 u \sin 2i \tag{4.10}$$

$$\frac{di}{dt} = -\frac{3}{4}\frac{J_2 \mu R^2}{hr^3}\sin 2u \sin 2i \tag{4.11}$$

$$\frac{d\omega}{dt} = \frac{3}{2}\frac{J_2 \mu R^2}{ehr^3}A_3 \tag{4.12}$$

The equations from 4.7 to 4.12 show the changes of orbital parameters in time, angular momentum, eccentricity, true anomaly, right ascension of ascending node, inclination and argument of perigee, respectively.

$$A_1 = \frac{h^2}{\mu r} sin\theta(3\sin^2 i \sin^2 u - 1) - \sin 2u \sin^2 i[(2 + ecos\theta)cos\theta + e] \tag{4.13}$$

$$A_2 = \frac{h^2}{\mu r} cos\theta(3\sin^2 i \sin^2 u - 1) + \sin 2u \sin^2 i \sin\theta(2 + ecos\theta) \tag{4.14}$$

$$A_3 = \frac{h^2}{\mu r} cos\theta(1 - 3\sin^2 i \sin^2 u) - \sin 2u \sin^2 i \sin\theta(2 + ecos\theta) + 2ecos^2 i \sin^2 u. \tag{4.15}$$

where $u = \omega + \theta$, By the help of the above equations with the given orbital parameters of CubeSats, it can be determined the effect of the J_2 perturbation on the variation of orbital elements. The averaging Ω and ω values along the period of the satellite will give nodal regression and apsidal rotation, respectively. They are secular effects. They can be expressed with a straightforward manipulation as Equation 4.16 and 4.17.

$$\dot{\Omega}_{avg} = \frac{1}{T}\int_0^T \dot{\Omega} dt = -\frac{3}{2}\frac{\sqrt{\mu} J_2 R^2}{(1-e^2)^2 a^{7/2}} cosi \tag{4.16}$$

$$\dot{\omega}_{avg} = \frac{1}{T}\int_0^T \dot{\omega} dt = \frac{3}{4}\frac{\sqrt{\mu} J_2 R^2}{(1-e^2)^2 a^{7/2}} (4 - 5sin^2 i) \tag{4.17}$$

where T is the period of orbit and a is semi-major axis length.

5. SIMULATIONS AND ANALYSES RESULTS

The initial launch and mission characteristics of QB50 mission were given in Chapter 4. In theory, the mission of CubeSats could be analysed from 380 km altitude with 0 eccentricity which makes the orbit circular, and 98°inclination may be sufficient for initial calculations in STK. However, RAAN, argument of perigee and the true anomaly are still missing to fully define the deployment state. Moreover, the orbit errors of Cyclone-4 can be found in Figure 5.1.

There are also other uncertainties. Some of these doubtful situations are deployment related, CubeSat related and calculation related issues that are sequence of deployment, time interval of successive deployment, direction of deployment, mass of CubeSat, reference area variations, density - gravity models and Sun's future activity could be given as examples. Because of these reasons, the simulations are modelled by means of the data provided by the user's manual of the Cyclone-4 [32] and suggestions provided by QB50 management team [33].

N°	Orbit	Parameter	Expected limit deviation ($\pm 3\sigma$)
1	Circular H_{circ}= 500 km i = 90°	Δh_{circ} Δi	± 5,0 km ± 0.05 – 0.08 degrees
2	GTO H_{α}= 35,980 km H_{π}= 170 km i = 5.1°	Δh_a Δh_{π} Δi	±100 km ± 5 km ± 0.05 – 0.08 degrees
3	SSO H≈ 700 km i = 98.1° ω = 84° e = 0.002	Δh Δi $\Delta \omega_{\pi}$	± 6 – 7 km ± 0.05 – 0.08 degrees ± 10 degrees

Figure 5.1: Orbit deviation values of Cyclone-4 [32].

According to the data provided by user's manual of launcher, upper stage will be used for orbit insertion of CubeSats. The mass of upper stage is 4300 kg and it has a liquid jet system which has 6 hours lifetime. Also, it has approximately 20 minutes thrust capability. This means that controllable orbit insertion for all CubeSats has to be within 20 minutes [32]. CubeSat deployment mechanism will be mounted to the upper stage of Cyclone-4 and deployment directions may change with respect to its orientation in space. This will be controlled by actuators of upper stage. Due to the integration restrictions of deployment mechanism, there will be a cyclic deployment normal to in-track during Cyclone-4 upper stage thrusting [34]. In addition, it is possible to deploy CubeSats from 0° to 360° azimuth and elevation angles with respect to velocity vector of the body of deployer. Possible deployment direction can be seen in Figure 5.2. Furthermore, initial values and specifications for orbital analyses are itemized below;

- Deployment is taken at the 380 km altitude with 0 eccentricity and 98°inclination.
- Deployment velocity is taken as 1 m/sec [35].
- The initial argument of perigee is set to 0°.
- The initial RAAN and true anomaly values are set to different values regarding to the simulations.
- BeEagleSat mass is taken 2.259 kg, 2.300 kg and 2.383 kg.
- Reference area of BeEagleSat is taken in a range of 0.010 m^2 to 0.022 m^2.
- Atmospheric density model is selected as NRLMMSISE-00 model [33].
- Gravity model is selected as WGS84-EGM96.
- Solar flux values are taken as monthly prediction from Celestrak [28].

As a consequence, all these estimations will change the orbital elements and corresponding results. BeEagleSat mass and reference area will change the ballistic coefficient. Also, corresponding to these solar flux, gravity models and density models will also change the behaviours of BeEagleSat in space. Due to uncertainties, level of the complexity of analyses is high. Important calculations of BeEagleSat's orbital missions are presented in the following sections as sunlight duration, lifetime prediction, coverage ability and communication ability. These estimations are used

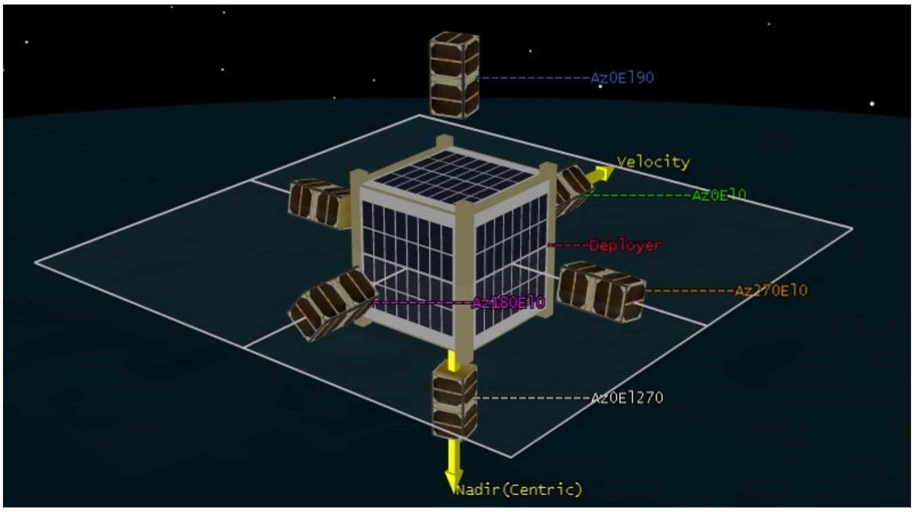

Figure 5.2: BeEagleSat possible deployment directions [8].

for simulations and analyses. The initial point is selected as deployment point for analyses. Reference orbital elements for deployment are given in Table 5.1. BeEagleSat reference area can be averaged in a range of $\pm$ 10° angle of attack (α). If a

Table 5.1: Initial orbital elements of deployment

Semi-major Axis, a	6758.14km
Eccentricity, e	0
Inclination, i	98°
Argument of Perigee, ω	0°
RAAN, Ω	233.89°
True Anomaly, ν	0°
Epoch Time	Jan 15, 2016 00:00 UTC

2U CubeSat in Y-Thompson phase, the reference area based on an geometric approach can be calculated by using;

$$A_{ref} = 0.02 \times sin\alpha + cos\alpha \times 0.01 \quad \textbf{(5.1)}$$

So, the mean value of reference area for $\pm$ 10°angle of attack is 0.01167 m^2. The reference are values for Y-Thompson phase is given in Figure 5.3.

5.1 Sunlight Calculations

When the BeEagleSat is orbiting from 380 km to end-of-orbit, the position with respect to the Sun is changing. This calculation is performed to determine the percentage of time the CubeSat spends in sunlight.

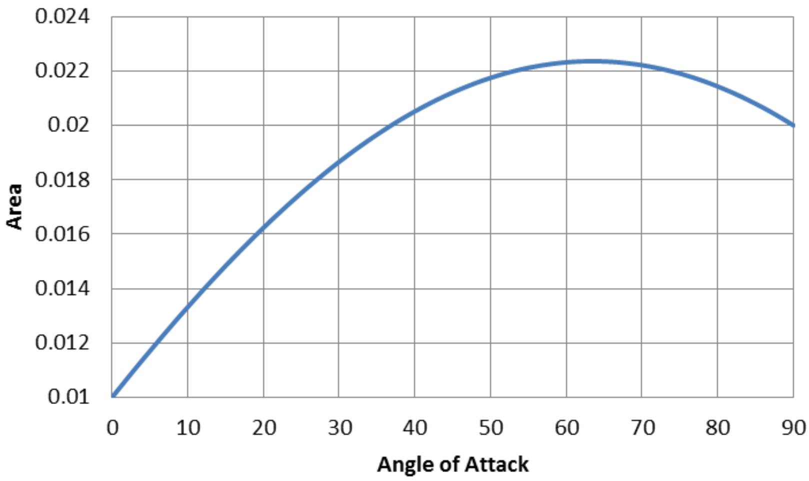

Figure 5.3: Reference area values for Y-Thompson phase.

Although the QB50 team announced that the orbit would be nearly Sun Synchronous Orbit, it is not possible to have a Sun Synchronous Orbit (SSO) by using 380 km circular orbit altitude and 98 degree inclination at the same time. In the case of the circular orbit, each altitude value has a corresponding value of inclination. The plot that represents the relationship between inclination and altitude to have a SSO is given in Figure 5.4 for circular Low Earth Orbit. According to the calculations;

$$\dot{\Omega} = -\frac{3}{2}\frac{\sqrt{\mu}J_2R^2}{(1-e^2)^2a^{7/2}}cosi \tag{5.2}$$

which is where the node precesses at a rate equal to the Earth's mean motion around the Sun;

$$\dot{\Omega} = 360^\circ/365.2422199 = 0.9856^\circ/day \tag{5.3}$$

According to the QB50 Project Management the orbital sunlight period of CubeSats is likely to be at most 65% of the orbit period and may reduce at lower altitudes. With the orbital initial values of BeEagleSat, corresponding sunlight calculation with Systems Tool Kit gives the results as cumulative sunlight duration is 274.628 days and cumulative gap is 163.372 days. According to this calculation, a visual chart which shows the percentage of sunlight can be seen in Figure 5.5.

5.2 Lifetime Prediction Calculations

There are several cases that affect the satellite lifetime. The most important one is atmospheric drag in Low Earth Orbit. After ejecting CubeSat from the deployer, tip-off rate causes tumbling and drag area increases correspondingly. Also, solar flux

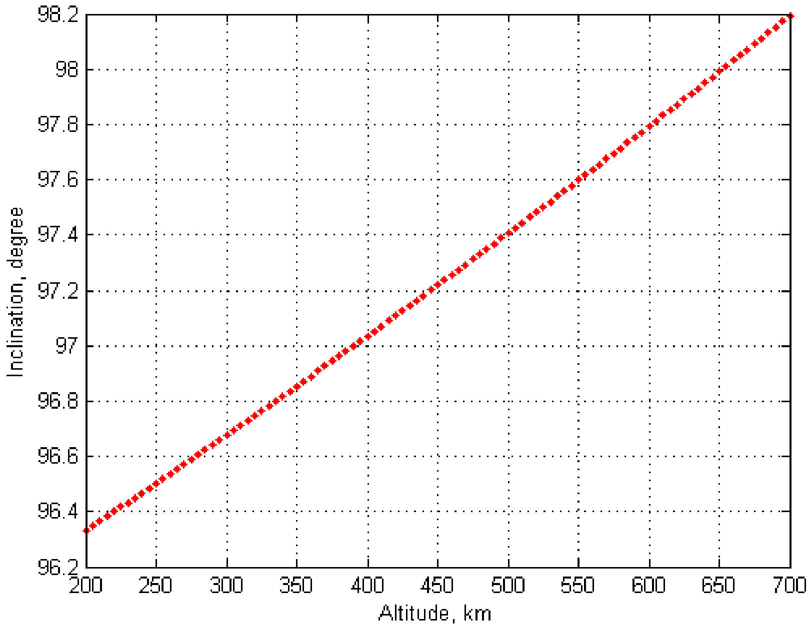

Figure 5.4: Altitude versus inclination plot for circular SSO in LEO.

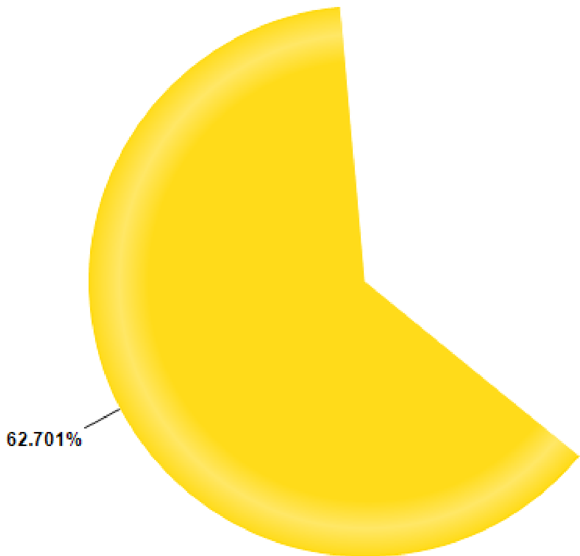

Figure 5.5: Sunlight percentage of BeEagleSat for whole orbital life.

values do not have constant values. Atmospheric model and gravity model, which are the vital part of lifetime calculations, should not be taken as basic models due to the establishing accuracy. Deployment directions and velocity changes are also important to investigate.

For lifetime prediction analysis, High Precision Orbit Propagator with Runge-Kutta-Fehlberg 7-8 integrator is adopted for all cases. Suggested mean coefficient of drag, atmospheric density model and gravity model are 2.3, NRLMSISE-00 and WGS84-EGM96, respectively. The mass of BeEagleSat is taken with three different values to see the effects on lifetime. Lifetime prediction calculations are investigated in two ways: The first one is the full simulation which is taking values from deployment to end-of-life altitude, the second one is the sum of partial simulations of the CubeSat from deployment to Y-momentum phase and from that phase to end-of-life. Y-momentum phase is relatively stable and small drag area can be achieved easier than the previous phase. There are numerous analyses to investigate, hence a systematic approach should be adopted to prevent redundant analyses. As a consequence, the adopted approach consists of five analyses. Each analysis is done to inspect the behaviour of the CubeSat as well as the lifetime of them.

5.2.1 Analysis of the effects of deployment velocity

The first step of the calculations is deployment velocity effect. As previously explained, NRLMSISE-00 atmospheric model and WGS84-EGM96 gravity model are selected. The reference area is taken as 0.01 m^2, C_D is taken as 2.3 and the mass is taken as 2.3 kg.

Deployment velocity is proposed to be between 1 m/s and 2 m/s [35]. This velocity change could affect the orbit mostly if deployment direction is in velocity vector (in-track) and anti-velocity directions. The change is relatively small because the mean orbit velocity at 380 km altitude in circular orbit is 7.6799 km/sec. Both delta-v changes and effects on original orbit is given in Table 5.2. These kind of small delta-v changes may seem to have negligible drawbacks; however, lifetime analyses show that the changes for lifetime from initial altitude to 200 km is approximately 1 month. Corresponding table can be seen in Table 5.3. In addition, a lifetime graph of BeEagleSat that have no Delta-v is depicted in Figure 5.6 in order to give a reference lifetime calculation.

5.2.2 Analysis of the effects of satellite mass

Secondly, in order to see the effect of masses in lifetime, three different masses are adopted. The least one is possible minimum mass (2.259 kg), the medium one is the QB50 proposed limit mass (2.300 kg), the largest one is current mass of BeEagleSat

Table 5.2: Orbital elements change with deployment.

Without Δv	In track, 1 m/s Δv	Anti Velocity, 1 m/s Δv
Apogee:380km	Apogee:383.463km	Apogee:380km
Perigee:380km	Perigee:380km	Perigee:376.527km
Inc:98°	Inc:97.999°	Inc:98.001°
ω : 0°	ω : 359.999°	ω : 180.001°
Ω : 233.899°	Ω : 233.899°	Ω : 233.899°
ν : 0°	ν : 0.001°	ν : 179.999°

Table 5.3: Deployment velocity effects to lifetime.

	Original Orbit	In track, 1 m/s Δv	Anti Velocity, 1 m/s Δv
Deployment	Jan 15, 2016	Jan 15, 2016	Jan 15, 2016
End-of-Orbit	Mar 2,2017	Mar 19,2017	Feb 14, 2017
Lifetime	1.1 Years	1.2 Years	1.1 Years

(2.383 kg). The drag area is taken as 0.01167 m^2 for each CubeSat. Gravity and atmospheric models are identical with the previous simulation. The effects of mass change to CubeSat can be seen in Table 5.4.

Table 5.4: Effects of mass change to lifetime.

Mass	2.259 kg	2.300 kg	2.383 kg
Deployment Date	Jan 15, 2016	Jan 15, 2016	Jan 15, 2016
End-of-Orbit Date	Feb 23,2017	Mar 2,2017	Mar 17, 2017
Total Lifetime	1.1 Years	1.1 Years	1.2 Years

5.2.3 Analysis of the effects of drag area

This analysis presents the effects of changing the reference area to the lifetime. In this analysis, CubeSat's mass is taken as 2.3 kg, and reference area is changed from 0.010

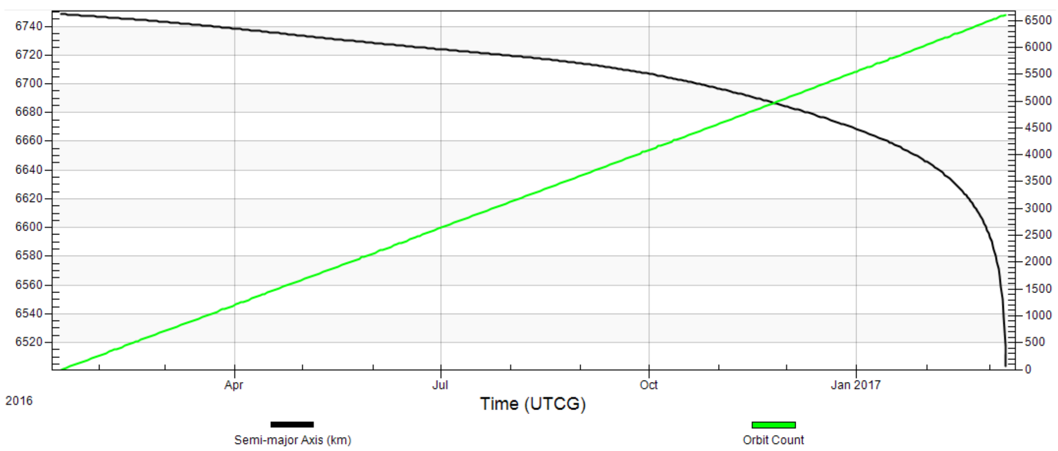

Figure 5.6: Lifetime prediction of BeEagleSat.

m^2 to 0.022 m^2 . The corresponding table of this analysis can be found in Table 5.5. It can be seen that the difference of the reference areas between 0.010 m^2 and 0.011 m^2 is changed the lifetime 40 days. This means that in order to have a longer life in orbit for BeEagleSat, transition between Y-Thompson mode and Y-Momentum mode should be obtained as soon as possible.

Table 5.5: Reference area changes effects to lifetime.

Reference Area m^2	Lifetime (days)
0.010	412
0.011	372
0.012	339
0.013	314
0.014	294
0.015	276
0.016	261
0.017	246
0.018	232
0.019	218
0.020	205
0.021	193
0.022	182

5.2.4 Analysis of the effects of ballistic coefficient

It is obvious that larger mass provides longer life in orbit but it is important to mention that this analysis made by assuming that the CubeSats have the same reference area. Hence, the ballistic coefficient is affected only by mass. Considering the mass of BeEagleSat has 10% deviation, ballistic coefficient normal distribution for mass can be calculated. Reference area is taken 0.01167 m^2 and mean mass is taken as 2.3 kg with 0.05 standard deviation. This analysis would help to show the effect of ballistic coefficient on the lifetime. Forty CubeSats with different masses are selected by the help of the calculation of the mass distribution. In this simulation, they are released from the deployment mechanism and the behaviour of CubeSats are inspected. The first noticeable behaviour of CubeSats is that their motion in space seems a spiral movement. This motion is a relative motion of CubeSats with respect to the upper stage. The figure 5.8 is captured after 3 hours from deployment. The second one, although there is no order of deployment such as from the largest BC to the least BC, the CubeSats tend to be in a sequence of the least BC to the largest BC. The

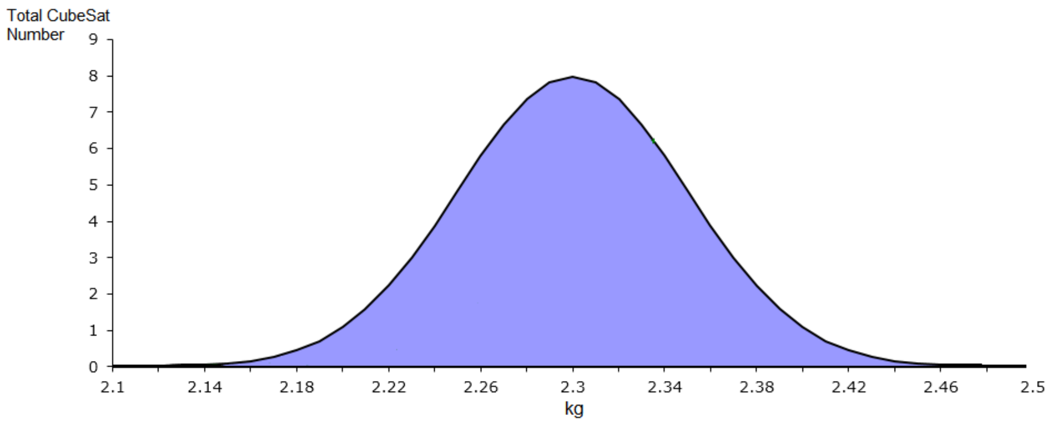

Figure 5.7: Normal distribution of BeEagleSat mass.

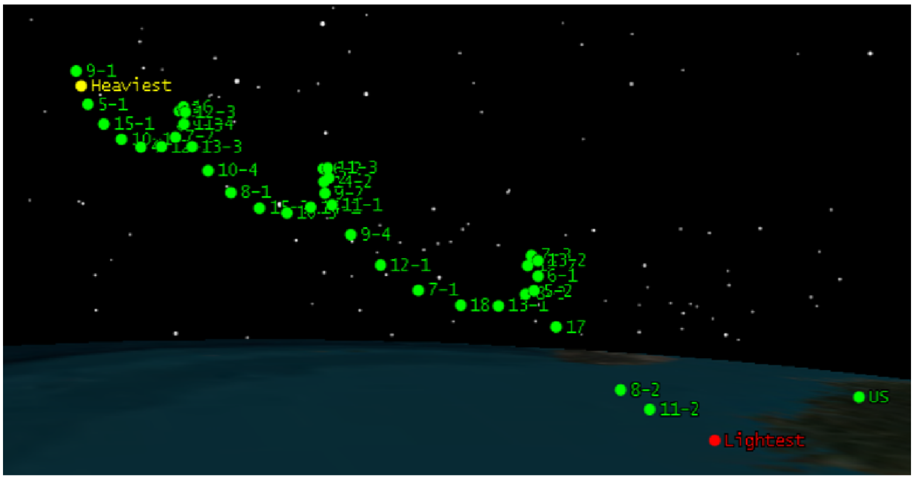

Figure 5.8: Relative motion of CubeSats.

corresponding effect can be seen in Figure 5.9 which is captured from the simulation after 3 days from the deployment. The third one is that the CubeSats reach the string of pearl formation nearly 20 days after the deployment. This can be seen in Figure 5.10. Also, according to the results, more ballistic coefficient provides more lifetime. It can be concluded that the important characteristic for lifetime is not the mass, but the ballistic coefficient.

5.2.5 Analysis of tumbling effects for two different initial conditions

The previous analyses show that the initial deployment velocity, mass of CubeSat, the drag area and the ballistic coefficient affect the lifetime. Besides, in order to perform a specific analysis for BeEagleSat, two different cases are simulated.

For the case 1, BeEagleSat has minimum mass, ejected in anti-velocity direction. In addition, drag area was taken near to maximum possible drag area from deployment to y-momentum phase. After this phase, BeEagleSat has nominal values. It is assumed

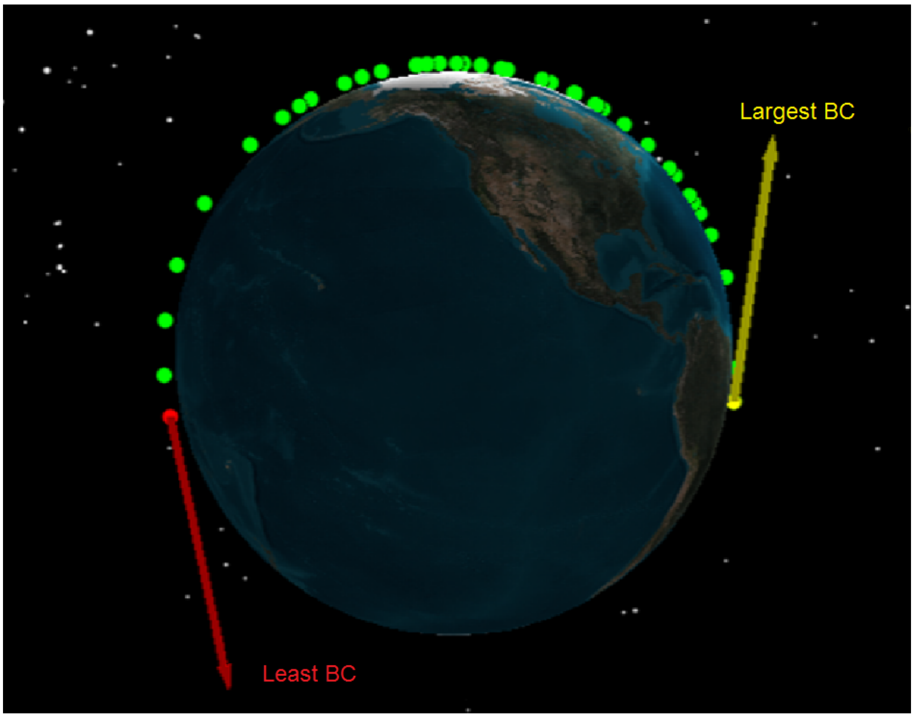

Figure 5.9: Effect of ballistic coefficient to position of CubeSats.

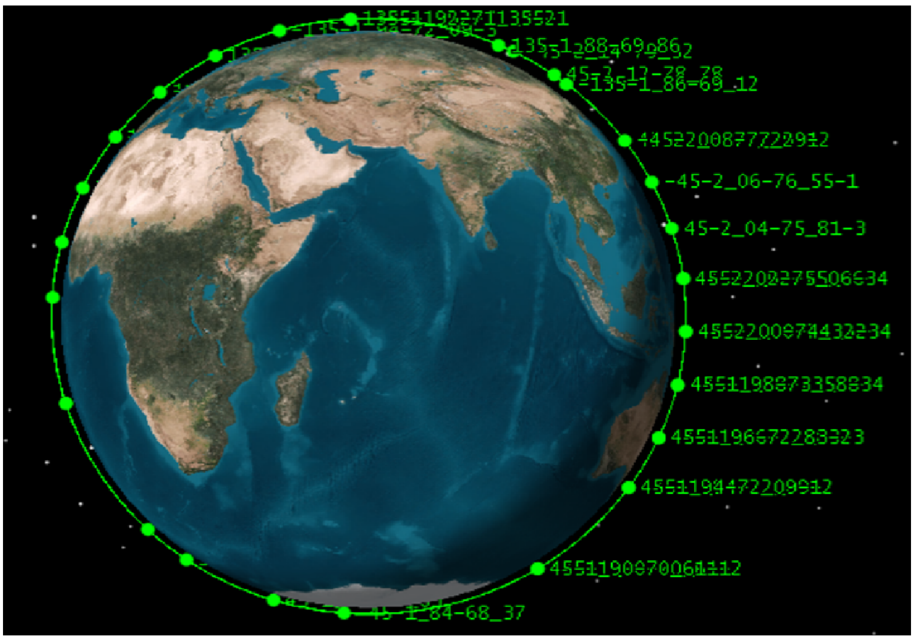

Figure 5.10: String of pearl configuration.

Table 5.6: Summary of case 1 and case 2 to predict lifetime.

	Case 1	Case 2
Mass	2.2590 kg	2.3830 kg
Deployment	Anti Vel. Direction	Velocity Direction
Tumbling Ref. Area	0.02200 m^2	0.01600 m^2
Op. Phase Ref. Area	0.01167 m^2	0.01167 m^2
Lifetime	329.0050 days	378.0640 days
End of Life Date	09 Dec 2016	28 Jan 2017

that initial deployment conditions of first case are not suitable. However, it is supposed that ADCS unit can handle to stabilize the CubeSat in 1 week.

For case 2, the mass of CubeSat is at the maximum value and the CubeSat is ejected in velocity-direction. The tumbling effects are assumed less than case 1 and drag area is taken as 0.01600 m^2 from the deployment to Y-momentum phase. Also, the time interval to be in Y-momentum phase is relatively less than previous one, 2 days, because ADCS unit can handle this small tumbling effects faster. This case is more suitable than case 1.

During operational phase, it is assumed that both satellites have 0.01167 m^2 average reference area. In the final analysis, total lifetime of case 1 and 2 can be summarized as indicated in Table 5.6.

5.3 Coverage Ability Calculation

The number of ground stations distributed all over the world for QB50 mission defines the probability that a CubeSat can communicate these ground stations in a period of time. Distribution of baseline ground station and coverage for 380 km altitudes for QB50 mission is shown in Figure 5.11.

The coverage data is important to estimate the communication intervals of the satellite. Its importance is getting more significant when the life of orbit is limited. To be able to estimate the necessary time interval, the ground track and accessibility should be analysed in advance. The ground track is the projection of the orbit of satellite on Earth surface. As an example, in order to observe initial ground tracks of BeEagleSat, accessibility simulation is performed between Istanbul Technical University Ground Station and BeEagleSat for the first thirty ground tracks. The corresponding ground tracks illustration is given in Figure 5.12.

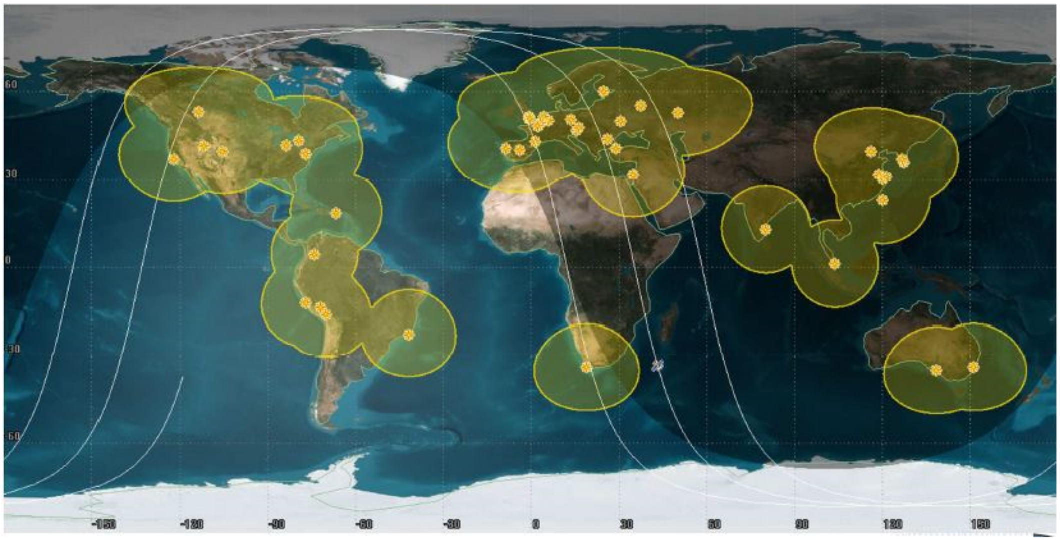

Figure 5.11: Distribution of ground stations for QB50 mission [13].

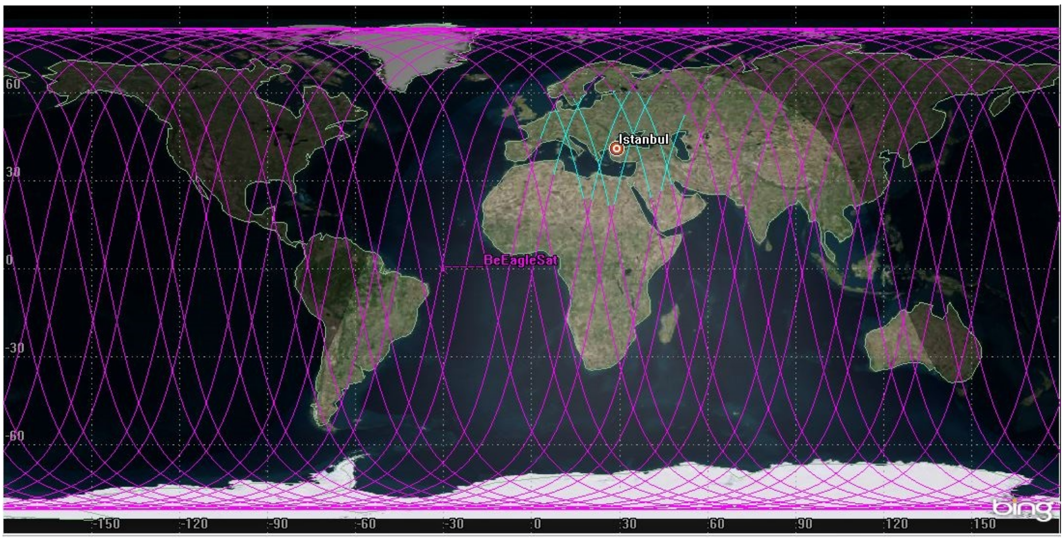

Figure 5.12: First thirty ground tracks of BeEagleSat.

5.4 Communication Ability Calculations

Generally, it is not enough to know the ground tracks of satellite in order to calculate communication time of satellites. This knowledge is just the beginning of further calculations. The satellites can be observed from horizon with respect to ground station; however, communication with satellite is possible with some degree of angles. The elevation angle is the main responsible of this. The value of the elevation angle is generally taken as 15 degree for calculations in order to take into account the effects of obstacles e.g. high buildings. Regarding to the elevation angle of BeEagleSat, communication may be interrupted at low elevations. The previous experiences

Table 5.7: Access intervals of BeEagleSat for seven days after deployment.

Day	Duration (sec)	Kbit data with 60% efficiency
1st	825	4749.742
2nd	790	4550.619
3rd	686	3953.935
4th	825	4749.742
5th	928	5346.179
6th	676	3894.555
7th	690	3976.957

of Istanbul Technical University Ground Station show that elevation angle can be considered as 10°. The elevation angle is demonstrated in Figure 5.13.

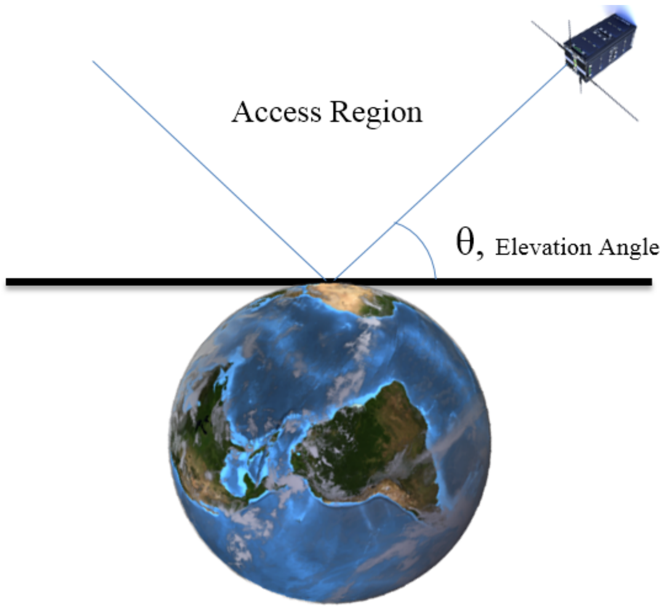

Figure 5.13: Elevation angle and access region of a satellite.

BeEagleSat downlink data rate is 9600 bps [14]. By the help of the coverage calculation results, daily data download size can be predicted as previously discussed. If the elevation angle is set to 10° as seen in Figure 5.14 and BeEagleSat access time is calculated until the end of life, the prediction of downloaded data can be determined. The initial deployment values of BeEagleSat are taken as 380 km altitude, 233.89° RAAN, 0 eccentricity and 98° inclination. The efficiency of the data is set to 0.6 in order to introduce the worst case scenario. The approximate downloaded data per day is given in Table 5.7 for the first 7 days of BeEagleSat after the deployment. Also, mean duration for last 10 days before the end of orbit is approximately 518 second.

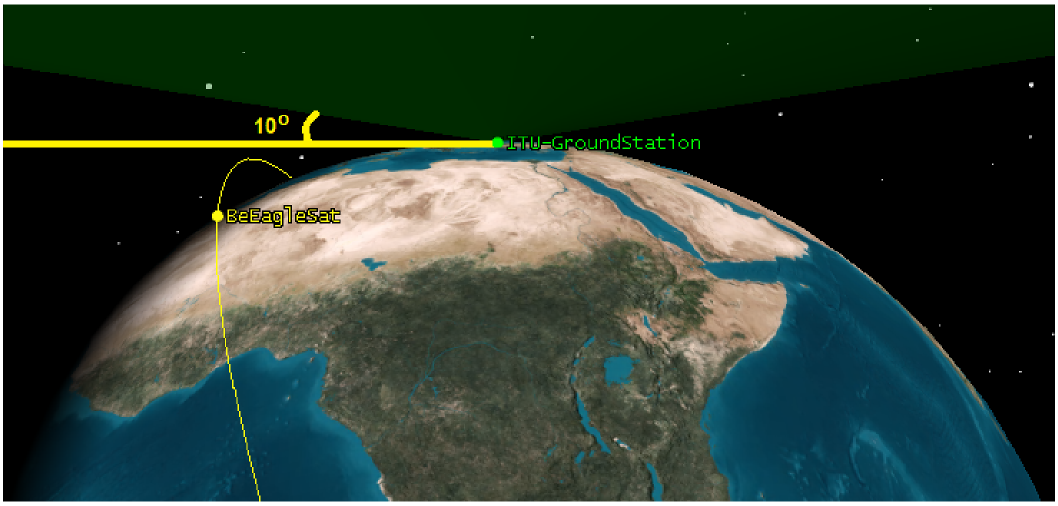

Figure 5.14: ITU ground station with elevation angle.

Table 5.8: Some of the gravitational parameters of 3rd bodies.

Body	Gravitational Parameter,$\mu(km^3/sec^2)$
Sun	1.3271e+011
Moon	4.9028e+003
Jupiter	1.2671e+008

In addition to these results, if BeEagleSat is deployed at 380 km altitude circular orbit with 233.89° RAAN to 98° inclination at the time of 15 January 2016, 00:00 UTC (it is not an SSO), the first access between BeEagleSat and ITU Ground Station will be approximately 5 hours after the deployment. The first access time is an approximate value which absolutely depends on the initial deployment values.

5.5 Extended Orbital Mission Analysis of BeEagleSat

In this section an extended orbital mission analysis of BeEagleSat is performed in order to make a comparison between basic analyses and extended analysis. In this analysis, the tumbling effects are also simulated. First of all, High Precision Orbit Propagator (HPOP) is used for propagation to make the analysis as accurate as possible. It is assumed that CubeSat is deployed in anti-velocity direction of upper stage with 1 m/sec velocity. Orbit epoch is 15 January 2016, 00:00 UTC. Also third body effects on the CubeSat are taken into consideration and investigated with the values of gravitational parameters, μ as given in Table 5.8.

C_D is taken as 2.3 and mass of BeEagleSat is taken as 2300 gr, which is also limit mass for a QB50 double unit CubeSat [22]. Atmospheric model is selected

Table 5.9: Monthly predicted solar activity values.

Date	*ND*	F10.7
Jan,2016	19	106.3
Feb,2016	23	104.8
Mar,2016	25	103.2
Apr,2016	2	101.7
May,2016	5	100.2
June,2016	9	98.7
July,2016	12	97.2
Aug,2016	16	95.8
Sept,2016	20	94.3
Oct,2016	23	92.9
Nov,2016	27	91.5
Dec,2016	3	90.2
Jan,2017	7	88.2
Feb,2017	11	87.5
Mar,2017	12	86.3

as NRLMSISE-00 and gravity model is WGS84-EGM96. On the contrary of basic traditional propagation, this analysis includes relativistic acceleration, albedo and thermal effects, ocean and solid tides effects. Ground reflection model is based on a research and data from Earth radiation pressure effects on satellites [36] and Earth radiation budgets [37], respectively. The corresponding calculation which is a part of a tool of Systems Tool Kit for albedo and thermal emissivity are;

$$a = a_0 + (c_0 + c_1.cos(w.\Delta t) + c_2.sin(w.\Delta t)).P_1(sin(\phi)) + a_2.P_2(sin(\phi)) \qquad \textbf{(5.4)}$$

$$e = e_0 + (k_0 + k_1.cos(w.\Delta t) + k_2.sin(w.\Delta t)).P_1(sin(\phi)) + e_2.P_2(sin(\phi)) \qquad \textbf{(5.5)}$$

where a is albedo, e is thermal emissivity, ϕ is latitude, Δt is time passed from reference epoch, $w = 2.\pi/365.25(rad/day)$, $P_1(x) = x$ and $P_2(x) = 0.5.(3x^2 - 1)$. Furthermore, solar activity values are monthly approximated values which are taken from Celestrak website [38] and can be found in Table 5.9. *ND* is Bartels' Solar Rotation Number which is a serial count. It enumerates the apparent rotations of the Sun as viewed from the Earth that is used for tracking certain changing patterns of solar activity [39]. F10.7 is 10.7 cm radio flux adjusted monthly average value which is a measurement of radio-frequency emissions from the Sun which is approximately correlated with the solar extreme ultraviolet flux [38]. Moreover, integration is done by Runge-Kutta-Fehlberg 7-8 integrator. Step size control done by relative error method with $1.0e-013$ error tolerance. The minimum step size is selected as 1 second and the maximum step size is selected as 86400 second. Interpolation method is done by 7th order Lagrange method. Propagation is set to stop processing at 90 km altitude which is a below altitude of re-entry condition even though BeEagleSat rate of decay below 180 km is very high (approximately 2-3 days).

The extended analysis is started at the given deployment inputs. In order to simulate the tumbling effects the initial angular rates are assumed as $\omega x = 3^\circ/sec$,$\omega y = 0^\circ/sec$ and $\omega z = -2^\circ/sec$. The initial rate values are taken the ZA-AeroSat [23] simulations and added to STK Attitude tool to simulate the CubeSat's attitude in orbit. Then, attitude section of the satellite is set to dump X and Z axis angular rates. It is also set to bring Y axis angular rate into an arbitrary selected reference value of $-3^\circ/sec$. The de-tumbling phase is started 92 minutes and finished 277 minutes from epoch. Then simultaneously, Y-momentum phase is started and it takes 108 minutes.

The extended analysis result for lifetime is that the decay date of BeEagleSat which is the end of life date is estimated to be on 24 March 2017 17:46:52.947 UTC after 6885 orbits. The lifetime is approximately 1.2 years (434.752 days).

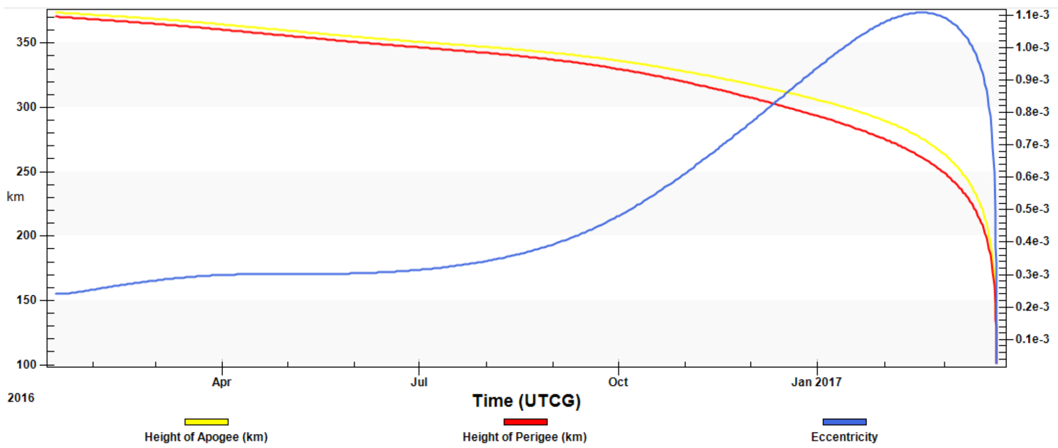

Figure 5.15: Accurate lifetime prediction of BeEagleSat.

The result of the access time duration is also obtained. The minimum access time duration of BeEagleSat is predicted at the 684th satellite pass at 12 September 2016 as 13.226 second. Maximum duration is predicted at the 31st satellite pass at 25 January 2016 as 359.079 second. The overall average duration of access is calculated as 250.852 second per satellite pass. The total duration of communication is calculated as 83.896 hours. It means that ITU ground station can communicate(data download or upload time) with BeEagleSat approximately 3.5 days of 435 days lifetime (0.8% of lifetime), if the elevation angle of ground station is 10°and there is no further constraint or obstacle to interrupt the communication.

The sunlight calculation result of BeEagleSat is quite similar to the previous basic calculations. BeEagleSat cumulative sunlight is 62.781 % which is equal to 275 days. Remaining percent of 37.219 % includes Penumbra and Umbra time interval which are 1.679 days(0.383 %) and 161.34 days (36.836 %), respectively.

6. CONCLUSIONS AND RECOMMENDATIONS

BeEagleSat, a double unit CubeSat prepared as a part of the EU/FP7 project QB50, will be one of the most advanced small satellites of Turkey developed with cooperation of Istanbul Technical University and Turkish Air Force Academy along with contributions from Sabanci University, HAVELSAN and ITU spin off SMEs Ertek Space and Gumush Aerospace. Moreover, being a part of an international CubeSat network will strengthen Turkey's capacity on the small satellite technology.

In the present work, detailed mission analysis of BeEagleSat is carried out. The BeEagleSat is also introduced regarding its design architecture and brief details of related subsystems along with payload specifications. Detailed mission modes are simulated and investigated. Calculations to predict BeEagleSat behaviour in its orbit are presented. As analyses points out, CubeSat behaviour depends on numerous disturbances present in its orbit. However, using up to date and suitable assumptions, in orbit simulations are carried out to obtain accurate results. Thus, BeEagleSat orbital lifetime, access intervals and sunlight exposure are all calculated. Moreover, electric power system, communication and, orbit attitude and control characteristics of the satellite can be accurately determined by carrying out a detailed orbital mission analysis.

The simulations show that higher mass with smaller reference area gives longer lifetime, as expected. Analysis of the effects of satellite masses is performed by using three different masses, 2.259 kg, 2.300 kg and 2.383 kg. Moreover, According to the drag area effect analysis, the difference of the drag areas between 0.011 m^2 and 0.010 m^2 is 40 days. Before launch, the response of ADCS unit should be measured to accurately detect the transition times between the mission modes. Furthermore, the effect of changing LTAN value on initial deployment altitude should be analysed. Sensitivity analyses by using more accurate solar activity index can be accomplished. Current analyses used the most up to date solar activity values present in the STK software. These values shall be updated prior to launch.

In the present work, value of drag coefficient is taken as 2.3, constant. In reality, this value is not constant and an investigation using varying CD values may lead to more realistic results. The POD deployment velocity Delta-V value which is provided by the deployer spring is assumed as 1 m/sec, any change in this value shall also be taken in to account in the analysis.

The horizon elevation value is taken as 10° for the ITU ground station; a more suitable value could be selected by analysing the environment around the ground station. According to the communication ability results, total communication time is approximately 0.8% of the lifetime. The efficiency of downloaded data is taken as 0.6, this could lead a worst case scenario. Moreover, any change in the value of horizon elevation shall be taken in to account in the analysis.

The sunlight calculation result of the BeEagleSat shows that, the cumulative sunlight exposure duration is approximately 63% of lifetime. In addition, lifetime of the CubeSat is varying; however, the lifetime is more than 6 months for the analyses performed in this study. As previously discussed, the lifetime is affected by the initial deployment values, the CubeSat specifications and the perturbations.

According to the ballistic coefficient effect analysis, approximate string of pearl of configuration shall be reached nearly 20 days after the deployment. An accurate analysis could be performed if certain values of the CubeSats masses and the deployment initial values are confirmed. It is also important to point out that, possible deployment directions out of orbit plane or across orbit plane which are perpendicular to the orbital velocity direction could be risky if any other CubeSat is deployed at the same time with BeEagleSat [8,34]. This should be carefully analysed to avoid any possibility of collisions.

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APPENDICES

APPENDIX A.1 : Basic Tutorials for STK Simulations

APPENDIX A.1

This appendix presents a number of tutorials to facilitate use of the STK for CubeSat mission analysis. The tutorial values of the examples presented herein may change depending on the parameters employed. First of all, a scenario has to be defined. The scenario includes desired lifetime for the CubeSat. In this tutorial it is assumed that the timespan is more than three months. For example, it may be from 15.04.2015 to 25.09.2015. After this, proper Keplerian parameters should be written as seen in Figure A.1.

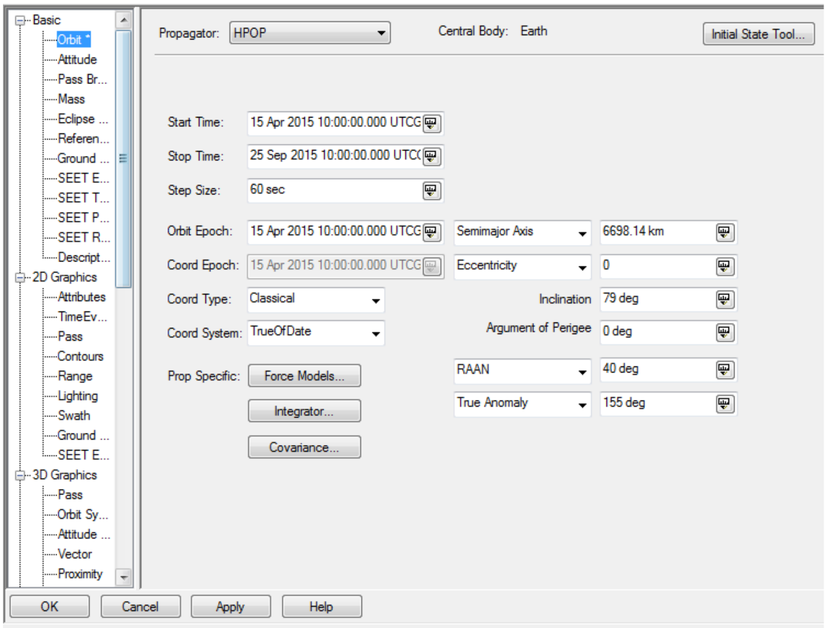

Figure A.1: Parameters for example

After setting the parameters, force models button may be used to enable advanced features. In order to propagate a CubeSat by using HPOP, atmospheric density model, solar flux index, drag values and any further variables may be specified as seen in Figure A.2. The 'Use' button activates 'Drag'. It is important to be sure about the mass of CubeSat, which it can be specified by clicking 'More Options'.

It may be changed atmospheric model while using HPOP at Force Models. Moreover, Solar Flux/GeoMag values may be changed either entering manually or using a file. STK gives a Solar Flux/Ap file if 'Use File' option is selected. This file includes daily and monthly estimated solar flux values.

In order to compare the atmospheric models, solar flux values or propagation methods in STK, the report tool may be used. Each scenario has a different report depending on the analysis of interest. The reports may be seen by selecting Report and Graph Manager. After that, user is able to generate the interested data file as a report or graph type.

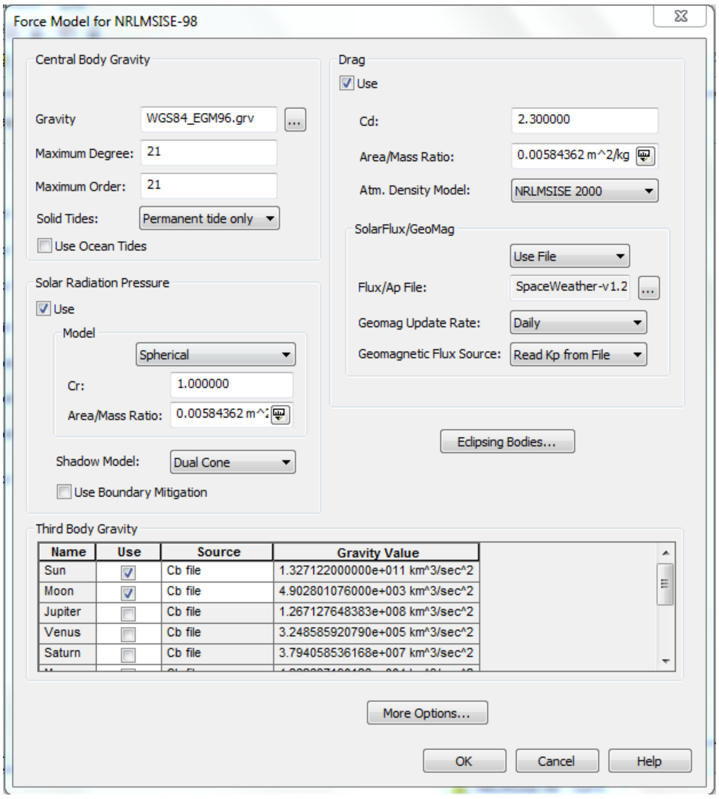

Figure A.2: Force models

An accurate lifetime calculation may be applied by selecting Ephemeris Data. There are also further options to calculate lifetime. STK offers the Lifetime Tool which may be enabled from the satellite options. It is needed to specify the values in the section of Lifetime and it is important to use the same values which is applied at the previous section. The last step should be using the Compute button. The lifetime report and graph types may be seen after using the Compute button.

STK AdvCAT plug-in may be used to observe the risk of collision. An individual AdvCAT may be inserted to the analysis by clicking the New Object button and selecting this object of interest. If the user has specific risk zones to investigate collision risk, it is needed to set these values to the properties of AdvCAT. In properties section of AdvCAT, it can be seen a page as indicated in Figure A.3. In this figure, there are several parameters which would affect the analysis. First step should be selecting the primary and secondary objects of interest at the lists. Several objects may be selected depending on the analysis. Threshold value reports a warning message when the distance between the threat volumes of the primary and the secondary objects falls below this value. The computation process can be started by pressing the Compute button.

In order to determine access time intervals between two objects such as, satellite and ground station, STK Access tool may be used. An access in STK is defined by at least two objects. The Access tool can be activated by selecting the CubeSat properties menu. Coverage data, communication intervals and download/upload data size may be obtained by using this tool.

Deployment of a satellite process from a launcher in STK is more sophisticated than orbiting a satellite using just a propagator in STK. The ejection of CubeSats may be

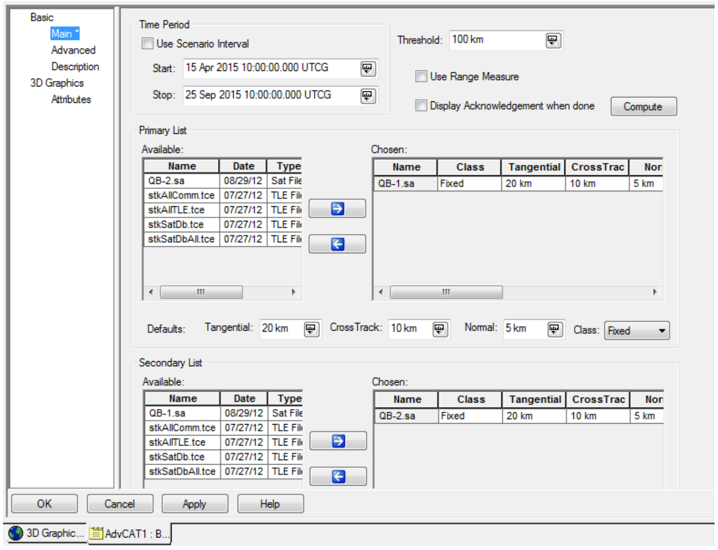

Figure A.3: AdvCAT properties

performed by using Astrogator propagator. The first step of deployment is setting values of a deployment mechanism. After this process, in order to deploy a CubeSat, the user should select the Astrogator as a propagator for deployment mechanism. It will enable the Astrogator plug-in as shown in figure A.4. It may be seen arbitrary launcher, orbital elements, deployment parameters and propagation specifications as shown in Figures A.5, A.6 and A.7, respectively.

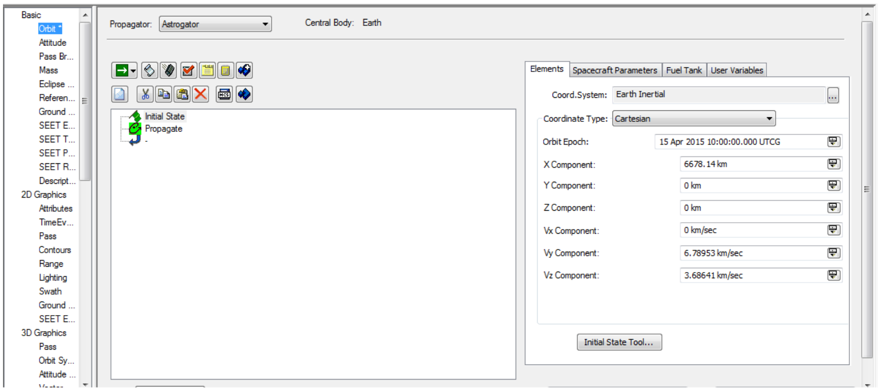

Figure A.4: Astrogator pop-up

After setting these values for deployment mechanism, the propagator of CubeSat should be selected as Astrogator. In order to deploy CubeSats from a deployer, CubeSats should follow deployer within a duration. It may be set the position of CubeSat in Leader's Body Frame option. In Follow section, a leader should be chosen. This leader will be followed from the beginning time to separation time which can be specified at Separation section. An example settings of two minutes deployment interval may be seen in Figure A.8. The Trip value is deployment time of the CubeSat. If the user has more than one CubeSat, it is needed to create and set additional CubeSats. Moreover, multiple ejections may be performed in STK. For example, in the case of 2 min per CubeSat sequence, the first CubeSat is ejected at t time, the 2nd CubeSat should be ejected at t+2 minutes, the 3rd CubeSat t+4 minutes and so on.

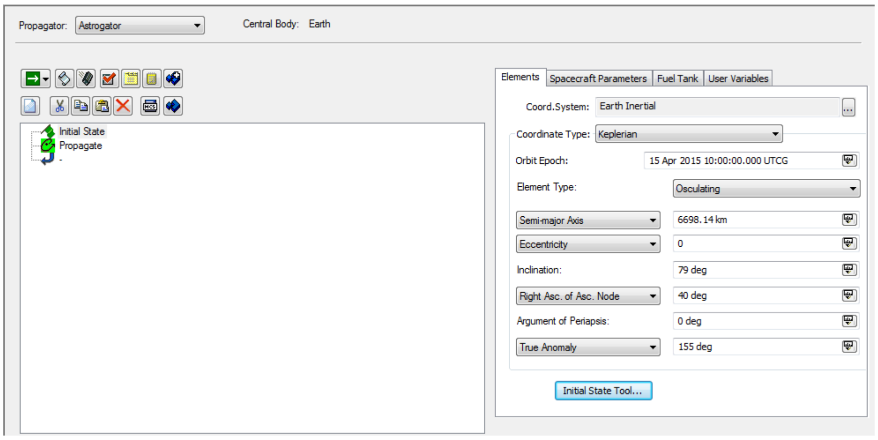

Figure A.5: Orbital elements for deployer

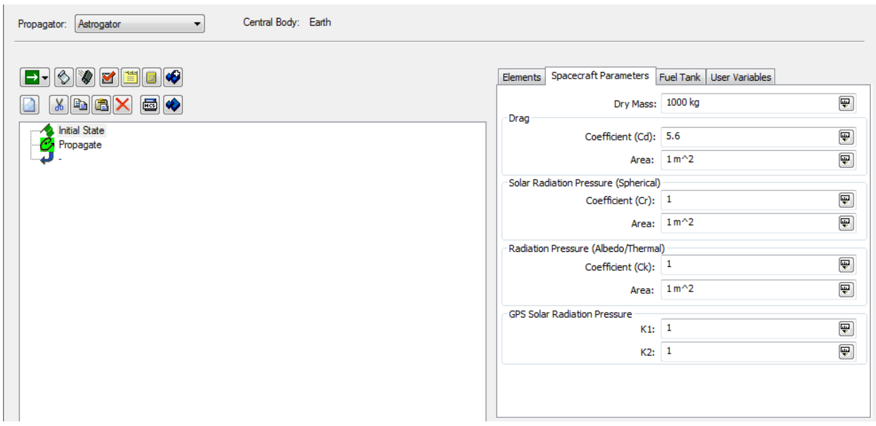

Figure A.6: Spacecraft elements for deployer

The Maneuver segment may be used to specify the direction of deployment. CubeSat deployment from a deployment mechanism is performed by using spring force. To simulate the deployment, at Attitude Control: Thrust Vector selection should be selected. This enables the Thrust Axes options. Although the CubeSat deployment is not performed by using propellant thrust in this tutorial, it may be assumed that the impulsive thrust is performed by the CubeSat. Example figure for -135°azimuth, 0°elevation and 0.0015 km/sec Delta V is shown in Figure A.9. These examples presents one CubeSat and deployment mechanism. In the case of multiple CubeSats deployment, the variables may be changed depending on the parameters employed. It is important that if the user changes the settings of any segment or add values to any segment in Astrogator propagator, the entire segment should be run. Otherwise, the differences may not be applied properly.

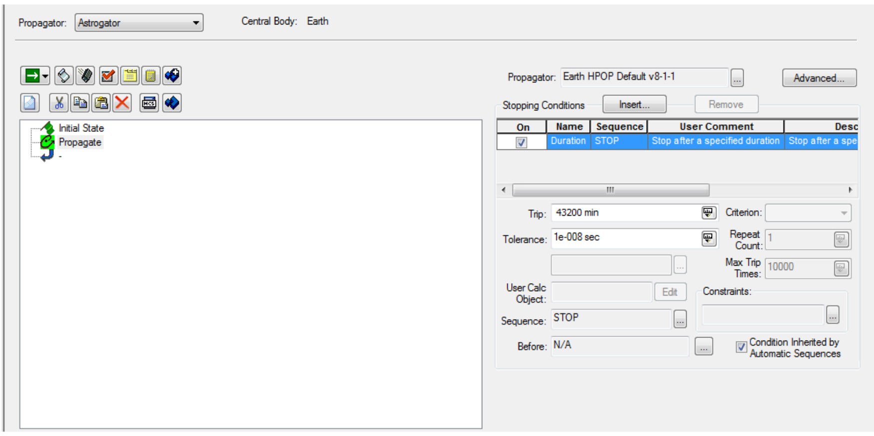

Figure A.7: Propagate options for deployer

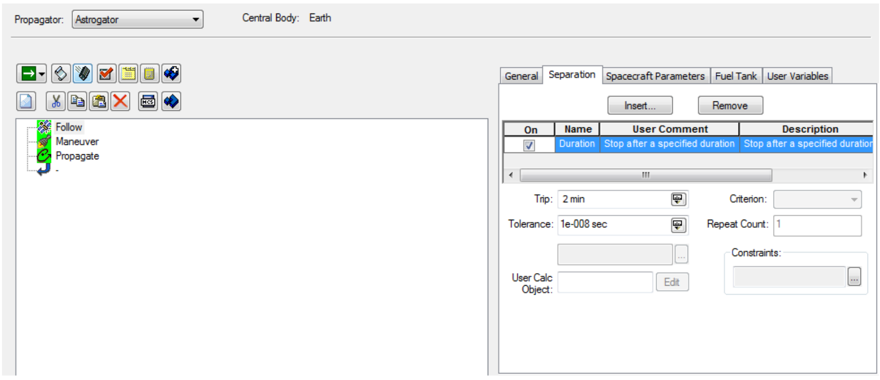

Figure A.8: Separation options for CubeSats

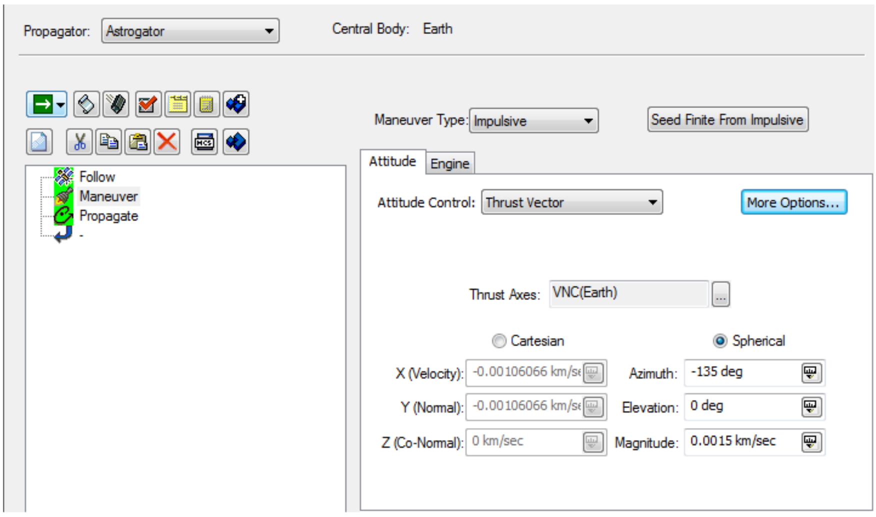

Figure A.9: Maneuver of CubeSat

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