

CRANFIELD UNIVERSITY



W. C. P. Fernando

FEEP THRUSTER NANO-SATELLITE APPLICATIONS

SCHOOL OF ENGINEERING

PhD THESIS

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CRANFIELD UNIVERSITY

SCHOOL OF ENGINEERING

POWER, PROPULSION & AEROSPACE ENGINEERING

PhD THESIS

ACADEMIC YEAR 1998-2001

W. C. P. FERNANDO

FEEP THRUSTER NANO-SATELLITE APPLICATION

SUPERVISOR: Mr. T. S. BOWLING

January 2004

Acknowledgements

I am grateful to all who gave me support and strength while completing my work.

Mr. T. S. Bowling, my supervisor, whom I am and will be forever grateful for his invaluable help, advice and encouragement. But most of all I thank him for helping me to find suitable training at FEEP test sites.

I wish to thank Dr. S. E. Hobbs and Dr. X. Zhang for taking the time to discuss issues and for giving advice during the course of the three years. My thanks also go to Dr. Les Oswald and Mr. R. Wadher who were willing to help and assist with the computer system.

To Dr. P. Roberts and Dr. D. G. Fearn for their advice and guidance.

I owe an overwhelming debt of gratitude to Prof. M. Andreucci, Dr. S. Marcuccio and fellow staff members at the Centospazio test facility, Pisa, Italy for their kind advice and support.

I like to thank Analytical Graphic, Inc. (AGI) for providing me STK training and license whenever I needed it. A special thank you to Dr. Emmet Fletcher and Ms. Lyn Flacco at AGI European head office, for giving me the support and advise for running the STK programme. They were always willing to help me.

I am grateful to Mr. Ton van Overbeek at ESA/ESTEC for providing the ESA/ADS simulation tool and for his kind advise, Mr. David Nicolini at ESA/ESTEC for providing FEEP-10 test data and to Dr. Barry Zandbergen at Technical University of Delft, Holland for sharing his FEEP experience with me.

To my wife, Dinesha for proof reading and most of all for her extreme patience during my studies. For our daughter, Mevanti, who always gave a loving smile to me, morning, noon or night. At the end of my thesis writing our son, Joshua was born. He changed our world, to see things in a different perspective.

Mrs. A. B. Fernando, my mother, for her sacrifices to educate me. Without her courage, I would never have reached so far. My late father, Mr. W. J. T. Fernando, who inspired me while watching the night skies in Sri Lanka. He was ready to sacrifice everything he had for my education. Although I lost him before I reached my goal, I thank him for being a great father. For all the help given to me throughout my studies, my sister Mrs. M. D. Wennberg & brother-in-law Mr. L. T. Wennberg. To my two brothers, Mr. S. N. Fernando & Mr. F. R. P. Fernando and their families, for always taking care of me and being there for me to fall back on at any time.

Mr. & Mrs. G. Persson for sharing all what they had with me and providing accommodation during my undergraduate studies at KTH, Sweden.

I think I am extremely lucky in my life. I thank almighty God for giving me all these beautiful people and my prayers are with them.

Dedication

To my Family for their love and support

To my parents who dedicated their lives to my education,

To my wife's support through this long task,

and especially to my daughter and son,

Mevanti and Joshua

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FIGURE H-3 NEA MODEL 9101B RELEASE DEVICE (PEFFER, 2000) 263

Abbreviations

μN	micro-Newton
μT	micro Tesla
ACS	Attitude Control System
ADS	Attitude Determination Software
AOCS	Attitude and Orbit Control System
ASE	Astronautics & Space Engineering
BNSC	British National Space Centre
BOL	Beginning of Life
CM	Centre of Mass
CP	Centre of Pressure
CPV	Common Pressure Vessel
Cs	Caesium
DARWIN	Detection and Analysis of Remote Worlds by Interferometry Nulling
DOD	Depth of Discharge
DoD	Department of Defence
DS	Deep Space
EO	Earth Observing
EOL	End of Life
EP	Electric Propulsion
EPS	Electric Power Subsystem
ESA	European Space Agency
FEED	Field Emission Electric Propulsion
GAS	Getaway Special
GDP	Group Design Project
GNP	Gross National Product
GOCE Mission	Gravity Field and Steady-State Ocean Circulation Explorer Mission (ESA)
GPS	Global Positioning System
GTO	Geostationary Transfer Orbit
In	Indium
IPV	Independent Pressure Vessel
IRF	Institute för Rymdfysik
I_{sp}	Specific Impulse
ISS	International Space Station
KTH	Kungliga Tekniska Högskolan
LEO	Low Earth Orbit
LISA	Laser Interferometry Space Antenna
LMFI	Liquid Metal Field Ion
MEMS	Micro Electro Mechanical Systems
Mo	Molybdenum
MPD	Magneto Plasma Dynamic
MST	Micro System Technology

MT	Magnetic Torque
MUSES-C	Mu Space Engineering Spacecraft-C
MUSTANG	<u>M</u> ulti <u>U</u> niversity <u>S</u> pace <u>T</u> echnology <u>A</u> dvanced <u>N</u> ano-Satellite <u>G</u> roup
MW	Micro Wheel
NASA	National Astronautics & Space Agency
NMP	New Millennium Program
NSSK	North South Station Keeping
ODIE	Orbiting Drag-free International Explorer
PCU	Power Conversion Unit
PPT	Pulse Plasma Thruster
RAL	Rutherford Appleton Laboratory, United Kingdom
Rb	Rubidium
RW	Reaction Wheel
S/C	Spacecraft
SIMS	School of Industrial and Manufacturing Science
SMART	Small Mission for Advanced Research in Technology
SPT	Stationary Plasma Thruster
SPV	Single Pressure Vessel
SSP	System Specific Power
STK	Satellite Tool Kit
Ti	Titanium
W	Tungsten
W-alloys	Tungsten Alloys
XIPS	Xenon Ion Propulsion System

1 Introduction

This chapter explains the background to the work in the thesis. It briefly presents major propulsion categories and important aspects when selecting an electrical propulsion system. Satellites are categorised according to their total mass and nano-satellites are considered as those satellites where the total mass is in the range 1 - 10 kg. The research objective are summarised together with the adopted methodology to reach this goal. For time being the reader can consider the Field Emission Electric Propulsion (FEEP) as the second-generation ion propulsion system. A short description of the FEEP system background, its current developments and planned areas of applications are discussed. Possible nano-satellite application areas and FEEP plan missions are discussed. A brief description of the outline of the thesis is given at the end of this chapter.

1.1 The attractiveness of electrical propulsion

The orbital perturbing forces such as aero and solar pressure drag, acting on a spacecraft (S/C) will eventually change its reference orbit and the disturbing torques (aero, solar, magnetic etc.) will change its orientation. The demand for a propulsion system arises, as a spacecraft in orbit should be able to counteract these perturbing forces and torques acting on it and to meet its payload demands. Importantly a spacecraft propulsion system is designed to meet the mission objectives and to operate without failure during the mission time. The required energy for these propulsion manoeuvres can be carried in many different ways. In the broadest sense different propulsion systems could be categorised according to the type of stored energy onboard. Basically there are three different types, when considered conventional energy conversion methods. They are:

- Chemical energy
- Electrical energy
- Nuclear energy

The spacecraft propulsion system transforms this stored energy into useful kinetic energy to perform a given manoeuvre. Chemical propulsion is energy limited (category: endogenous), compared to electric propulsion (category: exogenous) which is power limited. Therefore electric propulsion (EP) has the attractiveness that in principle, any amount of electrical energy can be added to a given amount of propellant. This enables the acceleration of propellant masses from moderate to very high exhaust velocities. Due to this unique feature, electric propulsion systems require comparatively low propellant mass per unit thrust. The traditional chemical propulsion systems give low exhaust velocities (Space Shuttle main engines around 5 km/s) compared to electric propulsion that gives around 10 – 100 km/s. This considerably reduces the required amount of propellant mass in EP. The major drawbacks of EP are comparatively low thrust, and the associated mass of the power plant.

Importantly electric propulsion is nothing new as it has been used successfully in many space missions for many years since 1960s and it will continue to do so. The main area of application in the past has been as a station keeping thruster for communication satellites but it has also been used successfully as a primary propulsion system for orbit raising [XIPS-25 on Boeing-702 satellite, (Cassady R. J., 2001)] and in interplanetary missions [DS-1, SMART-1 and MUSES-C (Kuninaka, 2002)] in recent years. Generally, when the mass of the spacecraft system is a decisive factor and when adequate electric power is available, the use of electric thrusters are advantageous. However, the selection of propulsion system depends mainly on the type of mission and must be identified during the preliminary design study. The study of electric thrusters (in this study the 'Field Emission Electric Propulsion' system) for nano-satellite propulsion applications is worthwhile because the available size and mass are limited in this category of satellites. The advancement of technology is towards reduction of structural mass and an increase in system specific powers (power per unit mass of the power source). During this study the satellites are categorised according to their mass as shown below.

- Large satellites: > 1000 kg
- Medium satellites: 500 ~ 1000 kg
- Mini- satellites: 100 ~ 500 kg
- Micro-satellites: 10 ~ 100 kg
- Nano-satellites: 1 ~ 10 kg
- Pico-satellites: < 1 kg

The categorization of satellites given above must take as a rough estimate. In reality, satellites categorisation with respect to their mass breakdown is quite unclear. However, throughout this study nano-satellites are referred to those between 1 – 10 kg of total satellite mass. The author believes that by keeping a clear definition for the nano-satellites, it will be helpful to understand the nano-satellites, electric propulsion system feasibility studies.

Generally in many space missions a large percentage of vehicle mass is propellant at the beginning-of-life (BOL). This is very costly in terms of launch and operation, and unfortunately reduces the most important parameter of the mission - the payload mass. From the designing point of view, the designer strives to increase the payload mass by optimising the various satellite sub-systems. The propulsion system mass for example, depends on the mass of the final S/C, type of propulsion system, propellant characteristics, its storage method etc. In order to reduce the propellant mass, selection of high specific impulse (this is a figure of merit of usage of propellant) thrusters are an obvious choice. In contrast to chemical thrusters, the electric propulsion systems have a comparatively low percentage of propellant mass usage. However when the total propulsion system is taken into consideration, a considerably higher percentage of propulsion system mass consists of power plant and power conversion unit (PCU). As a rule of thumb, the power plant mass is directly proportional to the system's total power requirement. With high specific impulse values, which give high exhaust velocities, the EP power requirement increases. The system specific power (SSP), which is defined as the amount of power generated per unit mass of power generating system, gives a figure of merit of the power plant efficiency. Representative values for near term modern lightweight power plants lie in the range of 5 to 8 W/kg. Research and development in the areas such as lightweight

materials, high efficiency solar-cells/batteries etc, will increase these figures by a factor of 2 to 3 in the near future. Thus with increased specific powers, high specific impulse electric propulsion applications become feasible. However before striving blindly for high specific impulse thrusters, one may need to examine the associated power plant mass and its feasibility for the considered S/C.

There are various types of electric propulsion systems available today. The type that is going to discuss in this study is called Field Emission Electric Propulsion (FEEP) system. In short it can be consider as the ‘second-generation’ ion propulsion system. The traditional Ion thruster generates ions by a method called electron bombardment of a gaseous propellant, such as Xenon (Xe). Applying an electric field then accelerates these ions to high velocities. However in FEEP’s ions are generated directly from a liquid metal ion surface, and then accelerated through an electric field. The main components of a FEEP thruster are shown in Figure 1-1.

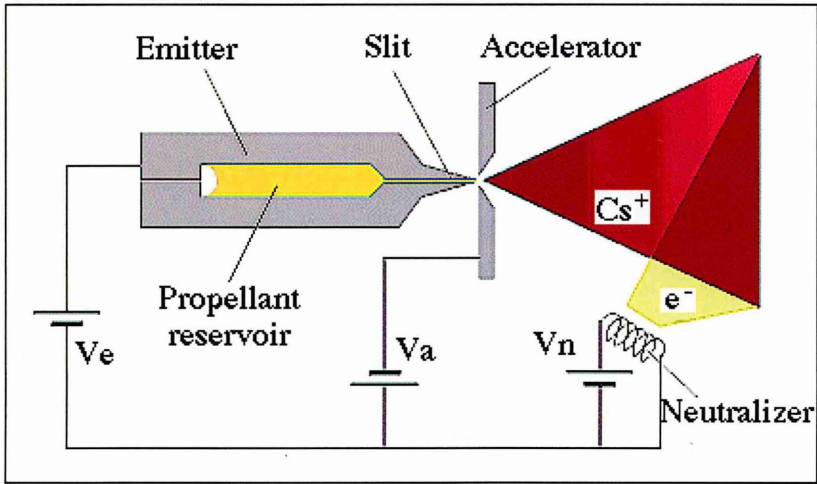


Figure 1-1 FEEP thruster components (Credit: Centrosazio)

As shown in Figure 1-1 FEEP thruster system consists of a metal emitter, which serves as the propellant tank as well as the positive electrode and accelerating electrode (accelerator) and a neutraliser. The emitter is set to a positive potential and the accelerating electrode is at a negative potential with respect to the spacecraft to establish the voltage difference needed for ionisation and acceleration. Thrust is generated accelerating a beam of mainly singly ionised atoms. The required ions are produced directly from the liquid metal surface due to field evaporation. A more detail study of this thruster is given in chapter 4.

1.2 Research objectives

The main objective of this study is to investigate the FEEP (Field Emission Electric Propulsion) thruster suitability as a propulsion mode for nano-satellites. This needs to be validated by studying a possible FEEP/nano-satellite mission. Taking FEEP as the main propulsion mode for a nano-satellite, its capability to provide the attitude manoeuvres while maintaining orbital perturbations within satisfactory limits must be investigated. The FEEP thruster advantages as a nano-satellite propulsion system as well as its limitations need to be mentioned.

Although the main objective of this research is to achieve the above-mentioned goals, secondary sets of objectives are derived in order to make this study more feasible. By studying the possible areas of nano-satellite applications, it was understood that low Earth orbit (LEO) applications are more feasible with the current nano-satellite development. Further by studying possible nano-satellite applications, it was identified that the selected propulsion system (FEEP) must provide high attitude and orbit control accuracy. Therefore in line with possible nano-satellite applications a secondary set of objectives were added to provide feasible design criteria. The following are taken as the basic design goals for a possible nano-satellite propulsion system.

- Low earth orbit applications ranging from 600 km – 1500 km
- Orbit maintenance and accurate attitude control with respect to ground stations and satellite constellations
 - Attitude control accuracy less than 0.5 deg
 - Orbit control accuracy less than 1 metre
- The propulsion system must fit into the nano-satellite mass criteria
 - Total spacecraft wet mass less than 10 kg at the beginning-of-life
- Identify a mission time for a possible FEEP demonstrator mission to validate its usefulness as an attitude and orbit control thruster for nano-satellites. Here the nano-satellite will be used as a platform to demonstrate the FEEP thruster capabilities
- Study on pulse and continuous mode operation of FEEP thrusters
- Identification of possible sensors and instruments for FEEP thrusters in a space demonstration mission
- Identify power system for FEEP thruster demonstrator mission taking current technology development (specific power in the range of 1 – 4 W/kg)

1.3 Methodology

The research is mainly restricted to LEO applications in 600 – 1500 km altitude range, as most of the nano-satellite applications will be in this range for a foreseeable future. Therefore the LEO impacts on the thruster and the satellite are studied in-depth. This gave a broader set of demands that any propulsion system must meet, in order to operate a given satellite successfully. In line with this, FEEP propulsion system capability with different emitters to provide basic attitude and altitude corrections are taken into study. Special emphasis has been laid on to investigate applications in proportional thrust control. This capability is demonstrated using a general case study. Main spacecraft sub-systems that will be affected directly by the FEEP thrusters are discussed in order to estimate the mass and power requirements. However an in-depth investigation is not carried out in these areas, as it deviates the main research objectives.

A case study is used to demonstrate the feasibility of a possible mission. This was selected in line with the parallel studies done at Cranfield University, in the study of nano-satellite applications. ASE (Astronautics & Space Engineering) Group design project results from 1998/1999 to 2000/2001 are used whenever appropriate, for the

considered demonstration mission. Appropriate modifications for the current FEEP thruster configuration and satellite are mentioned and discussed whenever necessary.

The operational mechanisms of FEEP thrusters are studied under the broad field of electrostatic thrusters. In order to highlight the main characteristics in FEEP, basic operational criteria of alternative electric propulsion systems are explained. Past and present missions with an electric propulsion system onboard are studied to get an overall view of possible areas of application. Preliminary investigations in this research study are split into two distinctive fields as shown below.

- The capabilities, demands and limitations of the FEEP propulsion system.
- Present and near term developments, possible applications and mission requirements of the nano-satellites.

A 'FEEP' demonstration mission was selected as stated above in line with the parallel work at Cranfield University. The goal for this demonstrator mission is set as the successful operation of the FEEP system in space, as this technology has no space qualification experience until now. The 'Cranfield University' proposed nano-satellite, MUSTANG (Roberts et al. 2002) was selected as a platform for a possible FEEP demonstrator mission. The following two mission scenarios are considered as likely to arise for a FEEP/nano-satellite demonstrator mission. These are analysed as separate case studies.

- The validation of the FEEP thruster performance, when launched as a payload on-board a nano-satellite.
- Formation flying scenario with respect to another spacecraft.

The latter condition arises, as nano-satellites are attractive when launched in stack configuration. The two S/C can be either identical or different, which will cause a differential drag. FEEP thruster performance with respect to both of these situations will be discussed. Importantly in the formation flying study, it is assumed that only one S/C is equipped with FEEP thrusters, as that will be the most possible scenario at this initial stage.

A possible nano-satellite/FEEP demonstrator mission needs to prove the high attitude and orbit control accuracy associated with FEEP thrusters to pave the way for future advanced missions such as XEUS, DARWIN and LISA (Bagnasco. G. and El Hamel, 2001), where precise relative spacecraft positioning is required. However this requires accurate attitude and orbit determination sensors and methods (requirement for sophisticated star trackers, control algorithms etc.), which may fall outside the current nano-satellite technology.

In order to meet the objectives of this study and reduce the complexity, the initial research study was performed mainly in two separate areas: the FEEP propulsion system and nano-satellite technology readiness. This allowed evaluating the capabilities of the propulsion system (FEEP) and nano spacecraft requirements individually. The adopted research outline is shown in Figure 1-2 and relevant topics are addressed separately.

For the ultimate success of a nano-satellite mission study, it requires a broad knowledge and in-depth investigations in almost all spacecraft subsystems. Therefore a literature survey had been used extensively to understand possible areas of application, sub-systems that have a direct impact due to FEEP thrusters, and their performances within the limits of nano-satellites. The encountered attitude and orbit perturbations for a given nano-satellite were studied to estimate the demand on thrusters.

The FEEP thruster operational oscillations (noise) are considered in order to understand the nano-satellite platform fine pointing capability with this propulsion system. Due to very low thrust levels, the required attitude and orbit correction times are studied in order to understand the operational frequency and the feasibility of this thruster. Different propulsion modes, such as thrust amplitude, its duration, associated power levels etc, are considered. Those propulsion modes that showed a more promising performance in nano-satellite applications than the rest are chosen for further study. Emphasis is given to understand the required modifications for optimal operation of the FEEP system. Following are the main areas of the thrusters physical and operational characteristics that are taken into in-depth study.

- Required thrust levels and hence the electrical power demand
- Propulsion system mass
 - Dry mass (Structure, tanks and tubes)
 - Propellant mass
 - Power generating and conversion system mass
- Size / mass reduction possibilities and their effects. (Miniaturisation)
- FEEP Performance within the limits of available power and mass
- Constant / variable thrust operation, type of oscillations and its propulsion period
- Suitable propellants and on board storage capability. (Contamination problem)
- Propellant management and feeding techniques.
- Thruster / satellite interference effects

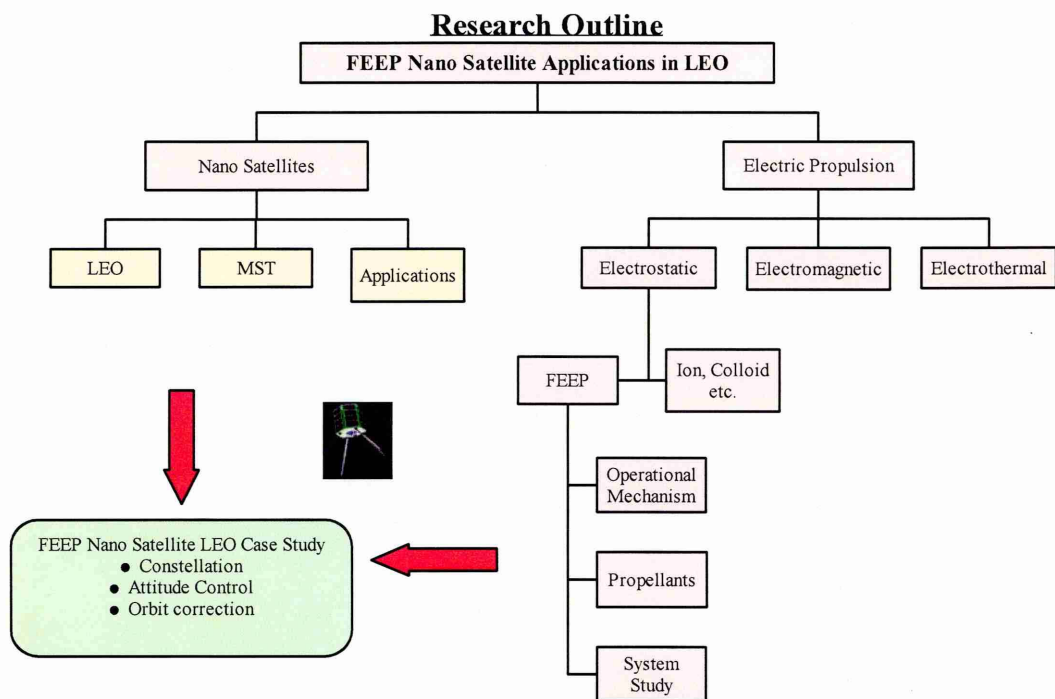


Figure 1-2 Adopted approach for FEEP thruster application in Nano-Satellites for LEO (MST: Micro System Technology)

For the initial sizing it was assumed the S/C power generation using ‘body-mounted’ solar-arrays for simplicity of analysis. The projected area of the spacecraft is considered as the effective area of the solar array, for the calculation of generated power. However, required trade off studies needed to optimise the satellite configuration and power system design are left for future studies. For a possible FEEP demonstrator mission it is assumed that the selected nano-satellite platform supplies the required power for the FEEP thruster operation. For a self-powered short-term (less than one week) demonstrator mission, solar-array/battery combination is considered as the power source. The mass of the power plant is taken as proportional to the maximum power, for simplicity of initial calculations.

Current technologies and possible future trends that will influence the developments of the FEEP thrusters are focused in this study. Alternative electric propulsion systems are categorised according to their thrust levels, power requirement, mass, size, propellant type etc. Possible miniaturisation technologies with these electric propulsion systems and their feasibility as a nano-satellite propulsion system are highlighted in different sub sections.

Past and present technologies with possible future developments are studied using trade-off scenarios to understand the feasibility of FEEP/nano-satellite missions. A case study in formation flying is carried out using two ‘MUSTANG’ type satellites. Perturbations encountered for the considered satellite is calculated for different circular low-earth orbits. Initially it is assumed that there are no other secondary actuators available and all the attitude maintenance is done, using FEEP thrusters. However, passive systems, such as reaction wheels are considered for attitude control applications. During the formation flying case study, simple targeting and range measurements are considered. Attitude control capability is analysed together with off the shelf available sensor characteristic.

The commercially available aerospace design program Satellite Tool Kit version 4.3, (Analytical Graphics Inc. 2003) is used mainly to analyse the formation-flying scenario. The ESA ADS-V2 attitude control and system design software tool, is used to investigate the FEEP and other passive actuators attitude control capability.

Special emphasis was given in this research study to look for possible solutions for FEEP/Nano-satellite applications, within the limits of FEEP thruster propulsion capabilities. The missions are analysed taking into account possible LEO orbits and their impacts such as perturbations, FEEP contamination problems etc. Those areas that are taken into study with respect to nano-satellites in LEO are given below.

- Encountered perturbations.
- Missions of interest in LEO with respect to nano-satellites.
- Nano-satellite development & possible areas of application.

Alternative electric propulsion systems are discussed in general at the beginning of the study. Their basic performance and applications are looked at from the perspective of nano-satellites. Then the FEEP operational mechanism and its different sub systems are discussed. Suitable FEEP propellants and the ways of operating the thruster/propellant contamination free are taken into consideration here. The power generation, storage and management is discussed in order to provide the viability of the FEEP thruster. Performance of the thruster/satellite is studied in order to optimize the integrated system. A case study is used to validate the performance of the total system. Simulations are performed to show the capabilities of the FEEP thrusters. Special attention is given for the FEEP thruster LEO performance. Finally to give the reader an idea about the accuracy of results, simulation programs that are used were discussed.

The adopted parallel study approach is outlined in Figure 1-2. The satellite and FEEP thruster are taken as two distinctive subjects for initial study. This independent approach is selected to gain unbiased results. However the generated requirements and capabilities are added to optimize the integration of the FEEP with nano-satellites.

The author took into account the results from the ASE student Group Design project work carried out during the calendar years 1998/99 to 2000/01. As the literature survey is an ongoing project, it was carried on throughout the research study. However it was tailed off towards the latter part of the research to place more emphasis on the trade-off studies. Taking into account the possible low earth orbit application, extensive studies of alternative propellants were done.

The three months on site experience gained from the 'Centrospazio' test facility in 1999, is used to give a unique perspective for this study. Although most of the experiments carried out during this time used Rb as propellant, its experience was extended to the Cs propellant. During a familiarization visit to ESA/ESTEC done in 2000, FEEP-10 chamber test were analysed. With this experience the FEEP thruster operational modes were compared with the Centrospazio and ESTEC vacuum chamber test results. However, computer simulations are used mainly to understand the performance of the FEEP thrusters. FEEP thruster performance evaluation is

highly relied on the required control modes and the vacuum chamber test experience. Required thrust levels, operational stability, power demand etc, are looked at when selecting an appropriate emitter for a possible demonstrator mission.

1.4 Literature Survey

According to the adopted methodology, the literature survey was performed in two parallel areas: propulsion system and nano-satellite performance. Literature studies were carried out in all major areas in electric propulsion in the beginning of the study. Importance was given to electrostatic thrusters as FEEP characteristics falls into this category. Some of the results in this research study are gathered from the parallel studies done at Cranfield University student design projects (GDP) at the Astronautics and Space Engineering (ASE) department from 1998 to 2001 (These MSc student group design projects are documented as internal reports at the Cranfield University, Astronautics and Space Engineering department.). However, the primary objectives of these MSc, GDP work is to generate a high level of design and developmental scenarios for a near term nano-satellite mission. These studies have shown that the success of a nano-satellite mission depends largely on the possibility of developing a low cost mission with high redundancy. The applied scenarios and results by these group studies are used here to develop possible requirements and demands for a nano-satellite demonstrator mission.

During the time period of this study (1998 – 2002) the amount of FEEP associated publications started to increase tremendously. However majority of these papers are concentrated around the FEEP ground test results mainly carried out at ESA/ ESTEC, Centropazio Italy and Austrian Research Centre at Seibersdorf. Therefore in order to use the first hand information, the majority of the work in this paper was taken from ESA and Centropazio internal reports (Aitken et al. 1975). Although this technology is almost thirty years old, there is still much controversy on its operational mechanisms. Zandbergen (Zandbergen, 1986) had carried out a detailed study on FEEP operational mechanisms under an ESA contract. The results of this study were published in 1986 through an internal report to ESA/ESTEC. Therefore his work is serving as a platform for the current FEEP thruster (slit type) studies. The comments that have been made here with respect to the applications and performances of the thruster are based on the observations done at the Centropazio test facility. However, the final works with respect to FEEP thrusters are still under development mainly at ESA/ESTEC and Centropazio/Pisa facilities. Centropazio is a space technological laboratory under the Consorzio Pisa Ricerche (Universita degli studi di Pisa, CNR, IRI). At the Centropazio facility there are three vacuum chambers available for electric propulsion tests (Centropazio Web Site, 1998). From these three, at the time of the visit, one chamber is completely devoted for FEEP thruster testing. Even though it is somewhat smaller than the rest, it provided the best vacuum conditions (2.3×10^{-8} – 2.3×10^{-9} milli-bars) needed for FEEP.

An experimental investigation on possible alternative propellants to Caesium was done at Fulmer Research Laboratories in UK (Stewart et al. 1980) under ESA/ESTEC contract. Several alternative propellants, both pure metals and complex alloys, have been identified as possible FEEP candidates. However, none of them

showed higher performance than Caesium. The major drawbacks encountered with these alternative propellants are due to their characteristics of:

- Low atomic mass, which in turn gave lower thrust than the reference Caesium propellant (the thrust is maximised for large mass-to-charge ratio).
- High melting point.
- Low wetting characteristics.
- High vapour pressure (eg. Cadmium containing alloys).
- Toxicity (eg. Thallium based alloys).
- Possible interaction with emitter material.
- Cost penalty (eg. Gold alloys).

For example with low atomic mass for a given charge, the power requirement will increase due to the increase in specific impulse. Experimental evaluations of these alternative propellants were carried out on Ti, W, Mo, W-alloys and stainless steel substrates as emitter material. This study showed that the above substrates maintained a relatively oxide free surface due to one or both of the following two reasons:

- High to moderate solubility of oxygen in the substrate.
- Instability of the oxide coating when exposed to high temperature, low-pressure (vacuum) environment.

At present majority of FEEP performance tests are carried out using Caesium, Indium and to some extent Rubidium propellants. However, within the frame of nano-satellite studies the following alternative propellants look promising or at least would provide a worthwhile study. This selection was made, taking into account the limitation in electrical power of a nano-satellite. The important characteristics of the selected propellants are summarised in Table 1-1.

Propellant	Primary characteristics			Secondary characteristics							
	T (°C)	M	$\frac{\rho}{\theta}$	P 20°C 50°C		η	Interaction Al Fe Mo Ni Ti				
Cs	30	133	0.0188	-5.5	(Cs) -4.6	1.0	ok	ok			
In-22Bi	72	136	0.0147	-19.4	(Bi) -17.3	0.987	ok	-	ok	I	I
In-37Hg	-30	147	0.0186	-4.1	(Hg) -2.9	0.984	ok	-	ok	I	I
In-31Bi-6Cd	61.5	134	0.0182	-7.5	(Cd) -6.2	0.983	ok	-	-	I	I
Cs-25Na	-29	105	0.0081	-8.8	(Cs) -7.1	0.831	ok	ok			
Cs-50Rb	9	109	0.0112	-5.7	(Cs) -4.7	0.988	ok	ok			
In	156	115	0.0077	-19.4	(In) -18.0	1.00	ok	-	ok	I	I
In-47Sn	117	117	0.0108	LV	LV	0.99993	I	I	I	I	I

Table 1-1 Comparison of some characteristics of preferred alternative propellants and Caesium {table11; (Stewart et al. 1980)}.

Where,
 LV: Low vapour pressure
 I: Inter-metallic compounds are formed or the propellant is soluble to the extent of at least 5%.

- Ok: No inter-metallic compound formation or appreciable solubility according to phase diagrams.
- M: Atom Mass
- T: Melting temperature, °C
- $\frac{\rho}{\theta}$: Electrical resistance (ρ) to thermal conductivity (θ) ratio.
- P: Vapour pressure of the most volatile species, $\log_{10}$ torr, at superheat temperature of 20°C or 50°C
- η : Calculated beam efficiency = Beam deviation with respect to Caesium (Cs) or Indium (In).

Parallel to the slit emitter studies, the ‘Austrian Research Centre at Seibersdorf’ is engaged in developing a pin emitter with Indium as propellant (Fehringer et al. 1999) under ESA funding. Originally the work has been concentrated on developing a spacecraft charge control device and mass spectrometer. However, this was recently modified as a micro-Newton scale pin emitter. The devise was operated with a carbon nano-tube field emission neutralizer built by FEPET (Field Emission Picture Element Technology, Inc., a subsidiary to SI Diamond, Austin, Texas, USA). The attractiveness of this tungsten emitter and Indium propellant configuration is that it was space qualified (William et al. 1998). It has recorded more than 1000 hours of successful operation, during different missions, in orbits as low as 400-km (MIR, GEOTAIL, EQUATOR-S, etc.). According to the feasibility studies carried out in 1996 {table 1; (William et al. 1998)}, this thruster showed stable operational characteristics in 1µN - 100µN thrust ranges. The Indium FEEP microthruster characteristics are given in below.

	InFEEP-25	InFEEP-100
Propellant	Indium	Indium
Thrust range (µN)	1 – 25	1 - 100
Thrust resolution (µN)	< 0.1	< 0.1
Thrust noise (µN)	< 0.1**	-
Minimum Impulse bit (nNs)	< 5	< 10
Specific Impulse (s)	10000	10000
PCU Input power (Thruster + Neutralizer) (W)	11*	22*
Mass (Thruster + Neutralizer + PCU) (kg)	0.8*	1.0*
Overall dimensions (mm3)	100x70x80*	120x90x100*
Status	Parameters exp. Verified	Planned

Table 1-2 Indium FEEP microthruster specifications. (Wolfgang, 2003)

- * Estimated
- ** Over periods of more than 100 s

1.5 Nano-Satellite Applications

The current nano-satellite development gained a new impulse with future visions to develop low-cost, highly reliable, frequent missions, both to explore our solar system as well as the universe. The identified areas of application are much wider, from

detection of objects to inspection and servicing as well as missions to demonstrate technology. There is a growing concern among the space community in the area of space-based surveillance to develop an early warning system. Space surveillance using visible sensors was identified and discussed recently by a couple of authors from Massachusetts Institute of Technology, in order to detect objects in the 'geosynchronous' belt (Gaposchkin et al. 2000), (Burnham et al. 2000), (Sharma, 2000). The 'Free-Flying Micro-Operator' (ESA Contractor Report, 1999) study done by ESA had showed possible areas of application for nano-satellites, especially in LEO. According to this study, a cluster of nano-satellites can be used to serve, especially the ISS (International Space Station), for inspection, maintenance and operational support

The NASA space-borne optical interferometer concept (Lau et al. 1997) is another area of application that is possible to achieve using a constellation of nano-satellites. Here, the spacecraft fly in formation, allowing a separation of a few hundred meters to thousands of kilometres. The required accuracy is in the range of ± 1 cm in relative distance, and ± 1 arc-minute relative angle. This study has identified both cold gas (GN_2) and Pulsed Plasma Thrusters (Teflon, PPT) as viable propulsion systems. However, application of FEEP thrusters in a formation-flying mission would be an attractive option and is taken into study here. An important feature in this study is the identification of the AFF (Autonomous Formation Flying) sensor for measuring relative distances and angles. It has been explained that this sensor could meet the above-mentioned accuracy and is based on the JPL TurboRouge/GPS receiver technology. Another important mission, using 'formation-flying' concept is the NASA's New Millennium Program (NMP) first Earth Observing flight (EO-1). The Earth Observing mission (EO-1) was launched successfully on 21st November 2000. EO-1 has been inserted into a 705 km orbit (circular, sun-synchronous orbit at a 98.7 degree inclination) flying in formation with the Landsat-7 satellite taking a series of images as of Landsat-7. The navigation and support for formation flying is performed with the use of a GPS subsystem and uses Pulsed Plasma Thrusters (PPT). EO-1 is planned to fly behind the Landsat-7, maintaining a one-minute separation to a tolerance of ten seconds (Bauer et al. 1997).

As pointed out above, with the current discussions on cheaper, faster, better missions, there is a growing interest on mass production of satellites and multiple launch possibilities. Especially concerning nano-satellites, similarities have been drawn between future nano-satellite development and the past semiconductor revolution. One such revolutionary development was the 'digital propulsion' (Iannotta, 1999), which uses the chemical propulsion technology. In the frame of nano-satellite micro-propulsion concept studies, (Janson, 1994) FEEP thrusters are identified as one of the attractive options. With respect to multi-launch configuration requirement, one need is to identify the possible launchers and the available launcher interfaces (Ward and Gardner, 1994). Whatever the outcome of these ideas, the future on mass production of satellites is dependent on the possibility of re-programming the same infrastructure for different missions as well as mass/ volume efficient miniaturisation of spacecraft sub-systems. For the commercial success of nano-satellites in the future, the unit development cost needs to be much less than the current values (~ 0.1 Million Euros per kilogram). As shown in Figure 1-3, the development cost of a satellite (micro/nano) does not depend solely by the mass. It is a function of type of mission, applied technology, performances etc. In order to keep the development costs low,

nano-satellites needs to be based around single event (carries a single payload) satellites. Early selection of the launcher is also an advantage, as the current launch cost does not priced as per kg. With the status of the current technology, if strived too hard for small mass the price of the satellite may go high as MST is still under development. In the frame of nano-satellite/FEEP thrusters combination, the initial development costs will be much higher than what is acceptable for the commercial market as discussed above. The main reason for this is that the FEEP thrusters had not gone through the space qualification tests yet and the nano-satellite on the other hand had not got the commercial interest for mass production. However once the space community gets more knowledge in these two areas the nano-satellite and FEEP thruster combination will be an attractive option for many scientific (such as Lisa & Darwin missions) and commercial (redundant communication spacecrafts already deployed in space) applications.

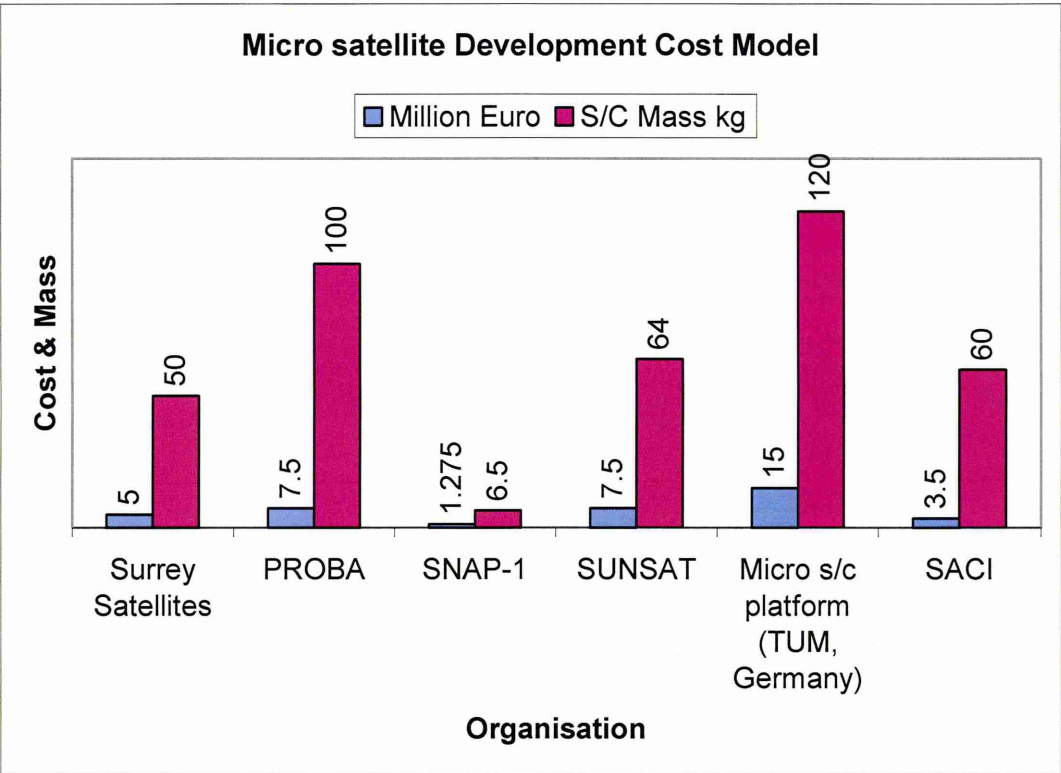


Figure 1-3 Cost and mass comparison of recent micro/nano satellites (Rycroft, 2001).

1.6 FEEP Planned Missions

The European Space Agency (ESA) is planning a following-up mission to SMART 1 (Small Missions for Advanced Research in Technology) for testing new technology. The mission is called SMART 2 (Bagnasco and El Hamel, 1995) and is planned to launch sometime in 2006, will consist of two satellites. The two satellites will be placed in Lagrange point 1 and 2. SMART 2’s purpose is to test and make way for two upcoming ESA missions called LISA (Laser Interferometer Space Antenna) (Danzmann et al. 1995) and Darwin (Fridlund, 2000), and is therefore an important cornerstone for the future. The objectives for LISA and Darwin are respectively, to

detect gravitational waves or ripples in space-time and to find terrestrial earth-like planets in orbits around other stars in the Milky Way. To test the new technologies needed for these two missions, it will be divided up in SMART 2 mission two spacecraft's. One of them will be carrying the instrumentation that will be tested for LISA and both of them will be used for testing the technology needed for Darwin. To accomplish the LISA mission the attitude determination and control system must be able to sensor and control relative angular variations within $\pm 10^{-10}$ radians (Taking into account 5 Million Km separation with a beam convergent mirror diameter of 1 m). The relative separation must also control with highest accuracy. For the Darwin mission the relative position between the two spacecraft, separated with hundreds of kilometres, must be controlled with an accuracy of about 1 micrometer ($=10^{-6}$ m). Therefore SMART 2 will be a test of how well spacecraft can be operated and controlled in space.

The LISA mission consist of three S/C placed in orbits that form triangular formation with centre 20° behind the Earth and side length 5 million km. Each S/C will fly in an individual Earth-like orbit around the Sun. The orbits of the three S/C must hold a relationship between inclination and eccentricity that inclines the plane of the formation by 60° with respect to the ecliptic. The nodal longitudes of the three orbits are shifted 120° to create the triangle. The final adjustments, attitude control and drag free, to the orbits will be done with FEEP thrusters. The launch is currently planned to be sometime in 2011. A summary of LISA mission is given in Appendix-A.

1.7 Outline of the Thesis

The chapters of this thesis have been arranged to cover three main areas: electric propulsion in general, FEEP thrusters and FEEP/nano-satellites performance study through simulation analysis. Chapter 2 gives a brief discussion about small satellites in low Earth orbits. The objective of this discussion is to give the reader an understanding of the current capabilities and limitations in small satellites (nano-satellites - mini satellites) and to investigate the orbital perturbations acting on a reference nano-satellite. Chapter 3 deals with different electric propulsion systems in general and more emphasis has given to ion thrusters and their performances due to its similarities with FEEP thrusters.

Chapter 4 is devoted for the FEEP thruster detail analysis. The current slit type FEEP thruster performances, available neutraliser technology, alternative propellants and safe FEEP operation methods are discussed. Chapter 5 covers the electrical power system current technology trends associated with FEEP thrusters within the limits of nano-satellites mass and power budget constrains. Chapter 6 explains the different simulation tools used for FEEP thruster nano-satellite performance study, which is discussed in chapter 7. Chapter 8 concludes this research study with some recommendations for the FEEP thruster future development.

2 Small Satellites

Development strategies for small satellites are discussed here. This has been further enhanced by taking into account launcher selection and multiple launch capability. In order to succeed in nano-satellite development, it was argued that it requires a shift from the current piggyback launch to economical, commercial, dedicated launcher that could inject a series of nano-satellites into an operational orbit. Some issues that must take into account with FEEP competitive micro propulsion systems are discussed. Orbital and attitude perturbations are taken into account and their effects were discussed. In order to understand these effects and to calculate analytically a simplified satellite model was selected together with primary disturbance torques.

2.1 General

The successful launch of the first satellite, the Sputnik, on October 4, 1957 by Russia, was soon succeeded by the launch of Explorer-1 on January 31, 1958 by the United States. In the beginning of the space era most of these satellites were launched for military purposes, although the first spacecraft was launched for publicity stunt. Since the first artificial 'moon' was launched, thousands of satellites have been launched by different nations around the world. Today with no 'space race', most of the space programs are revised to gain commercial benefits. Communication, digital TV and radio are few such areas that fall into commercial interest today. However, missions such as science and technology demonstrators still need government funding to fulfil their missions, but unfortunately many government space agencies are receiving fewer funds, and they are forced to reconsider about their development strategies. One of the new strategies is a "smaller, faster, better" concept from NASA. For many years development of small satellites are limited to various universities and institutions only. One of the main reasons was, with their limited resources, they could only manage to develop low cost simple satellites. Today there are many Universities and institutions around the world that develop small satellites with low cost but with advanced technology. University of Surrey, Swedish Space Corporation and Institute of Space Physics (IRF) Sweden are few such institutions that had gained experience developing small satellites. However, the applied development strategies and aimed areas of applications are somewhat different and depend on the available experience. The areas of applications of these projects are quite different and vary from students' hands on development projects to advanced national and international space missions.

Small satellite family consists of micro-satellites with masses ranging from few tens of kilograms to mini-satellites with few hundred kilograms. Small satellites can be categorised into three main groups: Commercial, Scientific and technology demonstration according to the financial area of interest. The power of commercial satellite communication was first recognised in 1964 with the real-time transmission of the Olympics. International phone calls took hold and volume pricing began as technology outpaced cost. Space programs became profitable. It has shown investments in the space sector to have a high multiplier effect on the GNP. Cost effective manufacturing, development and operational methods became important to

generate profits. The “Smaller, Faster, Better” is one such proposed concepts. There are more. Yet, none had proved to be the best. Maybe by implementing and testing one will understand what method to adopt.

For a given mission, it is important to know at the very early stages of its development, that the mission is economically viable and the set mission goals are possible. Regardless of the size of the satellite, it must have all the vital sub systems: Propulsion, Attitude control, Telecommunication, Command and data handling etc. It is true that some of the nano and pico satellites were launched in the past without any propulsion on-board. This is mainly with the reduction of mass and volume of a satellite, much of the worries in early stages were about on-board computers and communications. With the successful development of these subsystems, propulsion is becoming a key component of many small, low cost satellites. For many of these small satellites, cold gas propulsion (usually compressed Nitrogen) has been an obvious selection due to low cost and short mission life. But with increase power availability per unit mass and requirement for longer duration missions, electric propulsion systems are becoming a viable option. For example UoSat-12 which had a launch mass of 300 kg, was equipped with two types of propulsion systems (Curiel R.A. da Silva, 2003). While cold gas was used for general attitude and orbit control purposes, Nitrous oxide Resistojet was used to demonstrate its orbit raising capability. Consequently, when considered micro-/nano-satellites, conventional propulsion systems might not be the optimal solution. The field of micro propulsion is still in its infancy, and further development of current concepts is very much needed. On the whole, the following issues would require special attention when dealing with micro/nano-satellite propulsion systems.

- Material compatibility between the propellant and satellite surface material due to the material used in micro fabrication process.
- Contamination problems from the propellant
- Valve leakage
- Passage clogging in propulsion system micro channels.
- System reliability and durability with respect to hazardous propellant storage
- Manufacturing complexity
- Integration complexity

It has been identified that using ‘off the shelf’ components and innovative design features could lead to manufacture economically viable satellites (Watson et al. 1990). This assumes that space qualified miniature components are available. However, micro system technology (MST) applications for space are still in the infancy stage. With any new technology miniaturisation of components can drive the costs high and could lead to delay in programs in the near future. Taking into consideration the launch phase, small satellites do not fly necessarily in “small-rockets” such as Pegasus. However, current big launch vehicles (Ariane, Delta, etc.), offer opportunities to launch satellites in stack/series configuration to reduce launch cost per satellite. They also offer standard launch interfaces for this purpose. However, this development is based around small to medium size satellites. This opportunity can be easily modified towards nano-satellites to launch tens or hundreds of satellites in a single launcher. In order to meet this launch approach, it requires modifications to current launch interface and orbit insertion procedures. Further this requires on-board propulsion for each satellite or a ‘mother-ship’ to insert these

satellites to its correct orbit. However such a development requires a shift from the current 'piggyback' to a paid customer. Therefore choosing the right technology and marketing method (a commercial approach) is an important factor for the success in small to very small (nano-satellite) satellite development.

Research is an important if not a vital part of human evolution. Our search for the existence of new possibilities will never come to a stop. New ideas need new platforms. Today, with limited resources available, reduction of costs for a given performance is a major goal for the future space activities. Scientific research satellites are developed under limited resources. In many cases they are national or international joint ventures such as ESA. For small research communities, such as those from various universities and research institutions, costs for space-related activities are far too high compared to their research allocated budget proportions. It is quite common that many scientific payloads are waiting from months to years to find a suitable launcher and a platform. The commercial sector is mainly interested in return for investment. They are reluctant to invest in new technology development as long as the risk and costs are high.

Launch and operations are two main areas that need careful consideration in order to accomplish a successful mission. Most of the current launchers are capable of lifting multiple satellites and put into LEO or GTO orbits. Standard adapters are available from the launch provider itself for multiple satellite launch. So far secondary or piggyback launch opportunities are readily available from different launch vehicle providers. For European small satellite projects, it may be advantageous to optimise the design for ASAP adapter as 'Ariane-Space' promotes European payloads. The Ariane-5 type launcher has the capability to place 6800 kg single or 5970 kg double payload into 7°, GTO orbit (or 10,000 kg into 800 km 98.6° Sun-synchronous orbit) (ESA Launchers Web Site, 2001). As Ariane-4 & 5 launch into orbit quite often, it may give frequent launch opportunity by selecting this launcher. This launcher has different platforms to optimise its payload capacity depending on the mass and volume of it. The auxiliary platform ASAP carries mini- or micro-satellites as secondary payloads. It can be mounted on top of the upper stage, as well as on the 'Speltra' or 'Sylda' structures.

In addition to the above, Nano-satellite missions need to consider other launch providers and must investigate different launch configurations such as AMPTE type (Ward and Gardner, 1994) model. NASA space shuttle offers services to launch small satellites, which can be an attractive option for educational institutions. However due to rigorous safety requirements of shuttle launches and long waiting times at present adds considerably to cost. It is also important to select a standard launcher and orbit insertion method if required to launch tens or hundreds of nano-satellites. This approach will provide the scientific community a low cost, frequent launch mission option compared to today's 'piggy-back' configuration. It is true that 'piggy back' provides low cost launch option. However these 'piggy back' rides do not always insert the satellite where it is designed to be and may not be opened for commercial customers. Further piggyback launch requires an orbit correction propulsion system on-board the satellite. However one needs to investigate all the available options. Flight opportunities on Long March, Proton, H-2 and Pegasus may be viable launch options to consider.

Small satellite programmes are intended to have short implementation timetables and low cost approach. This requires careful consideration of the operational nature of the project. In recent years, small satellite constellations are becoming increasingly interested among the mission planners. However the mission complexity may grow when taking into account the size of the constellation. The desire to solve tasks with limited resources requires simple solutions. It has been identified that reusability is vitally important for the ultimate success of a small satellite mission (Willoughby, 1994). Reusability is not limited to design and manufacture but also for the operation and management too. Avoiding reinventing and revising programs for each mission can bring substantial cost savings. The Swedish Space Corporation small satellite projects can be used as a model in this respect. Their 'ODIN' satellite project used the Internet to distribute data and scientists were able to send commands directly to the 'FREJA' satellite (Grahns, 2003). Although, this is not new today, it is important to share the available resources across multiple organisations and locations. In order to meet small satellite goals, planning and scheduling long into the future are required. Importantly, sophisticated instruments and multi task missions may fall outside the scope of nano-satellite projects (Ovchinnikov, 1994).

Based on electronic technology, miniaturisation and mass production are key issues for a breakthrough in space system development. High levels of miniaturisation may be achieved by applying micro/nano technologies. But this technology is ideal in cases where very large production volumes are available (tens of thousands of units). Most European countries have the capability and the required technology in micro-system research. As an example on its commitment to develop micro systems and related technologies, EU had planned to invest 100-million ECU in the 'Fourth Framework Programme' (De Aragon and A. Martinez, 1996). Any reduction in mass, volume and power consumption is desirable and will have significant cost reduction in space programmes. However with the current satellite demand mass production in line with micro/nano technologies is still an elusive idea.

2.2 Orbital Perturbations

A satellite in an Earth orbit deviates from its desired orbit due to the external forces acting on it. The external forces on a spacecraft are all forces except the spherically symmetric Earth gravitational pull. In a conservative force field, in this case the spherically symmetric gravitational pull, the orbital energy remains constant. Therefore the classical orbital elements, except the true anomaly, will be constant for a given orbit. When external forces act on a satellite, one or more of the classical orbital elements vary with time. The magnitude of these external forces varies according to the size of the satellite and type of orbit. Some perturbing effects are short term, cyclic within one-orbit period, and others are long term, accumulating with time. An earth orbit is perturbed after a certain period of time mainly due to the following perturbation sources.

- Air friction.
- Earth's Oblateness: J2, J3, J4 terms.
- Earth Magnetic field.
- Attraction forces from the Sun and Moon and other celestial bodies.
- Solar radiation pressure and Solar wind.

- On-board thruster misalignments (ΔV maneuvers).

These perturbing forces not only affect the orbit but also change the satellite attitude. As an example, for a given mission it can be of interest to point the solar arrays towards Sun for optimal power generation, while communication antennas are pointed towards the Earth. Generally, these types of attitude requirements are common in many space missions and demand certain pointing accuracy. These pointing requirements can be achieved by either passive or active means. Commonly both methods are used to gain higher accuracy. Certain perturbing effects dominate the long-term evolution of some orbital elements, mainly:

- The mean longitude drift rate variation due to Earth non-spherical effect.
- Change in eccentricity vector due to solar radiation and sun/moon attraction.
- The inclination vector drift is dominated by the attraction of the sun and moon.

As shown in Table 2-1, at altitudes less than 1000 km the relative magnitude of perturbing effects are considerable and needs to be corrected. This is especially true for a longer duration mission. With increase altitude atmospheric drag reduces but solar radiation pressure and third body affects become dominant. Generally many small satellite missions operate in low altitude polar orbits, to give full Earth coverage and to achieve high resolution of ground images. Depending on the area of application and mission lifetime the adopted correction manoeuvres may vary. For many low Earth orbiting satellites, at the end of a mission, the perturbation forces can be used to de-orbit a given satellite effectively. Depending on the applied de-orbiting method, it may require an operating attitude control system to maximise the perturbation forces for efficient de-orbiting.

Disturbance Source	Altitude (h) 400 km	Altitude (h) 1000 km	Altitude (h) 1500 km
Circular orbital velocity, V (km/s) $V^2 = 398600/(6378+h)$	7.67	7.35	7.11
Mean atmospheric density (kg/m^3) (Wertz and Larson W. J., 1999)	2.72×10^{-12}	2.79×10^{-15}	5.21×10^{-16}
Air drag deceleration (m/s^2) (Ballistic coefficient, $BC = 55 \text{ kg/m}^2$)	1.5×10^{-6}	1.4×10^{-9}	2.4×10^{-10}
Solar radiation deceleration, (m/s^2) (Area/mass = $0.009 \text{ m}^2/\text{kg}$)	8.2×10^{-8}	8.2×10^{-8}	8.2×10^{-8}
Lunar attraction correction, $ \Delta a_{Moon} $ (m/s^2)	1.2×10^{-6}	1.3×10^{-6}	1.40×10^{-6}
Solar attraction correction, $ \Delta a_{Sun} $ (m/s^2)	5.4×10^{-7}	5.8×10^{-7}	6.20×10^{-7}
Gravity gradient torque, (N-m) Moment of Inertia, $I_{zz} = 0.0308 \text{ kg-m}^2$ Moment of Inertia, $I_{yy} = 0.0256 \text{ kg-m}^2$; $\theta = 30\text{deg}$	1.83×10^{-10}	1.42×10^{-10}	1.16×10^{-10}
Magnetic field torque, (N-m) Dipole moment, $D = 1 \text{ A-m}^2$	5.11×10^{-5}	3.94×10^{-5}	3.26×10^{-5}

Table 2-1 Accelerations and torque's on a typical LEO nano-satellite (Derived using STK and ESA/ADS tools).

Where,

$$\text{Ballistic coefficient, } BC = \frac{m}{C_D A}$$

For a typical nano-satellite type Mustang:

Mass, $m = 10 \text{ kg}$

Frontal area, $A = 0.3 \times 0.3 \text{ m}^2$

Drag coefficient, $C_D = 2.0$

Air Drag

As the air drag is a non-conservative force, it takes out the energy from the orbit. Therefore drag caused by the Earth's atmosphere, spirals satellites downward. As they spiral downward, they speed up and the friction effect is further enhanced. The associated change in orbital elements (mainly semi-major axis) simply tells us the rate at which the 'Orbital Motion' is changing due to drag or other related effects. It is typically a very small number. Common values for Low-earth-orbiting satellite decelerations are in the order of 10^{-4} m/s^2 , and for high-orbiting satellites are in the order of 10^{-7} m/s^2 or smaller. Drag is a retarding force due to atmospheric friction and is in the direction opposite to the spacecraft velocity vector. In an elliptical orbit, drag is most important at perigee because the density of the Earth's atmosphere, to which the drag is proportional, decreases exponentially with altitude.

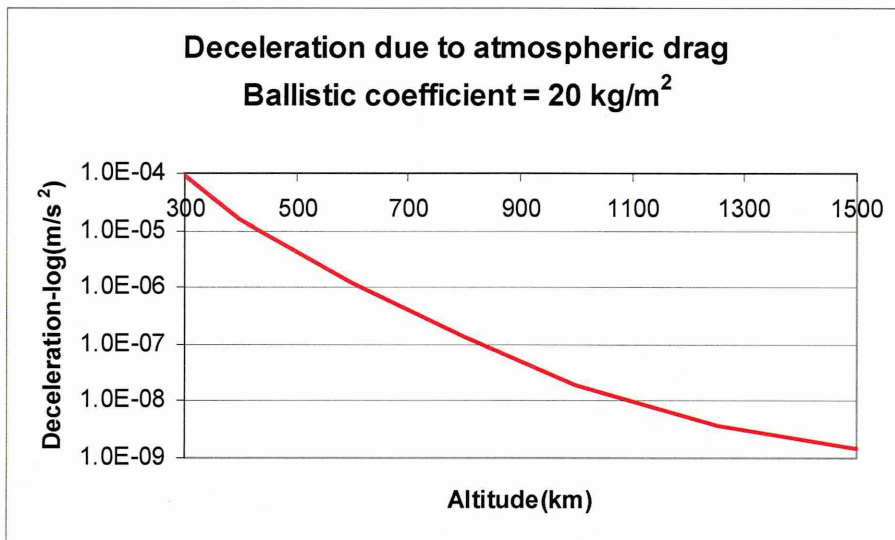


Figure 2-1 Deceleration caused by Earth low atmosphere on a typical nano-satellite.

The ratio $m/C_D A$, is defined as the vehicle 'ballistic coefficient' and varies around 20 ~ 100 for nano class satellites. The aerodynamic drag coefficient C_D , is independent of the size of the satellite but is a function of Reynolds number and the shape of the object. For small satellites this is around 2.0 ~ 2.6, which provides a good starting value for initial design calculations. During the initial stages of the mission, when the satellite is operating from transfer orbit to its final orbit, C_D changes somewhat within the above range. The main reason for this is due to the Reynolds number changes as the satellite moves from one orbit to the other. The drag deceleration is not a constant

as the air density is changing rapidly in Equation 2-1 due to variation in orbital altitude and seasonal effects (temperature variations). However for short period (close to orbital period) analysis the drag deceleration can be considered as constant. Importantly altitudes over 1500 km, air drag effects are insignificant for many applications. The air drag deceleration, a_d , is calculated as shown in Equation 2-1.

$$a_d = -\frac{1}{2} \rho \left(\frac{C_D A}{m} \right) V^2$$

Equation 2-1

m = Mass of the satellite.

A = Satellite projected area respect to the velocity vector.

C_D = Drag coefficient

ρ = Air density

V = Velocity

A satellite altitude does not remain constant throughout its orbit and during the mission time. Unless the satellite is fully symmetrical, its projected cross sectional area perpendicular to the velocity vector may vary with time due to attitude changes. If the satellite is in a highly elliptical orbit, significant atmospheric density changes appear during the orbit too. In addition, Equation 2-1 requires an accurate density model for precise calculations. The value of aerodynamic drag acceleration, a_d , in Equation 2-1 reduces when the non-spherical shape of the earth and its 'diurnal bulge' effects are considered (Hechler and Eckstein, 1970).

For a circular low Earth orbit, air resistance is the most significant drag force. As it reduces the orbital altitude, it needs to be corrected from time to time. However if more precise orbit corrections are required, continuous correction manures can be used (eg. GOCE spacecraft). In order to obtain the original orbit, a correction manoeuvre is required at the correct segment of the orbit. The corresponding ' ΔV ' requirement for a typical nano-satellite is plotted in Figure 2-2 taking the ballistic coefficient 20 kg/m². This shows that ΔV demands could be met with low thrust electric propulsion at altitudes not less than 600 km.

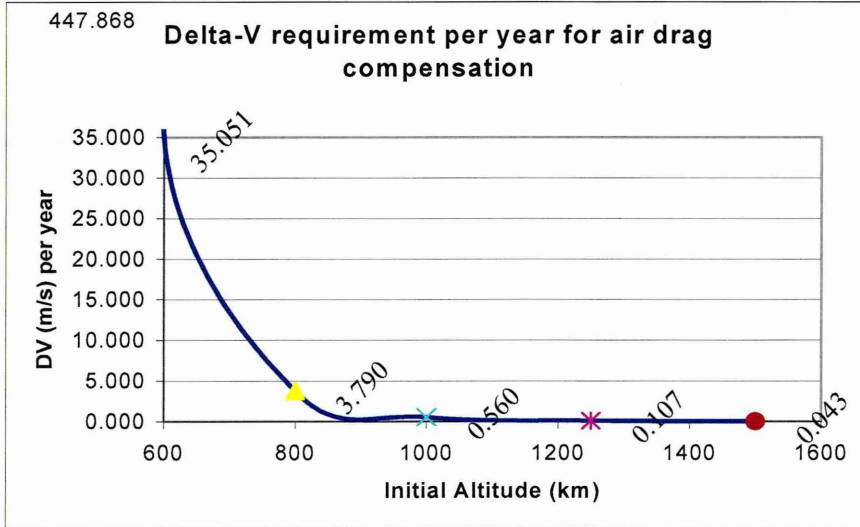


Figure 2-2 Typical nano-satellite delta-V requirement per year (ballistic coefficient: 20 kg/m²).

Solar Radiation Pressure

The electromagnetic radiation pressure exerts a force on the spacecraft proportional to its effective area, reflectivity coefficient of the surface material and the Sun angle. The orbit and attitude of the satellite may change due to this perturbing force and its effect is periodic. The solar pressure constant value may vary slightly (< 1%) as a result of solar activity changes. For small satellite missions requiring near circular orbits, deployed solar arrays will increase the perturbing force due to increase area. The exerted force on the satellite is calculated according to Equation 2-2.

$$F = PA(1 + \varepsilon)\cos(i)$$

Equation 2-2

Where,

F: Solar radiation pressure force

P: Solar pressure, (near the Earth, $P \approx 4.56 \times 10^{-6} \text{ N/m}^2$)

A: Projected cross sectional area perpendicular to the incoming sunrays.

ε : Surface reflectivity coefficient, ($0 < \varepsilon < 1$)

i: Sun angle (= angle between the Sun and the normal to the surface)

As the satellite moves towards the sun, solar radiation pressure forces tend to decelerate its orbital motion. The magnitude of deceleration is roughly estimated according to Equation 2-3. When the spacecraft moves away from the Sun, the solar radiation pressure forces accelerate it. Though it tends to reduce the circularity (semi-major axis), for many orbits it has a periodic variation and hence its net effect is zero.

$$a_p = \frac{F}{m} = \frac{PA}{m}(1 + \varepsilon)$$

Equation 2-3

Where,

a_p : deceleration (or acceleration) due to solar radiation pressure

Cross-section to mass ratio (A/m), of a typical small satellite ranges from 0.01 to 0.1 m^2/kg with current technology. The associated perturbation acceleration is typically in the order of $10^{-7} m/s^2$ and is switched off when the satellite passes through the shadow of Earth or moon. Accurate modelling of the solar radiation acceleration is difficult due to following reasons:

- Different surfaces have varying reflectivity and cross section and exposed to the sunlight at different parts of the orbit.
- Surface properties change due to out-gassing and ageing.

The solar radiation pressure effect is significant for large and medium satellites as they are customarily built with large solar arrays. For those missions that are intended to use a sun-synchronous orbit require thermal blanket to safeguard the spacecraft from overheating. These blankets usually have a high reflectivity coefficient, which increases the solar perturbation effect.

In some circumstances the solar radiation pressure effect has been used to navigate the spacecraft. This concept is called 'solar sailing' and is used in interplanetary (obviously in the vicinity of Sun) and earth orbiting satellites. However due to the technical complexity, this concept is not attractive for small satellite concepts. The major limitation with respect to small satellites in this application area is, that the required cross sectional area to generate enough forces are too large and the mechanisms needed to direct the solar pressure forces to the sailing direction are far too complicated for small satellites.

Regression of Nodes

The Earth is not a perfect sphere and most of its mass is concentrated around the equator. The polar and equatorial diameters are around 12713 km and 12756.3 km, respectively. This causes a component of gravitational force to be out of the orbit plane. This result the spacecraft Right Ascension of the Ascending Node (RAAN) to rotate (node rotation). This orbital nodal rotation is called as regression of nodes and its rate of change is called the nodal regression rate. The RAAN motion is westward for direct orbits (inclination $< 90^\circ$), eastward for retrograde orbits (inclination $> 90^\circ$), and zero for polar orbits (inclination $= 90^\circ$). The out of plane force f_z rotates the orbit plane such that the rate of change of the longitude of the ascending node or the so called nodal regression rate is (Sidi, 1997):

$$\frac{d\Omega}{dt} = \frac{r \sin \nu}{h \sin i} f_z$$

Equation 2-4

The component f_z of the perturbing force, which acts in the direction normal to the satellite orbit plane, is calculated according to Equation 2-5.

$$f_z = -\frac{3\mu_e J_2 R_e^2}{r^4} \sin i \cos i \sin \nu$$

Equation 2-5

Inserting Equation 2-5 in Equation 2-4 gives Equation 2-6.

$$\frac{d\Omega}{dt} = -\frac{3\mu_e J_2 R_e^2}{hr^3} \cos i \sin^2 \nu$$

Equation 2-6

Averaging Equation 2-6 with respect to ν over the satellite orbital period (Brown, 1998) results Equation 2-7.

$$\frac{d\Omega}{dt} = -\frac{3nJ_2 R_e^2}{2a^2(1-e^2)^2} \cos i$$

Equation 2-7

For a circular orbit eccentricity, $e = 0$ & semi-major axis, $a =$ orbit radius, r . Applying these in Equation 2-7 result Equation 2-8.

$$\frac{d\Omega}{dt} = -\frac{3nJ_2 R_e^2}{2r^2} \cos i$$

Equation 2-8

For low Earth orbiting satellites, according to Equation 2-8, the regression of nodes is minimised for high altitude polar orbits (inclination around 90°) and has its maximum for equatorial orbits (inclination $= 0^\circ$).

Where;

Ω = Right ascension of the ascending node (RAAN)

ν = True anomaly (= varying angle)

e = Eccentricity

R_e = Mean equatorial radius of the central body. (Ex. earth radius = 6378.2 km)

i = Inclination of the orbit

n = Mean motion

J_2 = zonal coefficient, (Earth orbits, $J_2 = 0.00108263$)

r = Orbit radius

The largest zonal harmonic coefficient J_2 takes into account the Earth equatorial bulge mass distribution (longitudinal Earth mass distribution). However in this discussion we have neglected the higher order zonal harmonic coefficients and must take into account for accurate orbit prediction. Although the orbital nodal regression is discussed here as a perturbing action, this nodal rotation has been used in many missions to get the correct Sun angle especially for remote sensing applications.

In a Sun-synchronous orbit the secular variations in the right ascension of the ascending node to the Earth's rotation rate around the Sun is matched. Since the mean Sun moves uniformly eastward along the celestial equator through 360° within a year (365.242 days), a nodal precession rate of $0.9856^\circ/\text{day}$ ($=360/365.242$) will match the Earth's average rotation rate about the Sun. Sun-synchronous inclination,

i_{ss} , for a spacecraft at a given orbital altitude h is calculated as of Equation 2-9 and is always greater than 90 deg (retrograde orbit) (Wiesel, 1989).

$$i_{ss} = \cos^{-1} \left\{ -0.098922(1 - e^2)^2 \left(1 + \frac{h}{R_e} \right)^{3.5} \right\}$$

Equation 2-9

For typical sun-synchronous orbits such as ‘noon-midnight’ and ‘twilight’ or ‘dawn-dusk’, the spacecraft orbit relative to the sun-earth plane is shown in Figure 2-3. In such an orbit the solar panels are designed to receive maximum energy flux by projecting perpendicular to the solar vector. As the solar radiation pressure distribution varies along the illuminated surface, according to the type of surface and satellite orientation, the resultant force may act in a location (centre of pressure) away from the spacecraft mass centre.

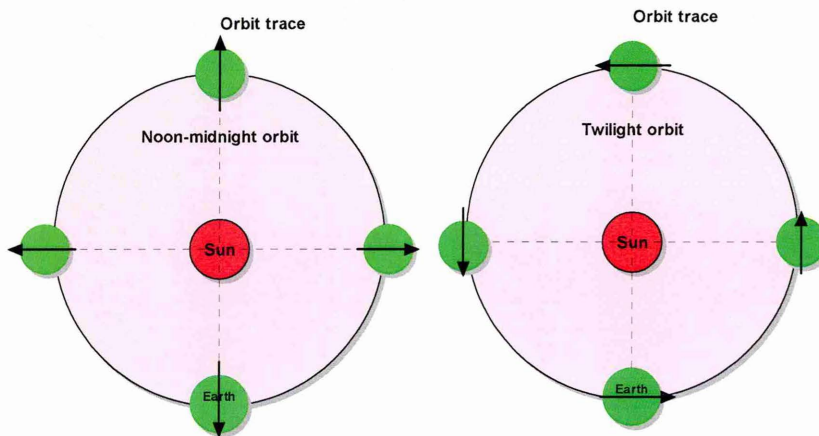


Figure 2-3 Sun-synchronous orbits: “Noon-Midnight” to left and “Twilight” orbit to right.

Remote sensing observations such as reconnaissance and Earth resources missions that uses optical sensors takes the advantage of ‘noon-midnight’ orbit as it maintains a constant Sun angle between the orbit plane and the Sun (use to measure shadows). The main disadvantage in this orbit is that the spacecraft goes into eclipse on every orbit. The science and radar remote sensing missions use the advantage of ‘Twilight’ or ‘Dawn-Dusk’ orbit. The main advantage in this orbit is that the solar arrays can generate continuous power for much of the orbit. The spacecraft solar arrays can be arranged perpendicular to Sun to generate maximum power while it always aligned along the velocity vector, giving minimum drag. LANDSAT-D, which is placed in a 709 km altitude orbit with a 98.2 deg inclination, is a good example of sun-synchronous Earth observation application.

Rotation of Apsides

This is similar to regression of nodes and is an orbit perturbation due to the Earth's equatorial bulge. Rotation of apsides or the perigee rotation is caused due to greater than normal acceleration in the equatorial plane caused by the gravitational force from the extra ring of mass along the equator (equatorial bulge). Thus each successive apogee and perigee will occur farther around than formerly. The line of nodes (line joining the perigee and apogee) rotates in the direction of satellite's motion for inclinations less than 63.4° or greater than 116.6° . It rotates opposite to the spacecraft's motion for inclination between 63.4° and 116.6° .

For a satellite over either of the Earth's poles, the distance to the Earth's equator is greater than the distance to the centre of the Earth. Therefore gravitational force due to Earth's bulging is less significant for polar orbiting satellites. In a polar orbit a satellite spends part of the time over the equator where the gravitational force larger than the mean and part of the time over the poles where the mean force is less. It can be shown that for polar orbits the net effect is rotation of the line of apsides opposite the direction of motion, although the rate of rotation is less than for equatorial orbits (Wertz, 1980). Nodal regression for circular orbits does not have any valid meaning, as perigee is undefined for circular orbits. Nodal regression rates are significant for low inclination, low altitude orbits.

2.3 Attitude Perturbations

The perturbing forces discussed above as well as others such as two-body attraction from lunar-solar gravitation provides only an approximate model. Their accuracy decreases as the time of propagation increases. All these perturbations cause changes in orbital elements (all six Keplerian elements) of the affected spacecraft. These effects are generally classified as secular, short period or long period. Secular variations represent a continuous monotonic change with time. Short-period variations are cyclic changes in the orbital elements with a period less than or equal to one orbit period. Long-period variations are the same as short-period ones but with periods greater than one orbit.

Unsymmetrical distribution of external forces acting on a spacecraft causes torques around its principal axes. An estimation of these external and internal torques is important for design and operation of a spacecraft AOCS system. A highly simplified estimation was made taking into account only the external forces acting on a simple rectangular spacecraft as shown in Figure 2-4 of size 300x300x200 mm. The corresponding disturbance sources, its influence and the estimations are summarised in Table 2-2.

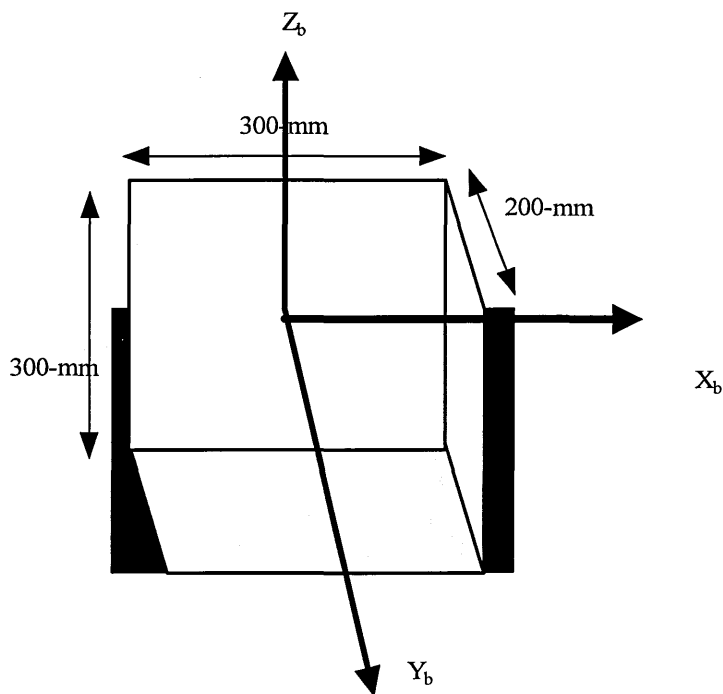


Figure 2-4 Simplified Mustang nano-satellite model used for hand calculations.

Table 2-2 shows that from all disturbance torques, The Earth magnetic field related torque have the maximum effect, if the spacecraft has a residual magnetic dipole D . This disturbance torque is cyclic and has the greatest influence on low, polar orbiting satellites. For spin-stabilised or momentum bias systems, the cyclic torques will cause periodic variations in attitude, while secular torques cause gradual divergence. Magnetic torque rods use this available Earth magnetic field, with a generated electrical current through a coil to compensate spacecraft residual magnetic fields or attitude drift from minor disturbance torques. The satellite attitude and orbit control system is sized to meet the mission requirements while maintaining the disturbance torques under satisfactory limits.

Disturbance	Influenced Primarily by	Estimation
Gravity gradient	<ul style="list-style-type: none"> Spacecraft inertias Orbit altitude 	$T_g = \frac{3\mu}{2r^3} I_z - I_y \sin(2\theta)$ $T_g = 1.88 \times 10^{-9} \text{ N.m}$
Solar radiation	<ul style="list-style-type: none"> Spacecraft geometry. Spacecraft face reflectivity Spacecraft cg location 	$T_{sp} = F(c_{ps} - c_g)$ $c_{ps} - c_g = \left(0.3/4\right) = 0.075\text{m}$ $T_{sp} = 7.0 \times 10^{-8} \text{ N.m}$
Magnetic field	<ul style="list-style-type: none"> Orbit altitude Residual spacecraft magnetic dipole, D. Orbit inclination 	$T_m = DB$ $B_{polar-orbit} \approx 2M/r^3 = 4.3 \times 10^{-5} \text{ Tesla}$ $T_m = 4.3 \times 10^{-5} \text{ N.m}$
Air resistance	<ul style="list-style-type: none"> Orbit altitude Spacecraft geometry and cg location 	$T_a = F(c_{pa} - c_g)$ $c_{pa} - c_g = 0.075\text{m.}$ $T_a = 4.85 \times 10^{-8} \text{ N.m}$

Table 2-2 Worst-case external disturbance torque effects on a spacecraft as in Figure 2-4 [Table 11-9a: (Wertz and Larson W. J., 1999)].

Where:

- T_g : Gravity gradient torque (N.m)
- T_{sp} : Solar radiation torque (N.m)
- T_m : Magnetic torque (N.m)
- T_a : Air resistance torque (N.m)
- I_x, I_y, I_z : Moment of inertia along the X, Y and Z axis respectively (Kg.m²)
- C_{ps} : Pressure centre from a given reference (m)
- C_g : Mass centre from a given reference (m)
- C_{pa} : Aerodynamic centre from a given centre (m)
- F : Solar radiation or Air drag force (N)
- θ : Angle between the Earth centre to S/C centre vector and the S/C z-axis (deg)
- B : Earth magnetic field strength (tesla)
- M : Earth's magnetic moment (approximately 7.99×10^{15} tesla.m³)
- r : Distance to Earth centre

2.4 Gravity Gradient Torques

The gravity gradient torques are caused due to the finite dimension of the spacecraft. These torques results from the difference in gravitational forces, mainly if the spacecraft is an elongated body. As the net result of this torque is to orient the spacecraft elongated axis toward the Earth centre, many spacecrafts uses this as a passive attitude control system. However in order to benefit this advantage the spacecraft should be a long elongated body and in a low Earth orbit. Although nano-satellite applications are mainly in the Low Earth orbits, the elongated (long rod type) shape may fall outside the nano-satellites design criteria. However by using a

telescopic type model, the nano-satellites can use the full advantage of launcher stacking and orbit transfer, while in orbit to use gravity gradient torques for attitude stabilisation.

3 Electric Propulsion

The conventional electric propulsion fundamentals are discussed in this section. Special emphasis is given to identify the parallel research work in electric propulsion with respect to small satellites. The current technology statuses of these thrusters are highlighted, together with the viable areas of application. Whenever appropriate, thruster propulsion capability with respect to nano-satellites are discussed in brief. FEEP thrusters are not taken into account here, as this is addressed in more detail, separately.

3.1 Historical Background

Space vehicle propulsion by means of electrically expelled particles, was probably first stated by R. H. Goddard in 1906. H. Oberth discussed electrical propulsion at some length in 1929, and in 1946, J. Ackeret provided theoretical details of rocket propulsion, which apply to electrical systems. In 1947, Seifert, Mills, and Summerfield mentioned the necessity of ejecting positive and negative particles simultaneously. The first of a more elaborate study in electrical propulsion appeared in the early '50s, notably by Spitzer. A comprehensive study of electrically propelled space vehicles, capable of manned expeditions to Mars was published by Stuhlinger in 1954 and expanded in the following years. In the beginning of 1956 Langmuir, Irving, Bussard and Boden contributed essential data to the theoretical and experimental knowledge of electrical propulsion systems (Koelle and Koelle, 1961).

Although electric propulsion has been considered for space applications since 1950s it has only begun to make widespread impact since the mid 1990s. During the last three decades more than 160 satellites using some form of electric propulsion were deployed in Earth-orbit and interplanetary missions (Cassady R. J., 2001). While some of them were experimental, others provided drag makeup, attitude control and station keeping or orbit adjustments (Sovey et al. 1992). USA, former Soviet Union, Europe, Japan and Peoples Republic of China are the main contributors to the development of electric propulsion for space application. Use of electric propulsion (EP) for commercial telecommunication satellites continues to grow worldwide. The ESA SMART-1 mission that uses EP for orbit-raising will further widen this area of EP applications in Earth orbit as well as interplanetary mission design.

3.2 Electric Propulsion for Small satellites

Since the 1960s electric propulsion has been used successfully in almost all categories of satellites (Pollard et al. 1993). The areas of application have been broad, from sun pointing to attitude and orbit control. Initially, electrical thrusters were developed as a secondary propulsion system to provide attitude control and station keeping, mainly for geostationary

satellites. However, with formation-flying missions and higher demands on control accuracy, the current trend is to develop it as a primary propulsion system. Space-borne ‘virtual platform’ missions such as LISA (Danzmann et al. 1995) and Space Technology-3 (ST3) (Lau et al. 1997) are two such examples that demand an accurate propulsion system. Thus, this requires a group of satellites, propagating in their respective orbits keeping their relative distances and angles under satisfactory limits. To accomplish these ‘formation-flying’ missions with high precision requires a propulsion system with very good control accuracy (for LISA mission angular position accuracy to about 1 arc minute).

Depending on the nature of energy extraction, electric propulsion could be classified into the following three categories.

- Electrothermal – propellant (fluid) is heated by an electric arc discharged through the fluid (arcjet), or by electrical heating through a wall/resistant (resistojet).
- Electromagnetic – electromagnetic pressure forces due to the electric and magnetic fields acting within plasma.
- Electrostatic – discretely charged particles accelerated by electrostatic forces.

In electrostatic and electromagnetic thrusters, the propellant is first ionised or charged and accelerated from moderate to very high velocities in order to gain the required thrust. Electrothermal thrusters use the traditional principles of thermodynamic expansion, to accelerate the electrically heated propellant. All types of electrical thrusters need an electrical power plant to generate the required energy. However, all satellite subsystems require electrical power. These subsystems may operate simultaneously or at different times depending on the demand. As the satellite power plant must guarantee continuous power supply to different subsystems, it requires pre-knowledge of their power requirements during the entire mission.

The size and mass of the power plant is a crucial design parameter for any satellite and this is especially true for electrically powered nano-satellites. As the available mass and size is limited in this category of satellites (Nano or Pico satellites), the design engineer must take into account all the subsystems during the preliminary design study. This shows that integration of all subsystems in a satisfactory manner is an important part in the development of small satellites. The complexity of these subsystems may vary according to the mission. As shown in Figure 3-1 there are many different subsystems to be included, irrespective of the satellite size and mass. Therefore the capacity of the satellite’s power system will determine the performance of the propulsion system. On the other hand, propulsion system operational frequency will affect the performance of other subsystems. Generally, higher the propulsion system power requirement, greater will be the power system mass. Other important parameters are the alternative power supply methods during peak power requirements, eclipse time, charging and discharging time of the batteries/capacitors etc.

High specific impulse values are attractive from the point of propellant mass but require large power plants to meet the extra power demand. This in turn increases the mass of the

total system. Therefore, when selecting thrusters with high specific impulse, care should be given for power plant mass. Due to the low thrust character of EP systems, propulsion times are considerably long. When the thruster performance deteriorates with time, it requires much longer propulsion time to perform a given manoeuvre. This is especially true at the end of the mission where the system had been degraded to some extent.

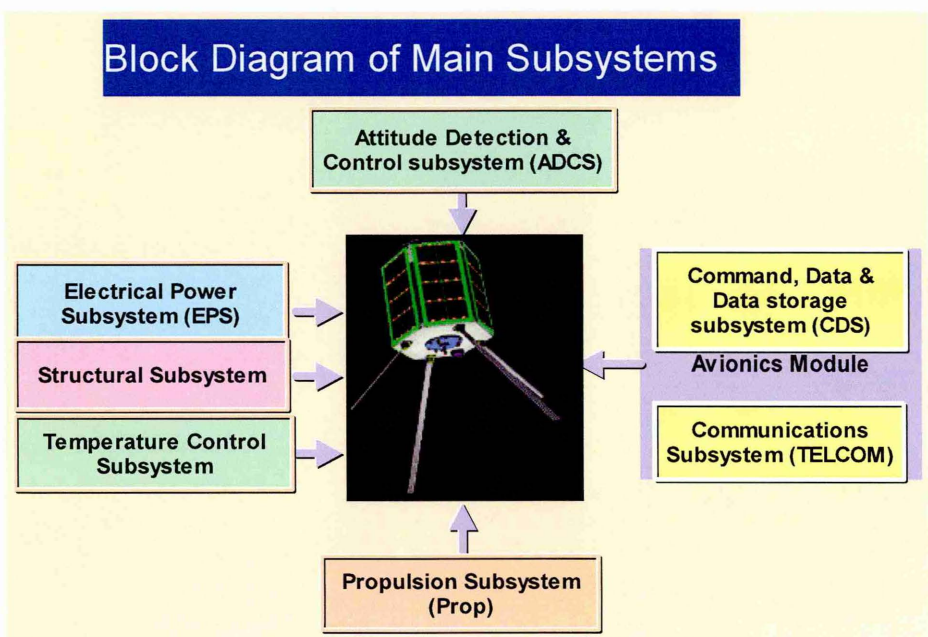


Figure 3-1 Main satellite sub-systems in a block format without payload.

It is quite common for very small satellites to have a long storage period before it finds a suitable launcher. If it is planned to launch as a piggyback payload, the launch opportunities may arise within short notice. Therefore small satellites should be built to perform ground tests with easy access to different subsystems. However as some electric propulsion systems complex in terms of testing and verification, it should be able to pre-test separately (usually in ground chambers) prior to its last minute integration to the satellite.

3.3 General Performance

An electrical propulsion system requires a separate power source, and a suitable power conversion plant, to impart the necessary energy to the propellant particles. Power source and conversion plant remains as an integral part to the propulsion system and adds considerably to the mass of the vehicle. The requirement of a separate power source and conversion plant creates a practical limitation to the rate at which energy can be imparted to the propellant. Due to this, electric propulsion systems have a comparatively low mass-flow rate. This results in characteristically low thrust and hence low accelerations (extended time periods for a given velocity change). Based on the 1960's technology on electrical

thrusters, the maximum acceleration is limited to around $5 \times 10^{-3} \text{ m/s}^2$ {Chapter 3, (Stuhlinger, 1964)}. This figure has improved by a factor of 4 with present technology and with the development of power generation and conversion units together with advance materials it may increase by a factor of 6 or 8 in the near future within the laboratory research experiments. These assumptions were made taking power-to-weight ratio as figure of merit and considering the relative development in the past. Satellites designed prior to 1980's had a specific power of 0.4-1.4 W/kg and in 1990's this value rose to 1.4-3.5W/kg (Pollard et al. 1993). The instantaneous acceleration (depends on S/C mass and thruster input power at a given time), which is different from maximum continuous acceleration, is proportional to the availability of electrical power for propulsion. The magnitude of acceleration, a , for a satellite, which has a mass M and electrical power input P to the thruster, is given by:

$$a = \frac{2}{g_e} \frac{\eta}{I_{sp}} \frac{P}{M}$$

Equation 3-1

Where,

a : satellite acceleration or the rate of change of velocity within a given time period

P : Input power from the power plant

M : Total mass of the satellite

I_{sp} : Thruster specific impulse

g_e : Earth gravitational constant at its surface (9.81 m/s^2)

η : Overall efficiency of the system taking into account the thruster and the power plant ($\eta = \text{Thrusters power output} / \text{Power plant input power}$)

By comparing Equation 3-1 with the S/C mean acceleration derived from thrust to mass ratio (F/M), it is evident that $(2\eta/g_e I_{sp})$ is the thrust-to-power ratio. Therefore for constant input power and overall efficiency η , as the specific impulse I_{sp} increases (increase in mass expulsion velocity), thrust levels drops. Electric propulsion with low accelerations requires much longer duration of 'thruster-operation' time to achieve a given velocity change (ΔV). Thus the effectiveness of thruster performance tends to reduce with higher specific impulse values. By knowing the thruster operational time, Equation 3-1 gives the maximum range of specific impulse. The selection of the right propulsion system is crucial and needs lot of consideration. There are a wide range of EP systems and their capabilities of performance are summarised in Table 3-1 for small to medium-sized spacecraft applications (eg. Ion thrusters have operated at $> 10 \text{ kW}$, but are not included below).

	Electrothermal		Electro-magnetic	Electrostatic		
	Resistojet	Arcjet	PPT (Pulsed Plasma Thruster)	SPT (Stationary Plasma Thruster)	Ion Thruster	FEET ¹
Propellant (Typical)	N2H4	N2H4	Teflon	Xe	Xe	Caesium
Input electric power (kW)	0.2 – 0.8	0.5 – 2.0	0.001- 0.025	0.15 – 1.5	0.4 – 2.0	0.00006 - 0.12
Thrust (mN)	200 - 800	200 - 250	0.01 - 1	40 - 200	15 - 200	0.001 - 5
Mission average Isp (s)	200 – 300	450 – 600	300 - 1000	1600	2800 - 3500	5000 - 10000
Overall Efficiency (η) %	80	35	20	48	60 - 70	80
Lifetime	>390 hrs	>830 hrs	> 10 ⁶ cycles	>6,000 hrs	>15,000 hrs	>2000 hrs

Table 3-1 typical performance values for few electric thruster types (Filliben, 1999).

According to the thruster performance, EP systems can be categorized into high and low acceleration thrusters. Most of the ‘resistojets’ and ‘arcjets’ fall into the high acceleration category. Here, the average duration of a manoeuvre is less than 1-hour per week, which is similar to traditional chemical thrusters. All electromagnetic and electrostatic thrusters fall under the low acceleration category. The manoeuvre duration is up to several hours per day and usually two symmetrical burns per orbit are required to cancel any anomalies. In order to reduce the required propellant load for a given mission, it is advantageous to use thrusters with high specific impulse. The Equation 3-2 gives the required propellant mass for a given manoeuvre, ΔV .

$$M_p = M_i \left[1 - \exp \left(- \frac{\Delta V}{g_e I_{sp}} \right) \right]$$

Equation 3-2

Where,
 M_p : Propellant mass
 M_i : Spacecraft initial mass prior to thrusters firing

¹ (Centrosazio Web Site, 1998)

ΔV : Velocity change due to thruster firing

I_{sp} : Thruster specific impulse

g_e : Earth surface gravitational acceleration ($= 9.81 \text{ m/s}^2$)

Thus for thrusters with a low specific impulse, the propellant requirement increases rapidly. This in turn renders a mass penalty on the systems due to the large propellant-tank and structures. Therefore from the propellant mass perspective, it can be advantageous to use relatively high specific impulse thrusters. However from the propulsion system selection perspective, the specific impulse is not the main important factor. Selection of a propulsion system requires careful trade-off studies. Besides the required performance of the thruster, it is important to consider the selected propulsion system interaction with the satellite and its different subsystems, the thruster behaviour during operational lifetime, amount of degradation at the end of life etcetera. For example thruster erosion, which is common for many electric propulsion systems, may cause long-term drift in the thrust vector direction. Moreover if any thrust misalignment is present, it may increase the uncertainty of the satellite mode of correcting the attitude. Other aspects that need to be taken into account depending on the mission are:

- Thruster erosion rate (life time)
- Material deposition on the S/C surface.
- Possible beam impact on the spacecraft needs to be avoided totally, due to erosion by sputtering.
- Thrust vector variation due to instabilities and movement of the beam.
- Thermal impact on the thruster/ spacecraft.
- Surface potential changes due to the plasma beam clamps spacecraft potential to space potential (an advantage).
- Electromagnetic effects due to the thruster on/off and steady-state operation.
- RF interface with the S/C antenna operation.

3.4 Electrothermal

This is similar in some aspects to the chemical propulsion. Generally in electro-thermal propulsion systems, the propellant gas is heated externally, without combustion, to a high temperature and expanded through a nozzle to produce thrust. The advantages here compared to chemical propulsion is that, the greater freedom to choose propellant type, gives favourable propulsion characteristics such as low molecular weight and high specific heat ratio, γ ($\gamma = c_p/c_v$, where, c_p : specific heat at constant pressure, c_v : specific heat at constant volume) than available from the combustion products. The electrical energy that could be added to the propellant may be larger than the energy available from combustion. However material limitations place a practical upper limit on the amount of energy added, which increases the propellant temperature.

Resistojet

Electrical energy is transferred to the propellant using an electrically heated surface. This is the simplest form of electrical thruster and has many similarities to the commonly used chemical propulsion. Different configurations of resistojets have been tested with many different propellants such as H_2 , CO_2 , CH_4 , N_2H_4 , NH_3 , N_2 , He, Ar, H_2O (in liquid and gas forms) as well as air were tested (Auweter-Kurtz, 1997). Exhaust velocities higher than $3,000\text{ ms}^{-1}$ ($I_{sp} = 300\text{ sec.}$, see Table 3-1) appears to be possible with tungsten heater elements and hydrogen as propellant. Due to its cryogenic nature and difficulties in handling, hydrogen is less of interest here. Although, earlier resistojets had disadvantage of a short mission time due to the erosion of the heater surface, recent tests had recorded more than 500,000 N-s total impulse (Pollard et al. 1993).

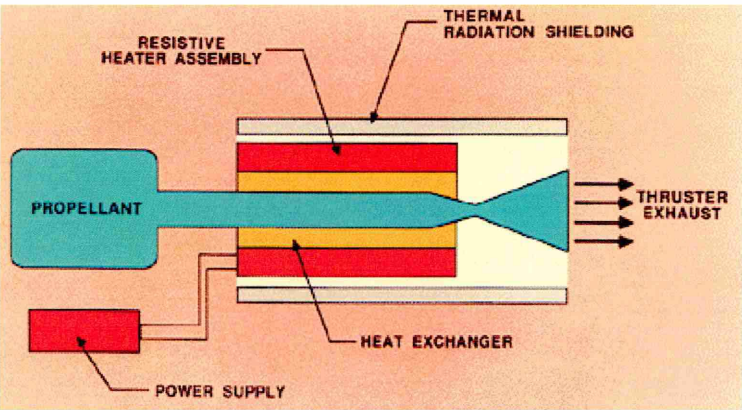


Figure 3-2 Resistojet thruster concept (Credit NASA/JPL).

Hydrazine resistojets with performance figures around 200-500 mN thrust and 0.6-0.8 kW input power levels have been used extensively since 1980's (Pollard et al. 1993). A power efficiency of about 50% was achieved with the current technology. These thrusters are commercially attractive, as it can be viewed as a simple design with 'off-the-shelf' technology. As a nano-satellite propulsion system, 'micro machined resistojets' provides a viable alternative. Janson had discussed a hypothetical 'micro-machined' 1-W ammonia resistojet (Janson, 1994) that could be of interest here. Few important aspects of this concept are given in Table 3-2, below.

Thrust produce	0.41 mN
Power consumption	1 W
Thruster efficiency	50%
Specific Impulse	250 s
Working fluid	Ammonia (N ₂ H ₄)
Temp	273K 400K 600K 800K 1000K
Isp	107s 130s 167s 203s 244s
Storage density	0.6 g/cm ³
Storage temperature	300 K
Storage pressure	1.06 MPa
Propellant flow rate at nominal thrust	1.7x10 ⁻⁷ kg/s

Table 3-2 Hypothetical “micro-machined”, 1W-ammonia resistojet performance characteristics (Janson, 1994).

The Arc Jet

The development studies of the ‘arc-jet’ were started in the early sixties in USA and around 1986 within ESA (Ghislanzoni et al. 1990). The studies of the Arc-Jet thruster were conducted in a power range of 1 kW to 30 kW. The NASA Lewis research centre (USA) and Rocket Research Company (RRC) have extensively tested EP performance. Within the framework of ESA, development of the ‘arc-jet’ was carried out in Italy (BPD Difesa e Spazio and University of Pisa) (Centrosazio Web Site, 1998) and at the University of Stuttgart. In the recent years the ‘arc-jet’ thrusters development is extended to much lower power levels using hollow cathode technology (few 10s of W to few 100 W).

In an arc-jet, the propellant is heated to a high temperature using a high current electrical discharge through the propellant and then expanded using a conventional rocket nozzle. It is advantageous to have the propellant temperatures under dissociation limits to extract maximum energy from the exhaust gas. In general, an arc-jet nozzle consists of a metallic material and serves a dual function, as an anode and an arc column constrictor, except for a divergent section.

The arc jet thruster anode, which serves as a plenum chamber, basically made out of materials with a high melting temperature such as pure Tungsten or Tungsten-Rhenium alloy. The cathode material is usually made of ‘Thoriated Tungsten’ and has the shape of a rod with a conical tip (Figure 3-3). Argon, Ammonia or catalytically decomposed Hydrazine is used as the propellant gas and is heated by an arc discharge. High current densities around the electrode deteriorate the performance of the thruster and are perhaps the dominant factor for the lifetime of the thruster.

Research efforts mainly concentrate around problems like high thermal flux concentration, efficient cooling methods and associated gas dissociation. In order to obtain high performance efficiencies, it requires high chamber pressures. However, this will reduce thruster lifetime considerably. Lifetimes greater than 800 hrs have been demonstrated. The exhaust velocities of very high power arc-jets have been reached to the order of $5,000 \text{ ms}^{-1}$ to $10,000 \text{ ms}^{-1}$ ($I_{sp} = 450 \sim 1000 \text{ sec}$) using low atomic mass propellants (H_2) with thrust levels 0.2 to 0.25 N (Sutton and Biblarz, 2001). Low power arc jets operating at input power in the order of 0.4-1.5 kW using Hydrazine mixtures have been recorded (Ghislanzoni et al. 1990).

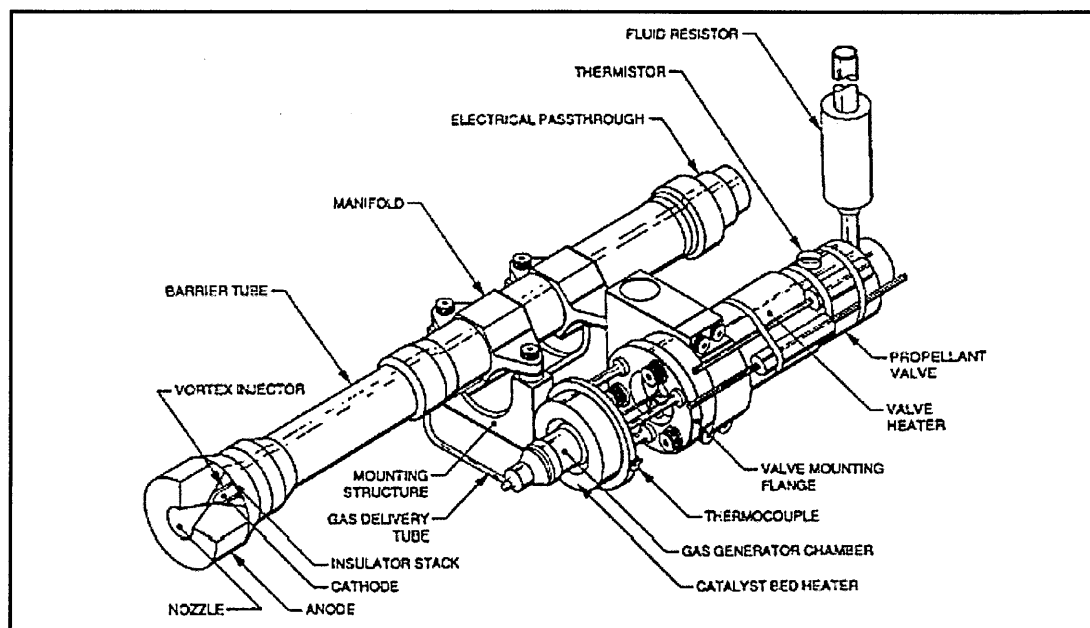


Figure 3-3 The commercially available 1.8 kW hydrazine Arc Jet assembly [from (Welker and Dudzinski, 2001)]

Recent experimental studies (Hideyuki and Itsuro K., 1999) had shown stable 'arcjet' operation at very low power levels ranging from 5-90 W. These experiments were carried out using small sized nozzles with throat diameters ranging from 0.3 to 0.7 mm and throat to exit-plane lengths ranging from 0.5 to 2.0 mm. Nitrogen gas was used as propellant. Recorded specific impulses were around 125 seconds with 5 W and proportionately changing with increased power levels. The plasma temperatures at the central parts were around 4000 to 6000 K and were increasing with reduction in size. These high temperatures together with the relatively large volume of the insulator (BN-material with 60 mm diameter, 40 mm in length, 0.3 mm diameter exit nozzle) constituting the main body of the 'Arc-jet' renders a practical limitation for this thruster miniaturisation for nano-satellite applications.

3.5 Electromagnetic and Plasma Propulsion

If a gas is sufficiently ionised to conduct a reasonable amount of current, magnetic forces can be exerted upon the gas to accelerate it. Generally this acceleration is achieved by the interaction of electric and magnetic fields on highly ionised propellant plasma at temperatures usually above 5000 K. Plasma contains positive ions, electrons and neutral particles. Therefore on the whole, plasma remains electrically neutral. If an electric current (j) is passed through a partially ionised gas and a magnetic field (B) is applied perpendicular to the current, a 'body' force ($f = j \times B$) will be produced. This force plus the usual pressure forces may be utilised to accelerate the gas to very high velocities (3 km/s to 16 km/s).

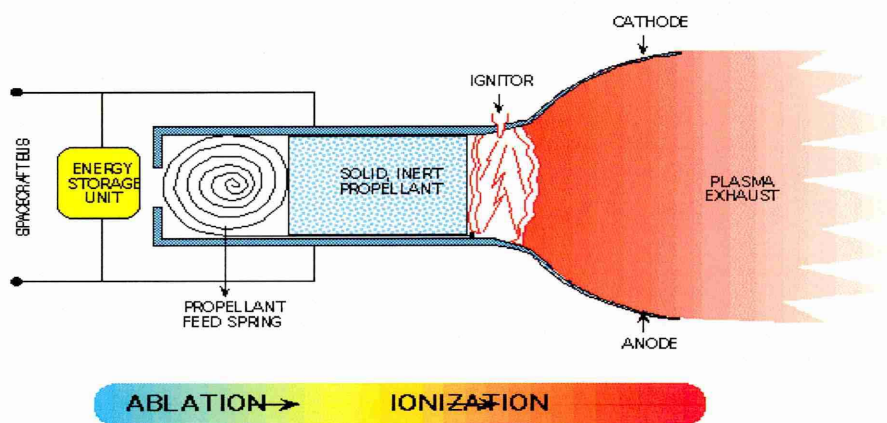


Figure 3-4 Schematic diagram of a Pulsed Plasma Thruster.

In Table 3-1 we categorised the Stationary Plasma Thrusters (SPT) under electrostatic thrusters as it is normally designated so. The SPTs use the Hall effect to set up an electrostatic field, which accelerate the propellant ions. Magneto plasma dynamic (MPD) thrusters use an electrical arc discharge, similar to an arc-jet, to charge the propellant. The propellant plasma is accelerated by the interaction of the arc and the self-induced and applied magnetic fields. Pulsed plasma thrusters (PPT's) accelerate the propellant plasma by interaction of an electric arc current with a self-induced magnetic field. The pulsed inductive thrusters (PIT's) accelerate the propellant plasma by the interaction of a generated azimuth current with a magnetic field from a coil current (Sutton and Biblarz, 2001).

Studies on electromagnetic thrusters were started in the early 1960s in the Soviet Union as well as in the USA. There is an extensive database of flight-qualification experience for solid propellant pulsed plasma thrusters (Figure 3-4). The major problems experienced during the course of PPT development were capacitor failure, propellant feed jamming, uneven erosion of the propellant and insulator breakdown between the capacitor and the

thruster assembly(Myers, 1993). Current research concentrates around problems of the transfer of momentum from the ionised to the neutral portions of the gas, heat transfer to the walls, maintenance of a very thin sheath carrying current and diagnostic methods in general. The lifetime of all plasma engines is still short, mainly due to sputtering and erosion of chamber and nozzle walls.

Within the frame of small to medium satellite applications, characteristics of Hall-effect thrusters and Pulsed Plasma thrusters (given in Table 3-3) are interesting, as their thrust and power range are of low power consumption while producing a reasonably high thrust. For small satellite application point of view the Earth Observation-1 (EO-1) spacecraft Pulsed Plasma Thruster with thrust to mass ratio of 306 $\mu\text{N/kg}$ (without power conditioning unit) looks promising. The near term applications in the regime of nano-satellites are doubtful. The major issue preventing these thrusters being applied for nano-satellites is its high power demand. By inspecting Table 3-3 power demand, these thrusters may find applications with spacecraft mass above 50 kg class.

	SPT-25	SPT-35	SPT-50	X-40	EO-1 PPT
Power, W	100 – 200	200	350	100 – 500	70 W @ 28 V
Thrust, mN	5 – 10	10	20	8 – 30	1.4
Specific impulse, s	1100 – 1400	1200	1250	1000 – 1900	1150
Overall Efficiency %	25 – 35	30	35	30 – 55	8
Thrust/ Mass Ratio	-	-	-	-	306 $\mu\text{N/kg}$

Table 3-3 Few SPT (Helvajian, 1995) and EO-1 PPT characteristics.

It had been identified that SPTs will probably be used with missions that need to fill the performance gap between hydrogen arc-jets and xenon ion engines in the near future (Pollard et al. 1993). However, as SPTs have a higher beam divergence than ion engines, and exhaust velocity has an azimuth component present, it's not capable as a precision thruster. As can be seen from Table 3-3, thruster efficiency as well as specific impulse drops for low power levels. Xenon and Teflon are being used as propellants for SPTs and PPTs respectively. Teflon PPTs' have been used for exact orbit maintenance in the past (Janson, 1993) and most recently on EO-1 spacecraft (Zakrzewski et al. 2002).

3.6 Electrostatic Propulsion

In this category, the propellant is first ionised or charged and then accelerated, using electrostatic forces. Electrostatic thrusters are classified, mainly into three groups, according to the mechanism that is employed to generate ions (extract electrons) from the neutral propellant. They are classified as follows:

- Electron bombardment thrusters.
- Contact ionisation thrusters.
- Field emission (FEEP) / colloid thrusters.

From all three types above, the electron bombardment thruster is widely developed and used. Contact ionisation thrusters operate by passing the propellant vapour, usually caesium, through a hot (1100°C) porous tungsten contact ioniser producing Ions (Sutton and Biblarz, 2001). However, this technology has not been developed in recent years and will not be taken into study here. As the work in this research study is mainly concentrated into FEEP thrusters that extract ions by field evaporation, a more detailed study on it will be carried out in the next chapter. Ion bombardment types will be studied here in order to generate the possible functional similarities with the FEEP. The following fundamental equations will be used to investigate the performance of an electrostatic thruster regardless of its category. ‘Stuhlinger’ had explained in depth, the fundamentals of all types of electric propulsion (Stuhlinger, 1964). Projects like NASA new millennium program, which tested an Ion thruster (NSTAR/DS1) in its deep space-1 spacecraft (Polk et al. 2001), will help with further improvement of electrical thrusters.

Operational Principle

In the conservation of energy equation the kinetic energy of a charged particle must equal the electrical energy gained in the field, provided that there are no losses. For a single particle of mass m , charge q , accelerating through a potential V_a imposed across the accelerating chamber or grids, and with final exhaust velocity u_e , conservation of energy equation gives,

$$\frac{mu_e^2}{2} = qV_a$$

Equation 3-3

If the exhaust beam current is I , the number of charged particles N , passing through a given exhaust cross section per second is,

$$N = \frac{I}{q}$$

Equation 3-4

Thrust F , is developed due to the rate of change of momentum of the system (Newton's equation). Therefore thrust force is calculated using Equation 3-3 and Equation 3-4 as below,

$$F = Nmu_e = \frac{I}{q}m\sqrt{\frac{2qV_a}{m}} = I\sqrt{\frac{2mV_a}{q}}$$

Equation 3-5

As it can be seen from the Equation 3-5, the delivered thrust is a function of propellant characteristics such as mass to charge ratio m/q , beam current I , and applied accelerating voltage V_a . For a given current and accelerator voltage the thrust is proportional to the mass-to-charge ratio of the charged particles. From the thrust point of view, propellants with heavy particles (high atomic mass) are of interest to gain high thrust. In practice the selection of propellant depends on many factors such as ionisation power requirement, propellant storage capability, hazardous factors etc. Mass-to-charge ratios for some candidate electrostatic propellants are shown in Table 3-4.

Propellant	Atomic mass (amu)	m/q (kg/coulombs)
Mercury (Hg)	200.6	2.10×10^{-6}
Caesium (Cs)	133	1.39×10^{-6}
Xenon (Xe)	131.3	1.37×10^{-6}
Indium (In)	115	1.20×10^{-6}
Rubidium (Rb)	85.5	0.89×10^{-6}

Table 3-4 Mass-to-charge ratio comparison of typical electrostatic propellants

The Equation 3-5 was derived assuming there are no losses in the process. Losses appear only when considering the fraction of propellant supplied which is actually accelerated, leading to a propellant utilisation efficiency, η_m . Also the fraction of the power supplied that is usefully employed in the beam, leading to an electrical efficiency, η_e . In electrostatic thrusters due to charge concentration, the beam diverges from it axial flow, leading to beam divergence efficiency, η_d . The total efficiency, η_t , is then,

$$\eta_t = \eta_m\eta_e\eta_d$$

Equation 3-6

Of course, this separation of efficiencies is possible only with certain thrusters, mainly gridded ion, FEEP and colloid devices. Others define only η_t , which combines all losses. The overall propellant utilisation is generally express as specific impulse, I_{sp} , which is defined as ratio of thrust, F , to rate of use of propellant in units of sea level weight, so that,

$$I_{sp} = \frac{F}{m g_e}$$

Equation 3-7

Where g_e is the acceleration due to gravity at sea level ($\approx 9.81 \text{ m/s}^2$) and m is the rate of flow of propellant in kg/s.

Only a fraction, η_m , of propellant flow rate, m is used to produce thrust, owing to losses. So,

Thrust, F = rate of change of momentum

$$F = \eta_m m u_e$$

Equation 3-8

From the definition of specific impulse,

$$I_{sp} = \frac{F}{m g_e} = \frac{\eta_m m u_e}{m g_e} = \frac{\eta_m u_e}{g_e}$$

Equation 3-9

Where $\eta_m u_e$ is defined as the effective exhaust velocity, charge particle velocity relative to the vehicle or thruster. From Equation 3-3 and Equation 3-9,

$$I_{sp} = \frac{\eta_m u_e}{g_e} = \frac{\eta_m}{g_e} \sqrt{\frac{2qV_a}{m}}$$

Equation 3-10

According to Equation 3-10, for a given propellant mass-to-charge ratio and propellant utilisation factor, η_m , increase in the accelerator potential will increase the thrusters specific impulse. This in return requires a greater input power from the power plant. Typical input power and specific impulse distribution for different beam currents are shown in Figure 3-5 below. It further shows, increase in beam current (higher mass flow rate for a given mass-to-charge ratio) correspond to increase in power demand and higher specific impulse.

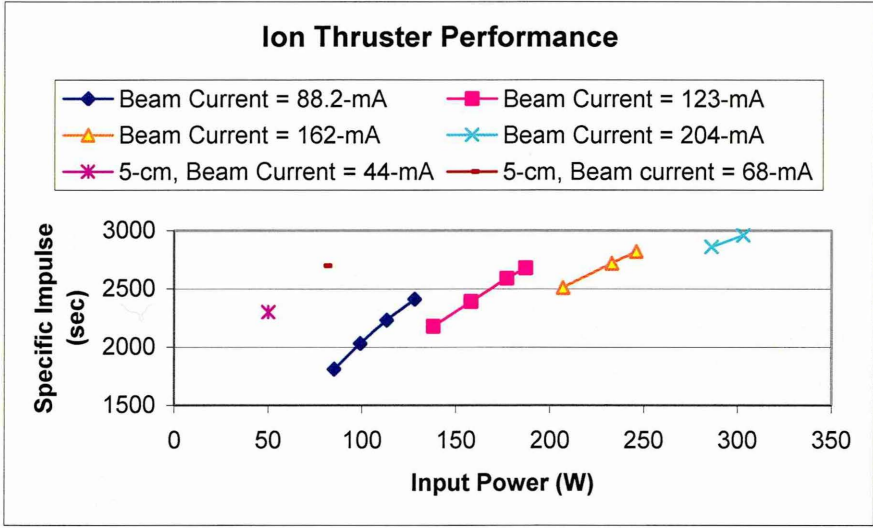


Figure 3-5 typical electrostatic thruster specific impulse distribution with respect to input power.

In electric propulsion requirement to have high specific impulse, I_{sp} , and high thrust, F , correspond to a increase power requirement for the thruster. Increase in power demand to the thrusters generally increases the size and mass of the onboard power plant. Of course it is possible to provide extra power to the thrusters without increase in size and mass of the power plant by efficiently distributing the power plant among different subsystems. According to Equation 3-10, application of a high accelerator voltage, V_a , increases the specific impulse for a given mass to charge ratio of a propellant. This is illustrated in Figure 3-5 for specific impulse variation due to applied input power ($Power = VI$), taking a 8-cm (Patterson and Oleson, 1997b) and a 5-cm (Gorshkov, 1998) xenon Ion thruster. It is evident here that with limited size and power availability the 5-cm ion thruster has attractive performance characteristics compared to the bigger 8-cm thruster. Although the 5 cm ion thruster gives much lower thrust than the 8 cm thrusters, for small satellites (micro and mini satellites) these thrust levels are adequate for many attitude and orbit control applications. From the nano satellites point of view, the input power demand of these two ion thrusters are above the current nano-satellites power generation capabilities.

Thruster electrical efficiency, η_e , is defined as the ratio of the beam power, P_b , to the total electrical power supplied, P_{in} , to the thruster. So,

$$\eta_e = \frac{P_b}{P_{in}} = \frac{\frac{1}{2} m u_e^2}{P_{in}} = \frac{\frac{I}{q} \frac{1}{2} m u_e^2}{P_{in}}$$

Equation 3-11

Equation 3-11 predicts that power efficiency at a given specific impulse ($u_e = I_{sp} \times g_e$) is greater for high atomic mass propellants, if losses remain unchanged. Obviously thruster efficiency will increase with higher specific impulses. Propellant utilisation efficiency, η_m , is defined as the ratio of the rate of mass usefully employed in the exhaust beam to the input mass flow rate. So,

$$\eta_m = \frac{\frac{I}{q} \dot{m}}{\dot{m}}$$

Equation 3-12

So for an axial flow without beam divergence, the total efficiency,

$$\eta_t = \eta_e \eta_m = \frac{\frac{I \dot{m} u_e^2}{2qP_{in}} \frac{I \dot{m}}{q \dot{m}}}{\frac{I^2 \dot{m}^2 u_e^2}{2q^2 \dot{m} P_{in}}} = \frac{I^2 \dot{m}^2 u_e^2}{2q^2 \dot{m} P_{in}}$$

Equation 3-13

As thrust is defined as the rate of change of momentum,

$$F = \frac{I}{q} \dot{m} u_e$$

Equation 3-14

From Equation 3-13 and Equation 3-14,

$$\eta_t = \frac{F^2}{2 \dot{m} P_{in}}$$

Equation 3-15

and,

$$\frac{P_{in}}{F} = \frac{F}{2 \dot{m} \eta_t}$$

Equation 3-16

From Equation 3-9 and Equation 3-16,

$$\frac{P_{in}}{F} = \frac{I_{sp} g_e}{2 \eta_t}$$

Equation 3-17

According to Equation 3-17, to achieve a higher specific impulse from a thruster, the applied input power must increase proportionately if the thrust remains constant. This can be generalised to all electric propulsion devices. Therefore for small satellites, demand for high specific impulse and high thrust electrical thrusters may have a higher mass penalty that may fall outside the design limits of this category of S/C. However high specific impulse and low thrust is entirely feasible for small (micro and mini satellites) and nano-satellites.

As shown in Figure 3-6 higher thruster efficiency and specific impulse correspond to increase beam current, in a particular thruster (not necessarily true generally). However the amount of power input to the thruster is limited mainly due to the limited power availability. For small size electrostatic thrusters the need to have a higher potential field for propellant charging and acceleration, may pose another limitation. The main reason behind this reasoning is that with high potentials the thruster electrodes (especially with FEEP thrusters) will tend to short circuit in the form of an electrical ‘arc’. This is true for all types of electrostatic thrusters, as the spacing between the electrodes are small and the applied potential difference are high. However, by maintaining the applied potential field just below the arc threshold this situation can be overcome in practice.

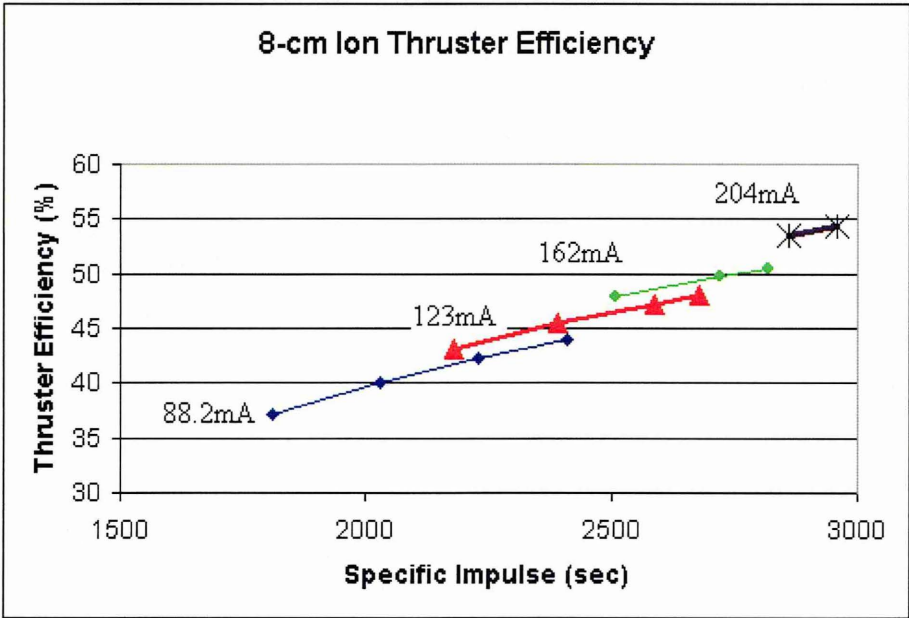


Figure 3-6 Thruster efficiency for an 8-cm xenon ion engine and the corresponding beam current {data from table 7, (Patterson and Oleson, 1997b)}.

Ion Thruster

Ion thrusters have been used successfully since 1964, when it was first used on the SERT-1 (Space Electric Rocket Test 1) spacecraft, in many space applications. The SERT-1 spacecraft had a 8 cm diameter caesium contact ion engine and a 10 cm diameter mercury electron bombardment ion engine and was the first successful flight test of ion propulsion. Since then many different types of ion engines have been developed. Ion thrusters are categorised as follows according to their employed method of ion generation.

- Contact ion engine
- Microwave ion engine
- Radio frequency (RF) ion engine
- Plasma separator ion engine
- Radioisotope ion engine
- Electron-bombardment (EB) ion engine

The contact ion engine creates ions by passing liquid caesium through a hot, porous bed of tungsten. The heated metal tungsten imparts energy to the propellant to extract ions. The resulting ions undergo electrostatic acceleration, producing engine efficiencies greater than 55% and specific impulses around 3500 to 8000 seconds.

The microwave ion engine creates ions in discharge chamber by microwave excitation of free electrons in the propellant gas. No cathode is used in the discharge chamber, which will enable use of propellants such as oxygen and carbon dioxide. Ion beam is accelerated through an electrostatic field.

The radio frequency (RF) ion engine creates ions in an insulating discharge chamber by MHz radio frequency excitation of free electrons in the propellant gas. An RF induction coil surrounds the cylindrical discharge chamber, and neither a discharge chamber cathode nor magnetic field plasma containment scheme is used. Highly efficient electrostatic acceleration produces thrust. Most work on the RF ion engine has been done in Germany. The 10 cm diameter RIT-10 developed by DASA (Daimler Benz Aerospace AG) was successfully tested on the EURECA (European Retrievable Carrier) in 1993 (Pollard et al. 1993).

Plasma separator ion engine uses no discharge chamber for ion creation. Rather, a high density, fast-flowing plasma is created by feeding propellant vapour (i.e. caesium) through an array of hollow cathodes, each incorporating a convergent divergent nozzle. The vapour is ionised by a coaxial discharge between the cathode and an annular anode ring located at the end of the nozzle. The dense plasma is expanded aerodynamically by a conventional large area multiple aperture electrostatic accelerator system. All development work occurred in the U.S. in the late 60s, and no space tests have been performed.

In the radioisotope ion engine, a charged beam of colloidal particles ejected at high velocity produces thrust. Electrical power for the colloidal particle accelerator system is obtained from the beta particle decay of Caesium¹⁴⁴ fuel, which is thinly applied over a large emitting surface. The radioisotope ion engine uses a potential difference created from the particle decay process to accelerate a high-energy colloidal beam. No constructive development work has been reported since its conception in the early 1960s in U.S.

The main components of a commonly used electron bombardment type thruster are shown in Figure 3-7. The electron bombardment (EB) ion thruster produces thrust by accelerating a beam of positive ions through an electrostatic field to high velocity. A neutral propellant is injected into an ion production chamber and the ions are produced from the propellant by electron bombardment. The ions then drift to a set of grids at one end of the discharge chamber where they are extracted and accelerated due to the applied potential gradient. The resulting change of momentum produces thrust. Electrons are injected separately into the beam at the end of the thruster, mainly to prevent the spacecraft acquiring a net negative charge. The electron bombardment source (Kaufmanns source) has a consistently high mass and power efficiency. At low exhaust velocities, its power efficiency is superior to that of other sources.

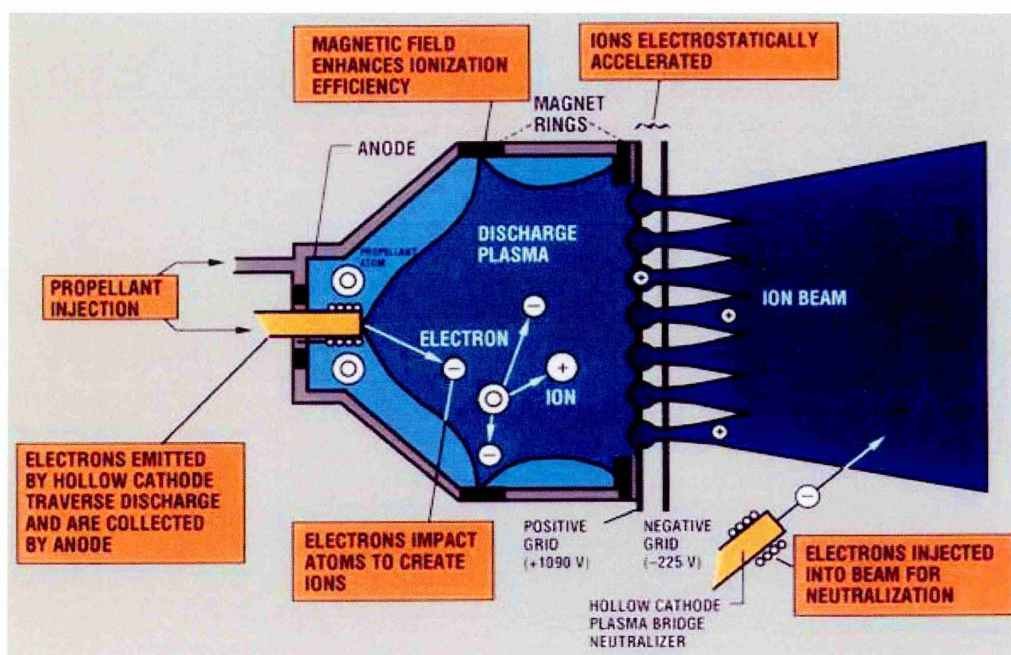


Figure 3-7 Ion thruster main regions (Rayman, 1998).

Ion thrusters are often categorised by the diameter of their grids because the power and thrust of the engine is proportional (if all else being equal) to the square of the grid diameter (i.e. grid area). The optimum diameter of a typical ion thruster is around 10 to 30 cm (example UK-10 and 25). Caesium, Mercury and Xenon have been widely used as a propellant for ion engines. The key life-limiting factors of the engine are (Fearn, 1987):

- Main life-limiting factor of the engine is loss of barium dispensers within the cathode and neutraliser
- Sputtered material deposition within the discharge chamber.
- Deposition of material sputtered from the accelerator grid.
- Heat dissipation to the spacecraft and deposition of conducting layers on insulators.

The Japanese ETS-6 uses contamination shields to prevent sputtered grid material reaching the spacecraft surfaces, and thermally-controlled mounting brackets for minimal ‘soak-back’ are two examples of the research and development in this area as of 1993 studies (Pollard et al. 1993). So far the studies on development of a low powered, miniaturised Ion thruster have limited results. At the ‘Keldysh Research Center’ in Moscow (Gorshkov, 1998), two types of ion thrusters were identified as viable options for future application for small satellites. Data on specific impulse and efficiency ($\eta = \text{Beam power} / \text{Input power}$) are presented in Table 3-5 without mass and energy losses in the neutraliser. Also it had been assumed that the beam is singly ionised.

Manufacturer	NASA Lewis	Keldysh Research Centre, Russia	Keldysh Research Centre, Russia
Discharge type	DC	DC	DC
Beam diameter, cm	8	5	10
Thrust, mN	3.6 – 10.9	1 - 5	6 - 19
Isp, s	1760 - 2650	3100 - 3700	2500 - 3500
Power, W	100 – 300	50 – 140	150 - 500
Efficiency (η), %	31 - 47	30 - 65	49 - 65
Comments	Laboratory development	Laboratory development	Laboratory development

Table 3-5 State-of-the-art small ion engines (Micci and Ketsdever, 2000)

Ion thrusters together with their high specific impulse and high thrust provide a mass penalty mainly in power conditioning unit. As Table 3-5 lists all current ion engines have very high power demand, which may be applicable to spacecraft mass greater than 50 kg. Therefore integration of ion thrusters for nano-satellite is not feasible with the current technology. Beside the power plant mass some of the main problems in development of a low-powered small-sized ion thruster include:

- Decrease in efficiency due to the small size of the chamber
- Low functional efficiency at low power levels with the neutraliser
- Primary electron losses at the anode (can be alleviated by correct magnetic field configuration)
- Decrease in power utilisation
- Decrease in electrons residence time

According to the extensive tests carried out at Defence Evaluation and Research Agency, Farnborough, UK, it had shown by decreasing the cathode orifice diameter, increase in temperature and discharge noise (Fearn, 1999). Further it has shown that the lifetime of a hollow cathode type is very dependent upon temperature. Extreme miniaturisation of Ion thruster is not an attractive choice owing to reduced efficiencies. However ion thrusters with parameters as given in Table 3-5 can be appropriate for external disturbance compensation in LEO, orbit raising and possibly attitude control (Gorshkov et al. 1998) for advanced communication and remote sensing satellites (50 to 500 kg). It should be noted that with low power ion thrusters, effective beam neutralisation is a serious problem, if hollow cathode is not used. This is especially true for the low power 5 cm ion thrusters (Akimov et al. 1997).

In order to perform a given manoeuvre using EP, satellite specific power at the end-of-life must be greater than a certain threshold value (Janson, 1992). For applications in drag compensation, power to the thruster can be supplied when the satellite is in direct sunlight. Assuming that the thrusters are operated only 25% of the orbit period and a nano class satellite in a typical low earth orbit, required acceleration is about 10^{-4} m/s^2 (typical air drag values for a 300 km altitude orbit and a satellite with a 20 kg/m^2 ballistic coefficient). This gives according to Equation 3-1 and Table 3-1 needed minimum satellite specific powers around 3.7 W/kg for FEEP Caesium thruster with a specific impulse (I_{sp}) of 6000 seconds, 2.5 W/kg for Xenon Ion engine with a I_{sp} at 3000 seconds and 0.5 W/kg for hydrazine resistojets at I_{sp} around 500 seconds. However the exact values are higher than this, since the above calculation is based only on requirement of thruster power.

For a given satellite on board power is mainly governed by the payload power requirement. For electric propulsion satellites it is common to have a distributed power between the payload and the electric propulsion system. However as the onboard power demand increases the satellite mass increases as well. Therefore the ratio between the solar array average power to satellite total mass, which is defined as the specific power, gives a reasonable estimate of the technology development in satellite power systems. Representative values for the specific power development of a few satellites are given in Figure 3-8 below. It should be noted that the SMART-1, part of the mission is an electric propulsion demonstration (ion thrusters) and is therefore can be treated as a payload power requirement. SMART-1 falls under the mini satellite category (total mass around 367 kg) and has deployable solar arrays extend about 14 meters. With deployable arrays current communications satellites power generation exceeds 10 kW. However due to the mass and volume constrain in nano-satellites the power generation may limit to body mounted solar arrays only.

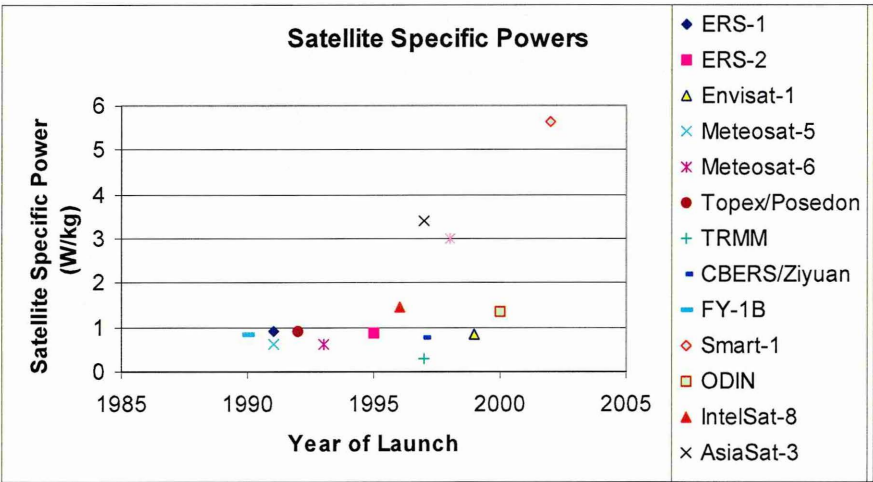


Figure 3-8 Specific power developments of few satellites according to the launch year (derived using information available at the ESA web site: (Fulvio Drigani, 2000)).

3.7 Summary

Conventional electric propulsion methods were discussed to show the characteristics differences in their operation. It has shown here that these conventional electrical propulsion systems are used successfully in many space applications. Comparisons were made with respect to FEEP thrusters in the areas of power demand, efficiency, lifetime and specific impulse. It has shown the total efficiency of an electrical thruster as a function of efficiencies due to electrical, propellant ionisation and beam divergence. Progress in the development of satellite specific power was discussed to explain that the high specific powers are feasible in the near future. Ion thruster technology was explained in more detail as the FEEP technology falls into this category. It has shown that due to the high power demand many of the available propulsion systems do not suitable as micro-propulsion thrusters. However FEEP thruster technology was not discussed here, as the rest of this study will be dedicated for that purpose.

4 Field Emission Electric Propulsion

A brief description of the FEEP thruster development and important milestones are given at the beginning of this chapter. Thruster performance is discussed, taking into consideration its operational behaviour. Main components of the FEEP system with their current development status are addressed. The corresponding literature survey was carried out from an engineering perspective rather than investigating in depth any physical behaviours. The current state of the slit-emitter together with the associated problems has been addressed. The thruster space application possibilities are mainly discussed with the encountered problems. Finally the FEEP test facilities at Centropazio, Italy and ESTEC, Holland are compared and the FEEP ground-testing procedures are explained from the authors first hand experience at these sites.

4.1 Background

FEEP technology has been developed within the ESA, under Liquid Metal Field Ion (LMFI) sources for Electric Propulsion (EP) programs. The concept of Field Emission Electric Propulsion (FEEP) is well known since the 70's. First proposed by ESA/ESTEC, this thruster was investigated by several European research institutes during the past decade. Initial research was carried out with the aim of developing a North South Station Keeping (NSSK) system for 'geostationary' satellites. However at the end of the 80's FEEP was eventually judged less competent than other electric propulsion concepts, (especially ion thrusters and arc jets), mainly because of its high specific power. There was renewed interest in FEEP at the beginning of the 90's following the need for high performance 'drag-free' systems. Hence research and development of the FEEP was given a new impulse (Zandbergen, 1986). Starting in 1972, the FEEP emitter has been progressively developed. It evolved from a simple pin emitter (1973) through linear arrays of stacked needles (1975) into the high efficiency, linear solid slit emitter (1979). Since then the solid slit emitter has undergone intensive laboratory tests mainly at ESA/ESTEC and at the test bed at Centropazio/Pisa. At present the FEEP thrusters development is taking place in two fronts: the original pin type emitter with Indium as propellant and the slit type emitter mainly using Caesium propellant. In Europe the Indium pin emitter development is mainly taking place at the Austrian research centre, Seibersdorf, Austria and the Caesium slit emitter development is taking place at the Centropazio test facility at Pisa, Italy. The main advantage of the solid slit emitter compared to pin emitter is its multi-emission sites that adapt according to the applied conditions. However from the propellant contamination point of view Indium is attractive, as it does not react with the residual oxygen as the Caesium does. The developmental milestones of the FEEP thruster are summarised in Table 4-1.

Year	Emitter	Reference
1973	Pin Emitters	(Zandbergen, 1986)
1975	Nickel Array Emitters	(Zandbergen, 1986)
1977	Solid Slit Emitter	(Zandbergen, 1986)
1979	High Efficiency, Narrow slit emitter	(Zandbergen, 1986)
1980	Closed geometry, positive pressure	(Zandbergen, 1986)
1980	Study on additional propellants	(Stewart et al. 1980)
1998	Endurance test totalling 2000 hrs of operation	(Marcuccio and Genovese, 1997)
2000 -2001	Planned Flight demonstration- Shuttle GAS.	(Marcuccio, 1998b)

Table 4-1 developmental milestones of the FEEP thruster.

The open slit of the solid emitter had various sizes and was manufactured either in rectangular or elliptical form. These different geometries were selected to reduce the possible edge effects while the slit depth correspond to the efficiency of the thruster. An increase in slit depth gave rise to neutral propellant flow, which in turn reduces the propellant utilisation efficiency. Taking into account various aspects such as manufacturing easiness, wetting characteristics etc, showed that slit depth around 1.0 – 1.5 μm gave optimum FEEP operational conditions. In order to introduce the propellant to the slit from the emitter tank, tests were also performed with higher-pressure gradient (close geometry, positive pressure). However with the optimum slit depth, pressure gradient requirement was eliminated as a result of capillary forces.

In the past, most FEEP development efforts were focused on extracting high thrust levels, at lest in the *milli-Newton* scale. At thrust levels of above a few *milli-Newton*, FEEP has some drawbacks due to its very high specific impulse, which is associated with a very high power-to-thrust ratio. The recent shift of perspective on FEEP technology, which now focuses on the very low current capability of this thruster, has drastically changed the situation. The new FEEP thruster received much attention especially from the scientific community mainly because of its μN range of thrust level. The need for a drag free, precise station-keeping platform is the baseline for many of the proposed future applications for the FEEP thruster. Few such projects are the ODIE (Orbiting Drag Free International Explorer, a 100 kg platform) and highly complex missions such as the LISA (Laser Interferometer Space Antenna, see Appendix-B).

The main areas of application for the FEEP thruster in the near future will be for ‘fine-tuning’ of spacecraft attitude and orbital parameters, where the required thrust levels are in μN range. At very low thrust levels the FEEP slit emitter performs best, and its features make it the only candidate for several space applications. Due to the low propellant consumption, the propellant tank could be integrated in to the emitter itself. Few grams of

propellant could be stored in this in-built tank. The propellant is supplied directly to the emitter slit, avoiding the complicated propellant feed system, taking advantage of the capillary effect and the applied electric field. In FEEP emitters, unlike most ion engines, ions (mostly singly charged) are directly extracted from the liquid phase leaving electrons in the liquid.

The FEEP thruster can accelerate a large number of different liquid metals. High atomic weight, good wetting characteristics and low ionisation energy are the main important characteristics in selecting a propellant (Wilson et al. 1989). The wetting characteristics of a propellant represent the effect of 'adhesion' forces between the propellant and the emitter. Importantly the capillary action is another manifestation of surface tension, which consists of 'cohesion' and 'adhesion' forces. The 'cohesion' forces arise due to the molecular attraction in the liquid propellant. Appendix-C summarises some important propellant characteristics that is useful in this study.

4.2 FEEP System

The total FEEP system could be divided into four main parts. Each of them has a unique function and demands skilled design and manufacture for problem-free operation. However after almost 30 years of intensive study, there are some technological challenges to solve before applying in an advanced space mission. The total FEEP system performance was investigated by studying the following four main sub components.

- FEEP emitter.
- Neutralizer.
- Propellant storage and feeding system.
- Power conditioning and control unit (PCU).

The current FEEP emitter consists of two almost identical halves clamped together as shown in Figure 4-1. These emitter halves are manufactured in a way that it could store few grams of propellant within its in built tank. A sharp blade is accurately machined on one side of each emitter half. The FEEP emitter/accelerator geometry facilitates the creation of the high electric field, which is essential for production of ions, without excessive potentials. The capillary forces and the applied electric field allow the propellant to be transported from the storage to the emitter slit. When the electric field is removed, due to the very small size of the slit, capillary forces become so strong that it prevents propellant from pouring out of the slit. Initially when the propellant starts to flow from its built-in tank toward the slit, the surface tension forces become increasingly high as the channel depth decreases. This explains the initial start up difficulties associated with the slit emitter. As the propellant consumption is very low in FEEP thrusters (due to high specific impulse), the emitters 'built-in' propellant reservoir is adequate for most of the applications concerned. The emitter lips as well as the propellant feed slit need a high degree of smoothness, to provide the desired 'wetting' conditions.

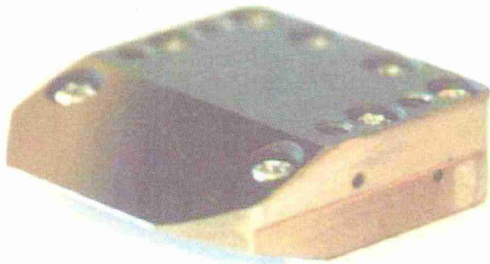


Figure 4-1 FEEP emitter (Credit: Centropazio, Italy)

When the ions are extracted the electrons are rejected in the bulk of the liquid. Therefore an external source of electrons (neutraliser) provides negative charges to maintain global electrical neutrality of the thruster assembly. The concept of neutralisation is similar to that used in other common electrostatic thrusters, such as ion thrusters. The most common and simplest neutraliser is the hot filament cathode type. However for the FEEP thrusters, it is advantageous to use a power and mass efficient neutraliser, in keeping with the FEEP applications. The current goal is to keep the power requirement under 2% of the total thruster power consumption. Much of the work with respect to this is under way at *Centropazio/Italy* and *Rutherford Appleton Laboratory in UK (Tajmar, 2002)*. In the past *SEP/France* had carried out some investigations with respect to FEEP neutralisers.

The power-conditioning unit (PCU) converts the unregulated power (from solar cells) to different voltages and current levels (required to start up and for nominal operation). It is important that the PCU tolerates any operating anomalies due to eclipses, seasonal effects etc. As the FEEP thrusters are 'power demanding' ($\sim 60 \text{ W/mN}$) and require a very high electric field ($\sim 10^9 \text{ V/m}$) at the electrode edges, the design and operation of the PCU is a complex component in the system. PCU's of low mass/size with low power consumption are the major drivers for the FEEP-thruster applications in nano-satellites. However, a detailed study of this is carried-out under 'Electrical Power' sub-system, and is addressed together with the spacecraft electrical power system.

Operational Mechanism

The main elements of the thruster are the emitter, neutraliser and the accelerator electrode to establish the voltage difference needed for ionisation and acceleration as shown in Figure 4-2. The present emitter module consists of two metallic plates with a built-in small propellant reservoir. A sharp blade is accurately machined unilaterally on each plate at the slit. A thin layer of material (*Ni*) is sputter-deposited on three sides of one of the plates, to act as a spacer. When the two-emitter halves are tightly clamped together, a slit of about $1.0 - 1.5 \mu\text{m}$ is left between the blades. The propellant flows through this tiny channel, forming a free surface at the exit of the slit. Radius of curvature of the free surface is in the order of $1 \mu\text{m}$ (as of slit width). A state of local equilibrium arises at the protruding liquid surfaces

due to the combined effects of the electrostatic force and the surface tension. The free surface of the liquid metal approaches a condition of local instability due to the increase in electric field strength. A series of protruding cusps, of the general shape predicted by Taylor called “Taylor cones” (see Appendix-C), are created due to the intensity of the electric field. When the electric field reaches a value of about 10^9 V/m, the atoms at the tip of the cones are spontaneously ionised and accelerated by the applied electric field, while the electrons are rejected in the bulk of the liquid. The accelerator electrode, located directly in front of the emitter, accelerates these ions to a very high velocity to produce thrust.

The main feature of the FEEP thruster is, direct ionisation of a liquid metal surface under a strong electric field. Ion emission is from a number of separate emission sites along the slit. The combined effect of the applied electric field and the geometry of the slit-end, determine the number of emission sites. The generated ions (positive) on the liquid surface are attracted towards the accelerator electrode due to its polarity (negative bias). The magnitude of acceleration is determined by the net potential drop from slit to accelerator and the charge to mass ratio (q/m) of the selected propellant.

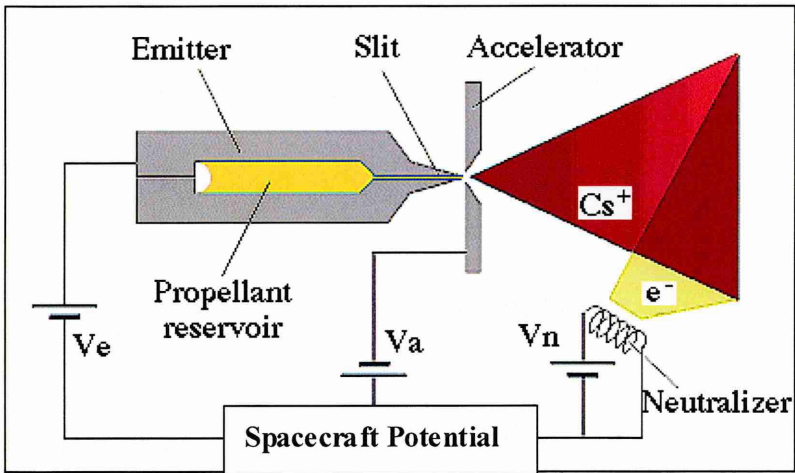
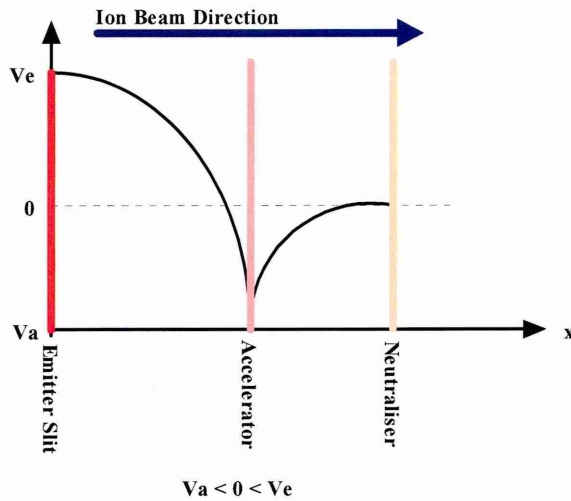


Figure 4-2 The FEEP thruster principle (Marcuccio, 1998a).

However, to produce a useful thrust, the ion beam has to be neutralised after passing the accelerator electrode. If not, the positive space-charge potentials within the beam cause it to stall or reflect upon itself. The neutralised beam induces a region of increased potential aft to the accelerator electrode, which reduces the final velocity of the ion beam. The final beam velocity (v) is proportional to the square root of the applied emitter voltage (V_e) and charge to mass ratio (q/m) as shown in Equation 4-1.



Final Beam Potential when passing the neutraliser = $V_e - 0 = V_e$

Figure 4-3 One-dimensional ion beam potential distribution (space potential is indicated by the zero potential line).

$$v = \sqrt{2(V_e - V) \frac{q}{m}}$$

Equation 4-1

Where, V_e is the applied emitter potential and V the potential at position x within the thruster. According to Equation 4-1 as V varies with position x , it gives the variation of velocity within the thruster. At far distance ($x \rightarrow \infty$) the beam potential is effectively becomes zero ($V \rightarrow 0$), the beam is fully neutralised as shown in Figure 4-3. However, for a given propellant, the final beam velocity can be calculated according to the Equation 4-1 with $V = 0$, neglecting beam space charge effect, and thus considering the final ion velocity as a function of applied potential difference. It had assumed here that the final beam potential had reached to zero due to neutraliser effect. In practice this requires a good neutraliser to bring the charge beam to zero potential. Ions are generated at the emitter slit due to the combined effect of the emitter and accelerator potential (net difference in potential). The amount of thrust, which is of prime concern from the engineering point of view, is obtained by calculating the rate of change of momentum. However not all the mass that expels from the slit contributes to the thrust. As only the expelled ions carry a sizeable momentum (only ions are accelerated in an electric field), the produced thrust is calculated according to the Equation 4-2.

$$F = \dot{m}^+ v = \left(2 \frac{m}{q} V_e \right)^{1/2} I$$

Equation 4-2

Where,

F : Thrust

$\dot{m}^+$: Singly charged ion beam mass flow rate

v : Ion beam velocity

Equation 4-2 shows that it is advantageous to use propellants with large mass to charge ratio (m/q) to obtain increased thrust. Therefore the delivered thrust is a function of selected propellant and applied electrical field. If λ represents the propellant characteristics (square root of twice the mass to charge ratio) then the thrust equation can be re-written as of Equation 4-3. For Cs propellants with $q/m = 7.24 \times 10^5$ C/kg, $\lambda_{cs} = 1.666 \text{E}^{-3} (\text{kg/C})^{1/2}$ and for Rubidium and Indium $\lambda_{Rb} = 1.332 \text{E}^{-3} (\text{kg/C})^{1/2}$ and $\lambda_{In} = 1.544 \text{E}^{-3} (\text{kg/C})^{1/2}$ respectively. The above comparison shows the superiority of *Caesium* compared to other competitive propellants. In ground testing the delivered thrust is calculated from Equation 4-3 by measuring the current flow and the applied voltage. Here V_e is the positive potential applied to the emitter. This is the standard accelerate/decelerate system into space plasma, where the final result (velocity and thrust) is determined solely by the potential of the source of the ions with respect to the space potential.

$$F = \lambda_{prop} I V_e^{1/2}$$

Equation 4-3

$$\text{Where, } \lambda_{prop} = \sqrt{2m/q}$$

F : Thrust

I : Emitter current

V_e : Emitter voltage

m : Mass of a single propellant atom (for Cs⁺, $m = 2.21 \times 10^{-25}$ kg)

q : Propellant elementary charge (for Cs⁺, $q = 1.6 \times 10^{-19}$ C)

λ : Propellant characteristics

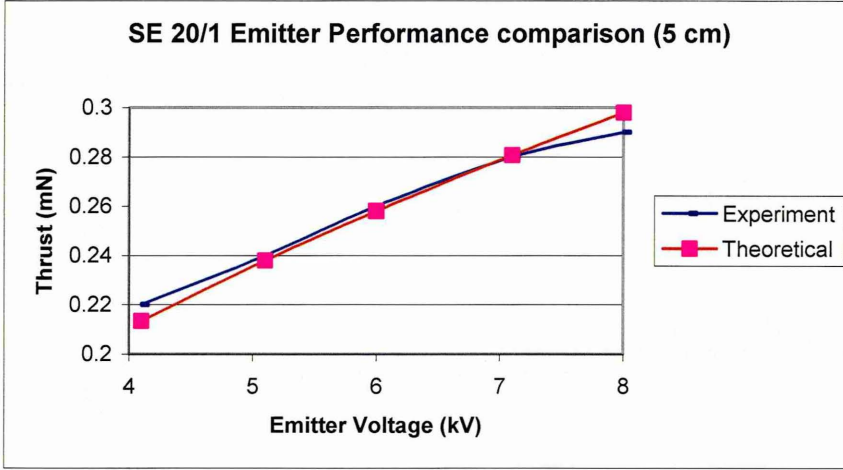


Figure 4-4 FEEP thruster evaluation results. {Table 5-5, Zandbergen,1986}.

Figure 4-4 shows the thrust calculated from the experimental test data and calculated thrust using Equation 4-3 as a function of applied emitter voltage at constant current ($I = 2mA$). It shows that Equation 4-3 is in general good agreement with the test results. However, divergence losses of the ion beam were not taken into account here. The generated ion beam diverges away from the emitter slit both in plane (XY) and out of plane (XZ), as shown in Figure 4-5. This effect is mainly due to the fact that ions are generated from concentric points along the slit, as well as due to the geometry of the accelerator aperture lens. Maximum divergence values in XZ-plane (α) are in the order of 45° and in XY-plane, (β) around 15° . Thrust degradation due to divergence, can be calculated according to the Equation 4-4. Nevertheless the maximum reduction in thrust will be less than 10% of the values calculated using Equation 4-3 (Marcuccio, 1998a).

$$F = I_e \left(2 \frac{m}{q} V_e \right)^{1/2} \frac{\sin \alpha}{\alpha} \frac{\sin \beta}{\beta}$$

Equation 4-4

The divergence losses becomes minimum when $\alpha \rightarrow 0$ and $\beta \rightarrow 0$, so that:

$$\frac{\sin \alpha}{\alpha} \rightarrow 1 \quad \text{And} \quad \frac{\sin \beta}{\beta} \rightarrow 1$$

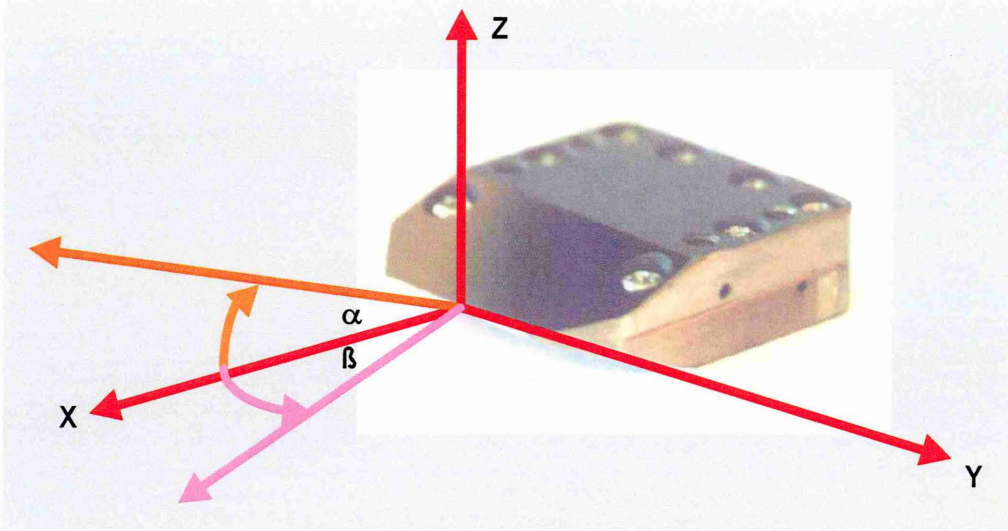


Figure 4-5 FEEP emitter module with beam divergence angles α and β .

B.T.Zandbergen {table5-13, (Zandbergen, 1986)} gives typical values of divergence in the XZ-plane. However, it is difficult to predict from these results, as these tests have been carried out with different emitters at various test beds in the past. Anyway, it is clear that beam divergence depends not merely on the accelerator electrode aperture but also the type of emitter, accelerating voltage, distance to the accelerator, composition of the ion beam etc. More investigations are needed in this area for accurate analysis. Therefore within the framework of this study the importance of beam divergence is studied to reduce the ‘plume’ impact on the satellite. In order to reduce the propellant impingement on the satellite body and over sensitive instruments, it requires a beam clearance of 15° and 45° in plane and out of plane respectively from the point of thruster installation. Great care must be taken when selecting locations for mounting the thruster, as propellant impingement could cause severe short-circuiting of the satellite due to the nature of the propellant (Alkali metal).

If the ions were perfectly neutralised and the space charge effect is negligible, the beam would diverge according to the Figure 4-5. Under this condition, the cross section area of the ion beam near FEEP emitter exit would be:

$$A_{ion} = (l + 2a \cdot \tan \beta) \cdot (w + 2a \cdot \tan \alpha)$$

Equation 4-5

Where (l) is the slit length, (w) is the slit width and (a) is the distance between the accelerator electrode and the emitter slit (see Figure 4-8). Typical values for FEEP-10 emitter are: $l = 10$ mm, $w = 1.5$ μ m and $a = 0.6$ mm.

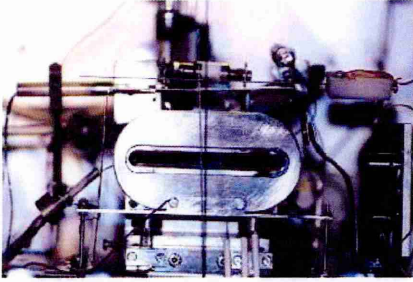


Figure 4-6 Elliptical accelerator being used at Centrospazio test facility.

The accelerator configuration that has being used at Centrospazio test facility is shown in Figure 4-6 above. The sharp elliptical aperture shape is of importance to provide an unobstructed path to the ions as well as to provide the required ‘edge’ effect to generate the maximum field needed for generation of ions. Aperture width in the order of 4-mm is typical for the Centrospazio accelerator.

Specific impulse, which is a figure of merit of propellant usage, is directly related to the beam exhaust velocity. According to Equation 4-1, in order to achieve high exhaust velocities, which in turn gives high specific impulse values, requires high voltage on the emitter (V_e). This implies to achieve increase thrust demands while maintaining high specific impulse, a high power demand according to Equation 4-6 from the power plant. From the nano-satellite point of view this is undesirable as to meet high power demands requires large power plants and power converters. Therefore it may be reasonable to operate the FEEP thrusters at low specific impulse levels in order to operate the thruster at acceptable power levels. In ground testing, the beam power is calculated by measuring the applied voltages and drawn out currents. For an accelerator/decelerator system, beam power, W_B , is calculated as:

$$W_B = V_e I = \frac{1}{2} F v = \frac{1}{2} \dot{m}^+ v^2 = \frac{1}{2} F I_{sp} g_E$$

Equation 4-6

Where,

V_e : applied emitter voltage

I : beam current

F : Thrust

v : beam exhausts velocity

$\dot{m}^+$: singly charged ion mass flow rate

I_{sp} : specific impulse

g_E : Earth gravitational constant at sea level ($\approx 9.81 \text{ m/s}^2$)

If there are no losses, the total power demand to the thruster is calculated according to Equation 4-7 from the measured voltages and currents. The FEEP thrusters has an overall

efficiency about 85% to 95%. The high efficiencies are achieved at low thrust levels and low efficiencies are corresponds to operation at maximum available thrust of the emitter.

$$W = V_e I_e + |V_{acc}| I_{acc} \approx V_e I_e$$

Equation 4-7

Where,
 V_e : applied emitter voltage
 I_e : measured emitter current
 V_{acc} : applied accelerator voltage
 I_{acc} : measured accelerator current

Usually the accelerator power consumption is neglected, as the current drawn from the accelerator electrode, which is defined as the accelerator current (I_{acc}), is much smaller than the current to the emitter (I_e , the emitter current). As shown in Table 4-2 the accelerator current is about 1000 times lower than the emitter current.

In Table 4-2 thrust was calculated from Equation 4-4 assuming zero divergence losses. Electrical power was calculated as of Equation 4-7 taking into account both the emitter and accelerator power consumption. This shows FEEP thrusters can be operated to give few milli-Newton thrusts, but it falls beyond the limits of power capability of current nano-satellites.

Emitter Voltage (kV)	Emitter Current (mA)	Accelerator Voltage (kV)	Accelerator Current (μA)	Electrical power (W)	Calculated thrust (mN)
5	12	2	22	60.0	1.13
6	170	2	33	1020.1	17.5
7	880	2	77	6160.1	98.0
8	2170	2	150	17360.3	258.4

Table 4-2 Chamber test results from Centropazio, 1-cm emitter with *Rubidium* as propellant (Credit: Centropazio, Pisa, Italy).

Ionisation

When a liquid is exposed to an electric field, a net charge is induced on its surface, unless the resistivity of the liquid is very high. The applied electric field distorts the energy balance on the liquid surface. In the framework of FEEP study, field ionisation and field ‘desorption’ have been identified as the mechanisms of ion formation {(Zandbergen, 1986) & (Patterson and Oleson, 1997b)}. In both of these categories, ions are extracted directly from the liquid metal surface, contrary to the electron bombardment method used in familiar ion engines.

According to the studies done by Schmidt {Schmidt 1964 #480} and Zandbergen (Zandbergen, 1986), it has been shown that field ionisation can only occur in a small region around the apex of protruding liquid cones. The work described by Zandbergen (Zandbergen, 1986) is directly related to the FEEP thrusters, while Schmidt had studied water vapour ionisation from solid wire tips. Secondly they had shown that for ionisation to occur under the effect of an applied electric field, it is necessary that the electrons at the outer shell of the atom be raised to the Fermi energy level. The energy applied corresponds to the atomic ionisation energy, which distorts the energy barrier and causes quantum mechanical tunnelling of electrons, leaving positively charged ions. Field ionisation requires potential gradients in the order of 10^9 V/m at the liquid metal tip. By studying the supply of neutral atoms to the region of ionisation, Zandbergen pointed out that no self-consistent model is possible for practical values of apex-radius ($r_a < 100\text{\AA}$) and temperatures with field ionisation. It has been further explained that the temperature requirement will increase beyond realistic values, to produce the necessary thermal flux with the field ionisation model.

The phenomenon of field desorption/evaporation is an adiabatically ionised process. The term 'Field desorption' is used to describe the formation of ions in the gas phase from a material (such as the Caesium propellant) deposited on a solid surface (emitter) in the presence of an electric field. As this process probably encompasses ionisation by 'field ionisation' as well as other mechanisms of ionisation the term 'field desorption' is not a precise one, despite its widespread acceptance. The 'Field evaporation' is the phenomena that atoms are ionised at the presence of extremely high electric field of the order of a few V/nm, around 50–60 V/nm for Tungsten (Zurlev and Forbes, 2003). Such high field can be achieved by applying a high voltage to a sharp needle or blade like specimen. G.Weizer (Weizer, 1971) has explained the principle of field evaporation for solid metal electrodes. The work done by Zandbergen is directly related to the FEEP liquid metal field evaporation. However the applied principles are the same. Basically without going into much detail, field evaporation can be explained considering the potential barrier of a liquid metal atom at its surface. The energy required (ϵ_i) for removing the valence electron, and to bring the atom to a singly charged state is represented in Figure 4-7. The application of a field to the liquid surface results in deformation of the ionic potential curve, as shown in Figure 4-7. As a result of the field, the energy barrier, ϵ_d , can be seen to be much smaller than ϵ_i . At sufficiently high fields, the ionic curve falls below the neutral curve, allowing desorption. In Figure 4-7, distance in horizontal axis represents the apex radius at the liquid metal tip. Zandbergen has shown that ion current increases with decreased apex radius and increased apex field. Allowing for uncertainty in the mechanism of ion formation, field evaporation is in good agreement with calculated and experimented data.

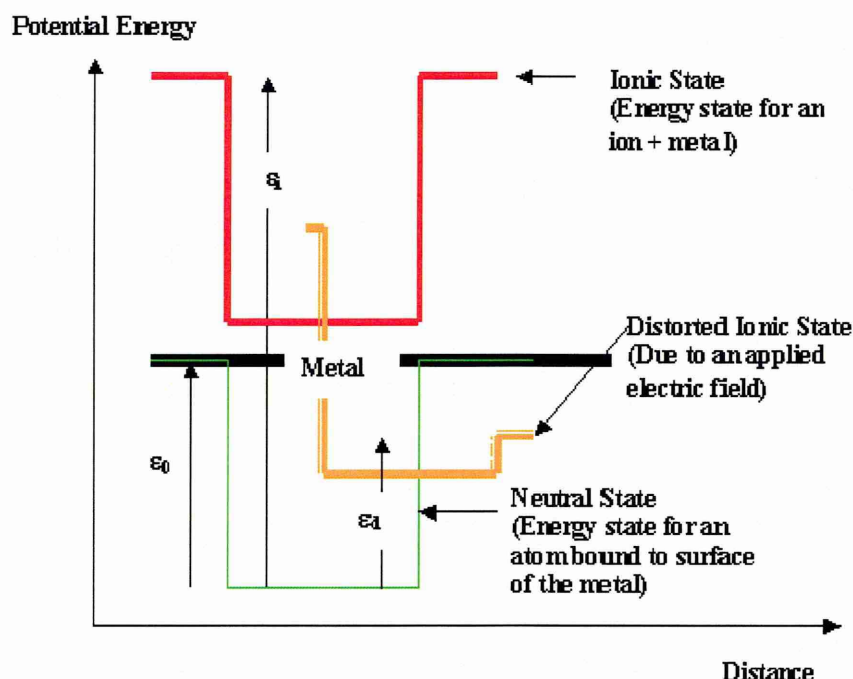


Figure 4-7 Potential diagram illustrating field-evaporation process. (Note. For illustration purpose only)

By limiting our discussion to low ion currents, as in the case of FEED, field desorption/evaporation will account for ion emission. Both Zandbergen and Gomer (Gomer, 1979) had done a more detailed work in this area, taking into account the reduced ion current due to the effect of space charges.

Space charge effects

An electrostatic effect occurs whenever there are more charge (positive or negative) of one kind in a given volume. This effect may become so large as to reverse the previous direction of the field and to greatly limit the amount of current flowing. Therefore due to space charge effects the process of ionisation is severely hampered and C. D. Child first observed this phenomenon as early as in 1911 (Stuhlinger, 1964).

Space charge effects become predominant when ion current exceeds $0.1\text{-}\mu\text{A}$ per emission site. Gomer (Gomer, 1979) had shown that the emission current ($i \geq 0.1\mu\text{A}$), is proportional to the applied voltage to the power $3/2$. If N represents the number of sites along the slit and A is a constant, Zandbergen showed that the current-voltage characteristics for the emitter was as follows (Zandbergen, 1986):

$$I = Ni = NA \left[\left(V / V_s \right)^{3/2} - 1 \right]$$

Equation 4-8

For a given emitter accelerator unit and a propellant, the ion emission can only happen when the applied electric field is higher than a certain value. This voltage is defined here as V_s , the starting voltage, which represents the 'onset' condition without the space charge effect. However due to the presence of space charge affects, the 'onset' condition, the liquid metal surface wave like deformation, occurs at a higher voltage than starting voltage. Furthermore, even though the liquid metal surface starts to deform, when the 'onset' condition had reached, in a wave like form, much higher voltages are required for ion emission. Zandbergen has explained that, this is due to the non-existence of 'ideal' Taylor cones, due to the presence of space charges at the vicinity of the tip. In principle, there will be a rounding-off effect at the apex. This rounding-off allows the equilibrium between the surface tension and the applied electric field stresses, to be maintained. Therefore the applied voltage, V , to achieve ion emission is much greater than the starting voltage, V_s .

Surface Tension

This is defined as the free surface energy per unit area. The existence of a free surface implies the presence of free surface energy, which equals the work that was done when the surface was formed. Total surface energy includes mechanical and thermal energy. Surface tension becomes important when solid boundaries of liquid surface are close together or when the surface separating two immiscible fluids has a very small radius of curvature (Massey B. S., 1968). In a channel with a very narrow diameter, the predominance of the surface tension force can completely cease the flow of a liquid. In FEEP thrusters, these surface tension forces act to prevent propellant spillage, when the thrusters are not in operation.

The molecules of a liquid have a tendency to remain as one assemblage of particles. This effect is called cohesion and is due to the molecular attraction of the liquid. Forces between the molecules of a fluid and the molecules of a solid boundary surface give rise to adhesions between the fluid and the boundary. If the forces of adhesion are greater than the forces of cohesion, the liquid molecules tend to crowd towards the solid surface. This increase in area of contact between liquid and solid is called the 'wetting' capability of a given liquid and a solid. Therefore the 'wetting' capability is not solely a property of liquid alone but a property of liquid and the solid surface. For all liquids, the surface tension decreases as the temperature rises and vanishes at critical temperature. It also reduces when impurities are present.

The need to achieve wetting without alloying or chemical reaction is an important factor when selecting a propellant and emitter-material for FEEP. It is important to provide smooth and clear surfaces for good wetting. If the emitter surface has any irregularities, disruption of the flow will be caused due to a volume change in the flow. The considered

LEO applications will expose FEEP thrusters to a comparatively oxygen rich environment. This may modify the thrusters' wetting conditions, if oxidation occurs on the inner surfaces. However it is possible to eliminate this oxidised film to a certain extent by selecting an appropriate propellant and/or using a more complex system. Alternative propellants with respect to FEEP were explained in detail in '*Fulmer Research Limited, UK*', additional propellant studies {section 2.5;(Stewart et al. 1980)}. It was shown here that solubility of oxygen in Caesium is about 20 PPM at 50°C, while solubility for alternative propellants used are several orders of magnitude less than that of Caesium. However the above report (in section 2.5) argues that alternative propellants are still attractive if the emitter material is of nickel or particularly molybdenum or Tungsten alloy.

Slit Emitter Performance

The liquid propellant at the emitter slit is held in position due to the capillary forces between the emitter channel and the propellant. When the applied electrical field strength increases the free liquid surface becomes unstable. Under the influence of the applied electric field and surface tension forces, perturbed spikes are formed on the free surface. These ideal cones are called 'Taylor cones'. In order to form the ideal 'Taylor' cones, it requires a cone-forming voltage, which is defined as the critical voltage, V_c , (Zandbergen, 1986).

$$V_c = 0.927 \left(\frac{2}{\epsilon_0} \right)^{1/2} (\sigma \delta)^{1/2}$$

Equation 4-9

Here, ϵ_0 is the dielectric constant ($=8.85 \times 10^{-12}$ F/m), σ is surface tension and δ thought of as form-factor, (defined by the spacing between electrode and accelerator). With the present emitter, such as that used at Centropazio, critical cone forming voltage for Cs is around 5.3-kV while for liquid Gallium about 16.8-kV. As the critical voltage does not guarantee the formation of cones at that voltage, a higher voltage would be required. The form factor used in Equation 4-9 is defined by Equation 4-10 (see Figure 4-8), (Zandbergen, 1986).

$$\delta = \sqrt{a^2 + b^2}$$

Equation 4-10

According to Equation 4-9 it is clear that critical cone forming voltage (V_c), decreases for decreasing values of electrode distance δ (reduced a and b). However there exists a minimum distance of electrode spacing for emitter-safe operation, without arcing. Present values (from Centropazio test bed) for the linear distance from the emitter to accelerator (a), and accelerator aperture ($2b$), are in the order of 0.6-mm and 4-mm respectively.

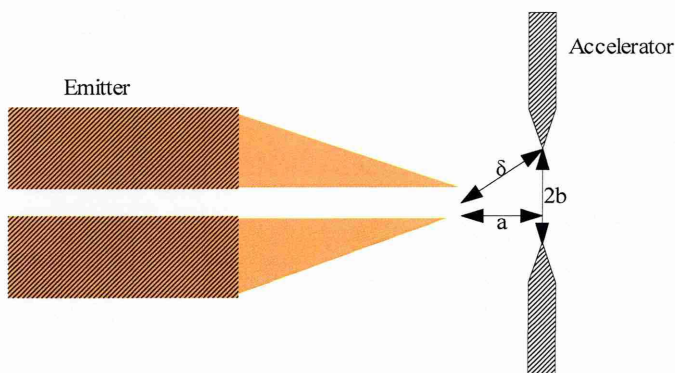


Figure 4-8 FEED emitter-accelerator spacing

It has been found that the thrust produced is directly proportional to the slit length. This is illustrated in Figure 4-9, for different slit lengths, using a constant accelerator voltage (-3-kV). Maximum thrust per unit length is in the order of 15 to 20 $\mu\text{N}/\text{mm}$.

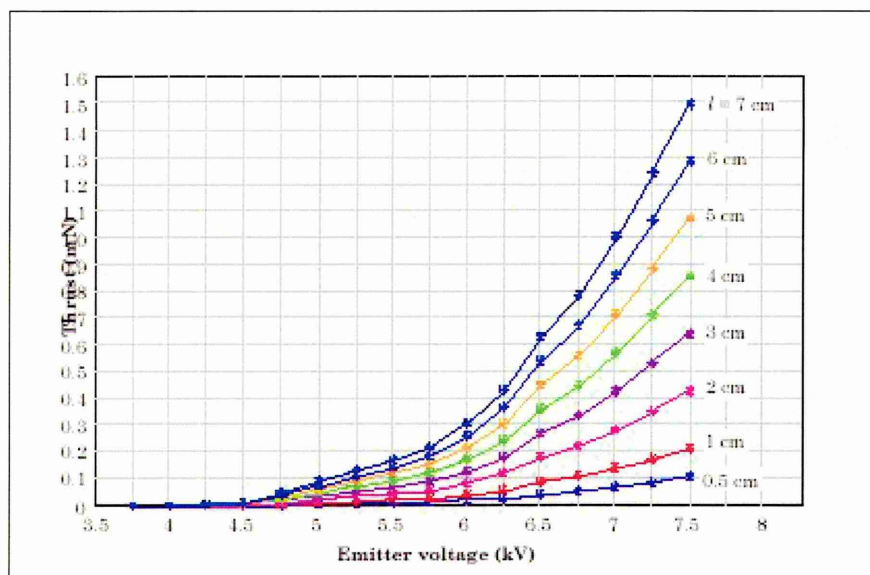


Figure 4-9 Thrust distributions (mN: Vertical axis) for varying emitter voltages (kV) using different slit lengths ($V_{\text{acc}} = -3$ kV). {S.Marcuccio, “Attitude & Orbit Control...”; (Marcuccio et al. 1997)}

With increased slit length, new emission sites will be introduced along the slit. Therefore increased slit length enables greater mass flow (new ion emission sites), which in turn produces increased thrust according to Equation 4-2. Zandbergen had shown, taking into account experimental results from Mitterauer, that the number of ion emission sites for a given emitter (constant slit length and width) is constant. If λ denotes the spacing between

two emission sites as shown in Figure 4-10, then the number of corresponding emission sites for a slit length of l is given as of Equation 4-11 below.

$$N = l / \lambda$$

Equation 4-11

In contrast to the slit-length, increased slit width will decrease the number of emission sites. The main reason for this is the increase in the emission site base area (see Figure 4-10 and Figure 4-11). As the emission site base area increases with increase slit width, the possible number of emission sites have to decrease for a given slit length. Although the number of sites decreases with increasing slit-width, the emission current increases. Mitterauer (Mitterauer, 1989) has explained this, associating the wavelength to the curvature of the liquid metal surface, which in return relate to the slit-width. According to Equation 4-11, an increase in wavelength decreases the number of emission sites but the area associated with each site increases (see Figure 4-11). Therefore the flow rate of the charged mass (current) tends to increase. Now the base area associated (λ^2) with each site increases proportionately to the square of the slit-width and the flow rate of the charged mass increases linearly to the slit-width.

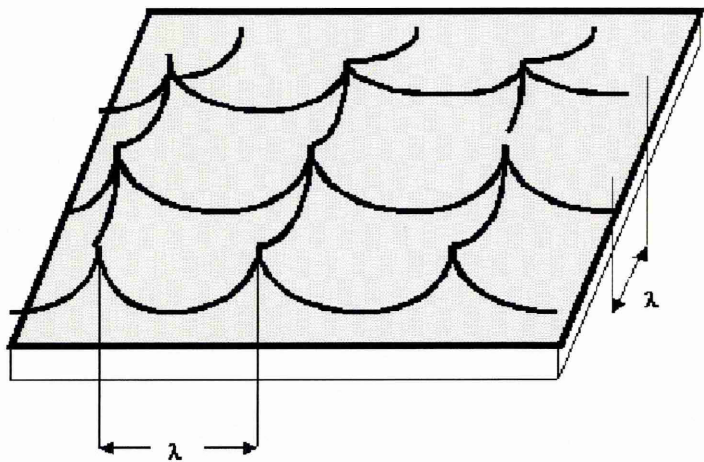


Figure 4-10 A square matrix pattern of cusp-like instabilities

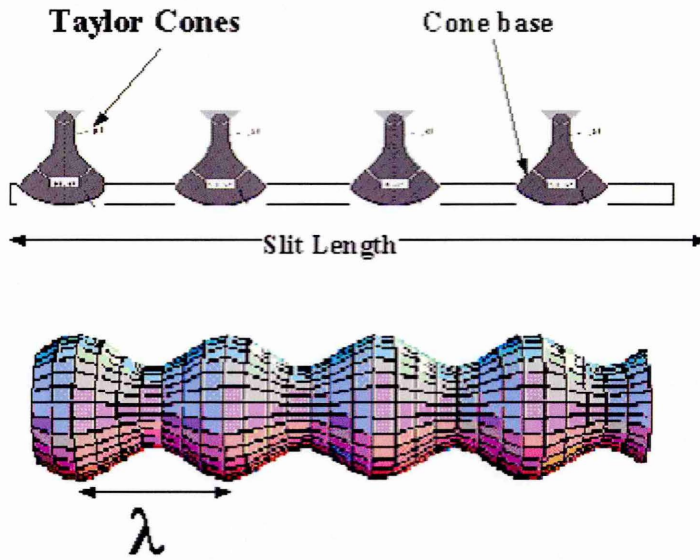


Figure 4-11 Liquid metal waveform instability and Taylor cone formation along the slit of a FEEP emitter

The above argument can be justified in the following way. For a given slit length (l) and width (t), number of emission sites (N) are given by:

$$N = \frac{l \times t}{\lambda^2}$$

Equation 4-12

Where,

λ : Spacing between two emission sites (see Figure 4-10)

It was explained earlier that slit length (l) is proportional to the number of emission sites (N) and the slit width (t) is proportional to the emission site base diameter or the spacing between two sites (λ).

Therefore for given slit length, increase in slit width (t) will increase the emission site base diameter (λ). According to Equation 4-12, a linear increase in slit width (t) will cause a square increase in base diameter λ . Thus the total number of emission sites (N) will decrease with increase in slit width (t).

However the mass efficiency will decrease with increasing slit-width as only ions contribute to thrust. Mass efficiency is defined as the ratio of the flow rate of the ion mass to the total mass flow rate. Bartoli (Bartoli et al. 1984) has measured mass efficiency of about 62% for a slit-width of 1.1 μm and 43% for a slit-width of 1.7 μm (both at 10 mA

current). The present values of slit-width, according to Centrosazio tests, are around 1.4 to 1.9 μm (at ESTEC: 1.2 μm and at Centrosazio: 1.4 μm) for optimal performance. The mass efficiency decreases with increasing voltage according to Equation 4-13.

$$\eta_m = \frac{\dot{m}^+}{\dot{m}} = \frac{1}{\sqrt{V}}$$

Equation 4-13

An increase in emitter length demands increase in power from the spacecraft. This is in good agreement with the increase in thrust in relation to the increase in emitter length. The power demand varies in linear relation to the thrust produced. As discussed earlier, FEEP requires high power per unit thrust. When calculated with modified thrust, power to thrust ratio is about 60 W/mN. With Caesium as the propellant, emission starts at a total voltage difference ($V_e - V_a$) of about 7 kV (see Figure 4-12). When emitter voltage increases beyond a certain limit, emitter power consumption will increase dramatically.

The specific power requirement (60 W/mN) for a given emitter drops by about 15% during low thrust applications (Genovese, 1995). As shown in Figure 4-12 when emitter voltage increases beyond about 6 kV (total voltage = 9 kV), emitter power requirement increases at a much higher rate. However according to Figure 4-9 the thrust production behaves in a similar manner beyond this transition voltage as expected. Therefore it should be the emitter current that increases at a much faster rate when passing this transition voltage (see Equation 4-4 & Equation 4-7). The increase in emitter current is in relation to the flow rate of the ion mass from the emitter. Taking into account (still much controversially argued) the actual mechanism of ion formation, one can conclude that ions are formed due to field ionisation (at least the dominant mechanism) instead of the much agreed field desorption. This is in good agreement with Gomer, who suggests that at high currents, field desorption becomes severely hampered by space charge effects, and that field ionisation seems to be a more likely theory. However, this conclusion needs to be investigated.

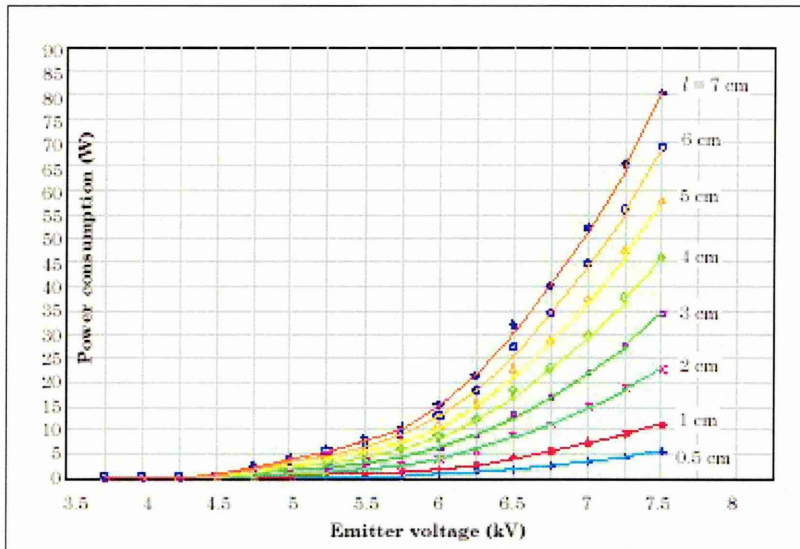


Figure 4-12 Emitter power (Watts: Vertical axis)) requirement for varying emitter voltages (kV) for a constant accelerator voltage, $V_a (= 3\text{kV}$, ref: (Marcuccio et al. 1997))

4.3 FEED Neutraliser

A parallel beam of charged particles will not be capable of progressing indefinitely into empty space. These particles will slow down in the vicinity of the field of space charge, and eventually reverse their direction of flow. The returning charged particles will enter into the thruster and will establish additional fields of space charge between the emitter slit and the accelerator electrode. These fields of space charges will counteract the original accelerating field, resulting in a small amount of ions leaving the thruster. As a net result the total thrust would be hopelessly small. Therefore charge neutralisation within the beam is an absolute necessity.

Ion beam neutralisation is a well-known technology. Generally hollow cathode neutralisers are widely used among ion thrusters. The ‘thermionic’ neutraliser that uses a hot filament to emit electrons is common with small electric thrusters such as those employing FEED and colloid principles. The power requirement in these types of neutralisers is quite considerable due to its heating element. Field emission from arrays of micro-tips, giving a more efficient neutraliser is being developed at Centrospazio (Marcuccio et al. 1997) and at many other organisations recently. These micro-tips can be generated from a metallic (Mo) or silicon substrate. When applied a potential difference (100V) between the substrate layer and the accelerator grid, the metallic substrate generated a greater electron density than Silicon (32 and 0.51 mA/mm^2 respectively) (Bartorelli, 1996-1997). The main disadvantage of the metallic substrate is however its high power requirement (0.12W/mA for Mo and 0.01W/mA for Si). A performance degradation of 10% had been recorded with the metallic substrate after 7000 hours of operation while with silicon it was not a

considerable amount. This concept is a further development of the earlier version of the 'Field Emission Electron' neutraliser (Valentian et al. 1989). The major development is to reduce the power requirement of the neutraliser to about 2% of the power demand of the FEEP thruster. The important features of the field emission neutraliser with respect to the conventional thermionic one, can be summarised as follows:

- No heat requirement as the source operates in principle similar to the FEEP thruster.
- The unit is extremely redundant as the electron beam is formed by a large number of emitting microtips ($\sim 10^4$ tips/mm²).
- Work in pulse mode as the ion source.
- Deliver an electron current density in the order of 1mA/mm².
- Low mass, simple and lower power requirement (Power 0.01 to 0.12 W/mA).
- The biggest disadvantages with this type are that a single short may eliminate the entire module and ion bombardment damage from plasma adjacent to thrusters.

In a laboratory scale test, the vacuum chamber walls provide the necessary beam neutralisation, as it is in ground potential. In order to operate a spacecraft with its sophisticated instruments, from a neutral platform in space, a neutraliser must be incorporated, to prevent accumulation of negative charges. A FEEP Neutraliser can be any source that emits enough electrons to achieve electrical neutrality. For ion thrusters the most common method used to produce neutralisation is hollow cathodes. The drawback of using thermionically emitting hot filament is that they consume very large amount of power (several Watts). Therefore it is not desirable as a FEEP neutralizer. In line with the SMART 2 mission ESA has contracted RAL to construct a field emission neutraliser with low mass, low power consumption and a long lifetime. The work started up in December in 2001 and has a deadline set in June 2003.

Review of Neutraliser technology

A theoretical study for a suitable FEEP neutraliser was performed, when RAL applied for the FEEP neutraliser contract. This study was then used as the base for further investigations. Things that were taken into account during these investigations were for example emitted current density, stability and lifetime, but also non-scientific issues such as supply and manufacturing. The selected technologies and their characteristics are given in Table 4-3 below.

Emitter Technology	Realistic emitted current density (mA/cm ²)	Emitted current stability (%)	Lifetime (Time to deteriorate to 50% of BOL) (hrs)	Power characteristic (W/mA)	Mass Characteristic (g/mA)	Voltage (V)
Silicon nanotips	160	1	100	0.001	0.002	100
AlN coated tips	<160	< 5	Unknown	0.001	0.003	200
Diamond-like-carbon coated tips	>500	1-10	100	0.1	0.0002	80
Nanocrystalline thin film diamond emitters	10	< 1	100	0.5	0.02	500

Table 4-3 The candidate FEEP neutraliser technologies and their characteristics (Credit Rutherford Appleton Laboratories, UK)

Silicon nanotips

This technology is a configuration of conductive silicon nanotips (n-type) that are set up in arrays as shown in Figure 4-13 (a) and (b). As seen in Figure 4-13 (b) the silicon field emitters (or tips) are placed in apertures. These are about 2 μm across and separated from each other with a distance of 10μm. The conic tips or spikes, Figure 4-13 (c), are by themselves about 2 μm high and have a diameter of 1μm at the base. The very top of the tip has a radius less than 10 nm. The small dimensions give a very high electron emission characteristic in response to an applied potential of about 100 V.

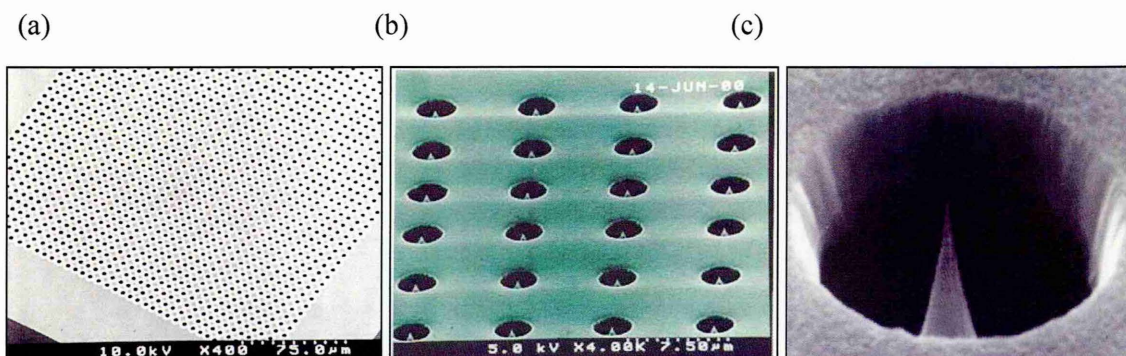


Figure 4-13 (a): 40 x 40 emitters chip. (b): Close up of the array showing the high uniformity used. (c): A single silicon emitter tip and its aperture only 2 μ m across.(courtesy of RAL)

The main advantage with this technology is that silicon tips have very low power consumption and very low mass characteristic. Although the emitting current fluctuation about $\pm 1\%$ or even more without any controlling device, this does not matter for neutralisation, provided that total emission capability is adequate.

Aluminium Nitride coated tips

Aluminium nitride (AlN) coated tips are silicon tips that have been coated with AlN. It has been shown (Zhirnov et al. 1997) that this kind of technology reduces the switch-on voltage by 33% compared with common silicon tips and also increases the current output. According to other experiments (Günther B. et al. 2001) AlN coated tips have a stability of $< 5\%$ even though there were bigger changes in stability between the different samples used. Further investigations regarding lifetime, stability etc, are therefore needed since no documented tests have been performed for long-term applications.

Diamond-like carbon coated tips

Diamond-like carbon (DLC) is an amorphous carbon mixture in which some proportion of the carbon atoms are bonded as in diamond and therefore gives the mixture some diamond-like qualities. The ratio between the graphite-like carbon and the more diamond-like carbon gives then the mixture its physical and electrical properties.

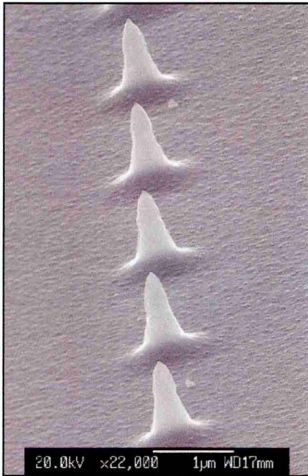


Figure 4-14 DLC coated silicon tips (Courtesy of RAL)

DLC coated silicon tips (Figure 4-14) have so far not been tested in any greater extent. The few test performed have so far however shown that they are more stable than uncoated tips (Jung J.H. et al. 1999). It has also been found (Zhirnov et al. 1997) that DLC coated tips have better emission properties such as reduced switch on voltage by 50% and higher current output (factor 3) compared with uncoated tips. As for AlN coated tips DLC coated technology needs to be tested further due to the small amount of documented experiment available. From the experimental tests performed so far at RAL, Central Microstructure Facility (CMF) current fluctuations as low as 3% at an emission current of 100µA has been demonstrated. These pre-study results give high hopes for this type of tips.

Nanocrystalline thin film diamond emitters

Nanocrystalline thin films (Figure 4-15) are constructed by letting 2.45 GHz microwave plasma deposit on a surface in an argon rich environment. Mirror-polished silicon is then used as a substrate, which has been treated with diamond powder, to promote diamond nucleation. The substrate is heated up to about 800 °C during this phase for deposition. Short-term emission tests have shown variation less than 1% during 3 hours. It has also been found that it is possible to achieve a current density of 10 mA/cm² and that the electron emission initiates at approximately 10V/µm.

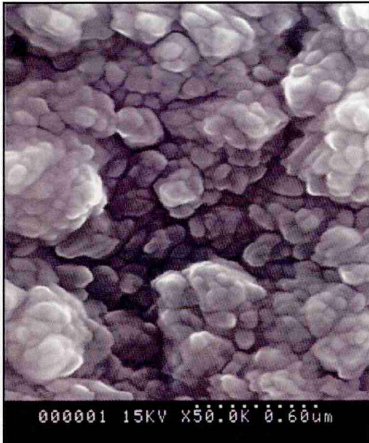


Figure 4-15 Micrograph of nanocrystalline diamonds (Courtesy of RAL)

4.4 FEED Propellants

Propellant Selection

The mass of propellant stored onboard determines the lifetime of the spacecraft for almost all space missions. In chemical rockets, the selected propellant should carry the required energy, while in electrical thrusters the propellant is merely a working fluid. When using electrostatic thrusters, the selected propellant should be able to ionise easily, prior to its acceleration. Ions could be generated in many different ways. The selected propellants' atomic properties decide the performance, and their nature determines the propulsion system complexity for electromagnetic & electrostatic thrusters. It is beneficial to employ a propellant with low chemical activity, which is easy to handle. This is especially important when safety is a critical factor. A chemically inactive propellant simplifies the thruster design and construction. Another important factor is the toxicity of the propellant. This is particularly important during the propellant handling, ground testing, contaminating the spacecraft instruments, environmental hazards etc. The need to ensure safety adds considerably to the development cost. For nano-satellites, when piggyback launch is of prime interest, long term storage capability and hazard-free storage are key factors for propellant selection. Failing to comply, one or the other may cause increase in the insurance premium, or missed launch opportunities.

Due to its very nature of ionisation, alkali metals are considered as best for the FEED thruster propellants. FEED thrusters have been operated successfully in the past using Caesium, Rubidium and Indium during ground testing. From them liquid Caesium (Cs) has been used widely due to its high atomic mass and good wetting properties with 'Inconel' as the emitter material. However a variety of propellants have been tested in combination with different emitter materials in search for alternative propellants (Wilson et al. 1989). Pure metals such as Gallium, Indium and some binary alloys have qualified during these

experiments (Zandbergen, 1986). However none of them had showed promising in performance as Caesium.

The need for liquid phase propellants imposes further demand for the FEEP operation and propellant selection. This implies, propellants with low melting points are advantageous, to reduce power requirement for melting. Caesium starts to solidify when temperatures fall below 28.5°C and the corresponding values for Rubidium is at 38.9 °C. This imposes a need for a heater to melt the propellant from its stored solid-state prior to use. This is especially true for those thrusters mounted at the far side of the s/c (shadow side), missions with long eclipse periods etc. In contrast to this argument, if the propellant temperatures exceed far beyond its melting point and evaporation starts, the thruster operation degrades due to non-homogeneity in the flow and losses due to evaporation. In general, 40°C was taken as the upper limit for Caesium propellant for a safe operation without the risk of propellant evaporation. The normal operating temperature for the FEEP thruster in the vacuum chamber is about 35°C.

The main concern with the use of alkali metal propellant such as Caesium and Rubidium is its interaction with environment gases such as water vapour, carbon dioxide, and molecular and atomic oxygen. If these metals, (Cs, Rb), get exposed to the environment, it will react with the environmental gases and produce crystalline substances with high melting points and may clog the narrow emitter slit. This imposes a severe penalty especially when using alkali metals in low earth orbit satellites as well as during the pre-launch and launch scenarios. In fact obstruction of the slit can be caused by interaction of the emitter blades with the atmosphere too, forming oxides. If contaminated, the emitter-slit gets obstructed, resulting in a poor performance (partial obstruction) leading to spotty ion emission, and eventually lead to total emitter failure, depending on the severity of the contamination. Therefore, if alkali metals (eg. Caesium, Rubidium) are to be used, then the operation should be limited to high altitude orbits. However as FEEP thrusters using Caesium have not been operated in any space mission up to now, the safe operational altitude is only a controversial guess. From the laboratory tests done at various institutions, altitudes over 400-km are thought of as safe to operate. Furthermore, as seen in laboratory test experience, Caesium showed increased resistance against contamination during operation (Genovese et al. 1997). Slit emitters have shown somewhat greater resiliency to contamination compared to pin emitters, as long as the ion emission occurs even over a short slit section, allowing to blown away the formed crust (Micci and Ketsdever, 2000).

Propellant Storage & Feeding

The built-in propellant tank in the FEEP emitter has a limited volume and can store only a small amount of propellant. The propellant will flow towards the slit due to the capillary forces and prevent spillage due to the surface tension forces. To enable the first wetting, when no electric field is present, capillary forces should overcome the flow impedance in

the channels. Therefore feeding channels should maintain a favoured geometry to create the capillary forces.

If the storage capacity is to be increased using the current system, then the emitter size and mass will not be in favour of a space application as present emitter materials ('Steel' or 'Inconel') are quite heavy. The current FEEP propellant reservoir has a convergent design. This special design helps to 'feed' the propellant towards the emitter slit. The emitter half (Figure 4-16) has two channels from reservoir to the emitter-lip to help the propellant flow to the emitter-lip.

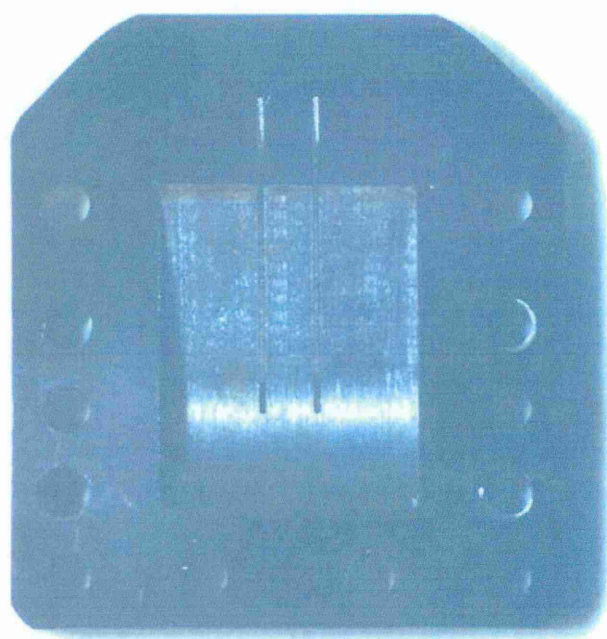


Figure 4-16 FEEP emitter half with built-in reservoir.

Figure 4-17 illustrates the possible ΔV , if the current SE20/1 emitter propellant tank scales linearly with the emitter length. The ΔV is calculated assuming that the total propellant mass is consumed in a drag free environment. This shows the propellant storage limitations and need to increase the emitter depth in order to provide increase storage facility. In order to guarantee the supply of propellant to the slit, it is argued to introduce channels as shown in Figure 4-16. However, according the tests done at Centropazio, these emitters have a higher failure risk due to the trapped particles in these channels especially during the lapping process. This process is a kind of 'sand blasting' to smoothen the emitter slit using fine diamond particles.

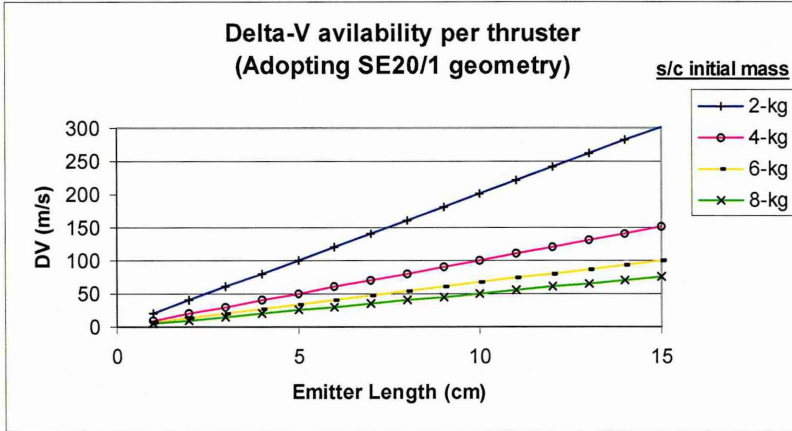


Figure 4-17 Total ΔV availability if FEEP SE 20/1 propellant storage scales linearly with the emitter length.

Those missions that last for several years, or when thrusters are operated frequently, the propellant requirement may increase beyond the current storage capacity. In order to meet this high propellant requirement, it is important to provide adequate storage volume as well as efficient feeding techniques. Introduction of pumps will certainly increase the mass and power requirement, which is undesirable for a nano-satellite design. Although a separate propellant storage tank with lighter material is in favour of this argument, capillary feeding still requires to be maintained to keep the essence of FEEP thrusters. This conclusion led the Centropazio FEEP team towards investigating capillary feeding effects using a central tank. During their preliminary investigation study, using Ethyl Alcohol, they were convinced that it is possible to build a separate propellant storage tank and still maintain the capillary feeding. The maximum size of the tank is governed by the critical capillary height achieved at the mission altitude. The capillary height, H , to which the capillary action will 'pump' the propellant is depend upon the weight of the propellant to which the surface tension, σ , will push (see Figure 4-18).

Balancing the forces gives:

$$2\pi s \sigma \cos \theta = \pi s^2 H \rho g$$

Hence the capillary height:

$$H = \frac{2 \cos \theta}{s} \frac{\sigma}{\rho g}$$

Equation 4-14

Where,

H : Capillary height

$2s$: Capillary slit width

σ : Surface tension

ρ : Liquid density

θ : Liquid / Solid contact angle (~ 10 -deg for Caesium).
 g : Local acceleration along the capillary tube (For stationary tubes on earth this is approximately 9.81 m/s^2).

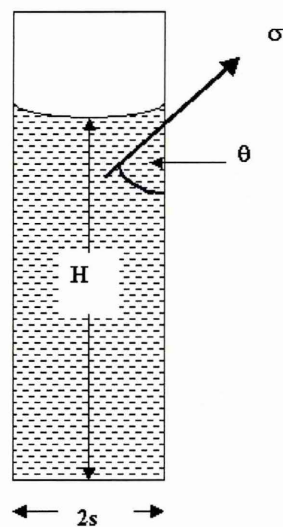


Figure 4-18 Capillary height

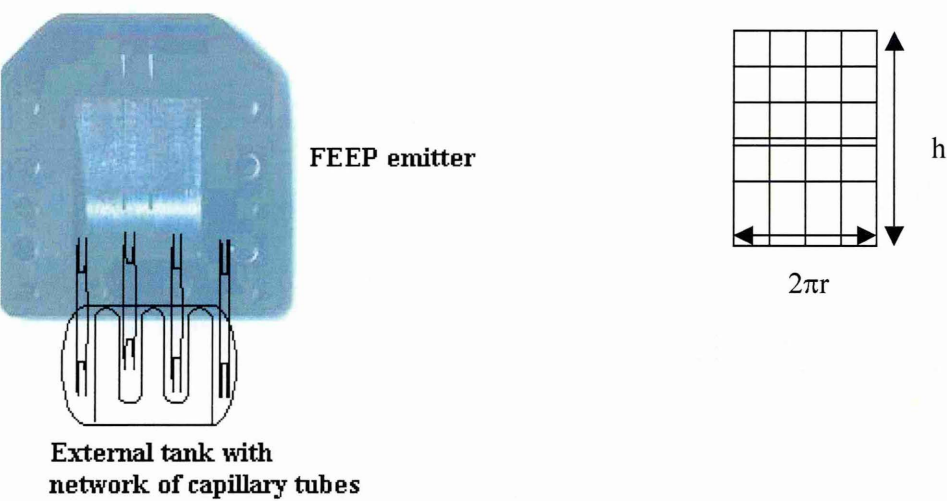


Figure 4-19 Emitter propellant feeding through an external and a cross-sectional cut of a propellant feeding tank with capillary channels.

According to the studies done at *Centrosazio* (Fabrizio, 1996), it was shown that it is possible to feed an emitter using a mesh of channels in the tank. The tank dimensions are constrained by the critical height calculated, using Equation 4-14. However there will always be a certain amount of propellant left in the tank, trapped between the mesh elements. Therefore, the finer the applied mesh, lesser will be the trapped propellant mass. A typical capillary network arrangement is shown in Figure 4-19, for a cylindrical tank. Here the capillary channels are distributed horizontally and vertically. Minimum size of the grid is decided by the propellant usage efficiency.

If a laminar flow model is assumed, it will be of interest to examine the flow velocities inside a channel. This is mainly important with respect to 'first' wetting of the slit when the propellant is fed from a central tank. Once the mass flow rate through the slit has been determined, propellant mean velocity can be calculated according to Equation 4-15.

$$u = \frac{\dot{m}}{(\rho A_c)}$$

Equation 4-15

Where,
 u = Mean velocity.
 ρ = Propellant density.
 A_c = Slit cross sectional area.

The time (t) it takes the propellant to run through the slit is given by;

$$t = d/u$$

Equation 4-16

Where,
 d = Length of the flow channel (slit depth)

Typical mass flow rate values for a 1.7 μm emitter slit thickness are around 2.9×10^{-8} kg/s. With Caesium propellant and for a slit-depth of 11 mm, slit-wetting time is in the order of 30 seconds. This shows that with increased channel lengths, appreciable transient times may occur.

4.5 Safe Operational Methods

The Field Emission Electric Propulsion (FEEP) concept will certainly play an important role in the future space missions as a fine pointing thruster to control attitude. This thruster could operate in both continuous as well as in pulse mode, with high reproducibility of the pulses. Either the emitter or accelerating electrode voltages can be pulsed to obtain the required pulses. According to the ESTEC test results (Valentian et al. 1989) with a 3-cm

long emitter, impulse bits less than 10^{-8} N.s. and minimum usable pulse length of 40 milliseconds have been recorded.

FEEP was shown to have irregularities during the emitter 'first wetting' according to the tests done at Centropazio. However, once started, it has been noticed that the thruster could be shut off for a prolonged period of time, and restarted without a problem. Therefore to ensure that the thruster functions properly while in orbit, the first wetting should be done in a controlled environment, prior to thruster integration with the spacecraft. This may require a complex format of handling. Therefore for the successful operation of this thruster the requirements would be state-of-the-art manufacturing, skilled handling and safe integration procedures. These may prove to be costly, leading to the delays of the missions.

The main possible reason for a malfunction of this thruster, may due to the emitter/propellant contamination as discussed before. The final 'lapping' process of the emitter slit, which is of vital importance for the emitter performance, may cause flow interruption due to the trapped 'diamond' particles. This indicates the importance of maintaining the thruster in a controlled environment from the point of manufacture to its operation. There are two types of contamination that must be removed from the emitter itself. One is to separate any oxide layer, if present, from the emitter surface. The second is to remove (to de-absorb) trapped gases and impurities. By introducing a surface cleaning method such as 'cleavage' {(Morrison, 1977); 4.1.2 "preparation of a clean surface"}, after the lapping procedure, foreign particles can be removed from the emitter surface. However, a simple heating method may provide a more efficient way of 'de-absorbing' all foreign species.

The main concern and probably the most severe problem with FEEP thrusters is the contamination of Caesium propellant with environmental gases. Although Caesium itself evaporates at about 35°C , its reaction products usually have a high melting point. Typical Caesium reaction products and their melting points are Caesium hydroxide (CsOH , 272°C), as a result of the interaction of Cs with water vapour, various caesium oxides (Cs_2O , 490°C ; CsO_2 , 433°C ; Cs_2O_2 , 594°C) due to Caesium reaction with oxygen, and Caesium carbonate (Cs_2CO_3 , 610°C) as a result of Cs interaction with carbon dioxide (Micci and Ketsdever, 2000). There are few methods under consideration to reduce the degree of propellant/emitter contamination. One such method discussed at *Centropazio, Italy* (Marcuccio, 1998b) is to use an emitter-container model with a lid as shown in Figure 4-20. During the pre-flight preparation the emitter is sealed (lid closed) in the container with an inert gas. One of the major disadvantages with this concept is the extra mass incorporated due to the assembly of the container. Further in a situation of the inert gas leakage, then the Caesium gets contaminated due to the exposure to atmosphere. Although the water vapour poses the most serious contamination problem, building up a considerable higher degree of reaction products in the emitter slit, it has shown that it is possible to blow out the oxide layer by applying a higher voltage difference (Genovese et al. 1997).

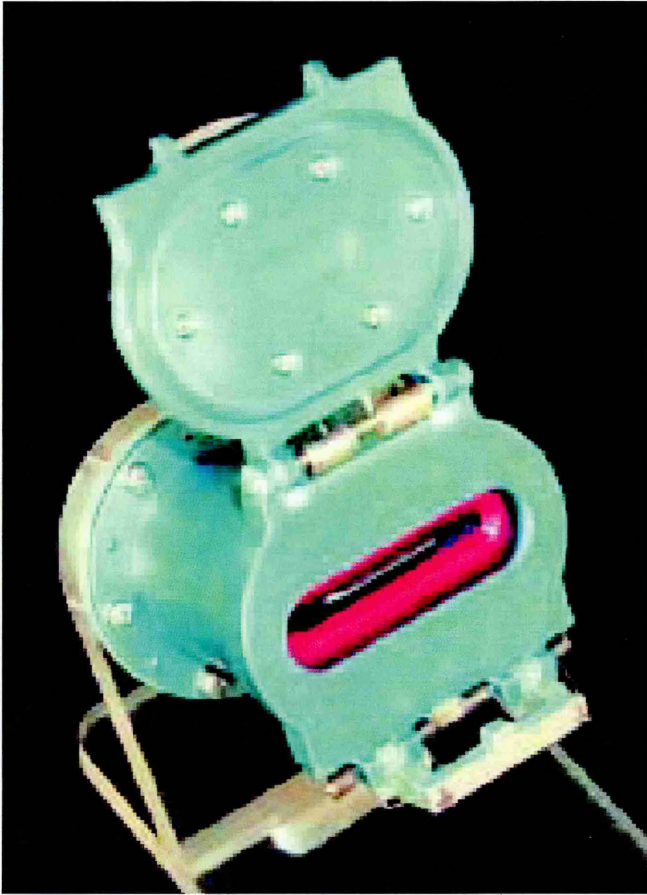


Figure 4-20 FEEP thruster protective container (Courtesy of Centropazio)

In order to develop a mass efficient system, it may be interesting to study a concept, which allows storing the propellant externally, in a sealed container as shown in Figure 4-21. These containers can be somewhat similar to that of the Caesium ampoule used for storing. While in orbit, these ampoules can be punctured using a needle. The propellant can be made to flow from the ampoule to the emitter, through the same needle that was used to break it. Inert gas pressure or capillary methods can be used to introduce the Caesium to the emitter. The Emitter temperature can be increased prior to the introduction of the propellant, allowing a process of 'baking' similar to that used in vacuum chambers. The primary disadvantage of this method is the possibility of environmental contamination of the emitter, during pre-launch and launch phase. However emitter baking can minimise these effects. Instead of individual containers, a single container can be used to a FEEP cluster. The container walls can be built with solar cells and the opening of the container can be used to provide a solar array. This way an extra power can be generated for FEEP operation. It should be noted any Caesium deposit on the solar cells would evaporate as the cells temperature would increase beyond that of the Caesium evaporation temperature.

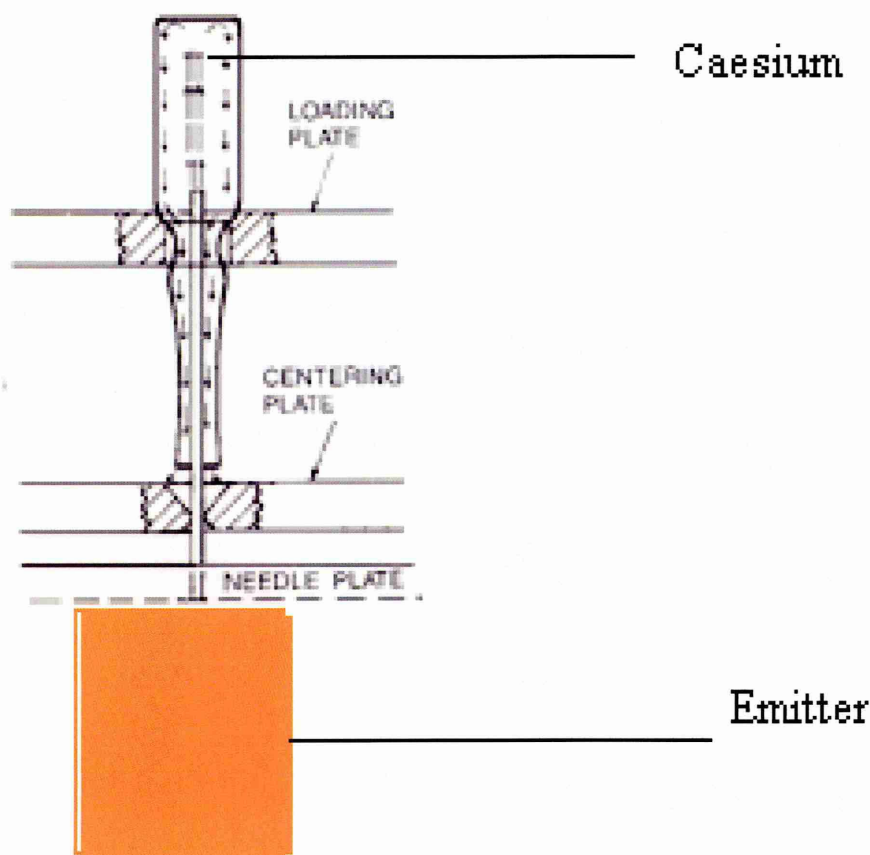


Figure 4-21 FEEP emitter with Caesium ampoule

According to the FEEP pulsed tests (5-Hz) carried out at Centropazio, the thrust output had shown good reproducibility. However at higher power levels it had shown an overshoot and then falling down gradually. The thruster had responded almost instantaneously for the applied voltage. As explained earlier, the specific power requirement increases during high thrust applications. The thruster power efficiency too increases with high power applications.

FEEP thruster performance characteristics are attractive for attitude control and fine pointing applications. Conventional (chemical) thrusters command the propellant flow to be proportional to the demand. For fine pointing and small attitude control manoeuvres, this would require extremely small propellant control valves and thrusters with degraded reliability. For an electrical thruster, such as FEEP, varying the applied voltage controls the thrust variation. Therefore by commanding the emitter voltage, which is highly reliable,

controls FEEP thrust. The FEEP thruster proportional and pulse mode operation capability makes it an attractive attitude control actuator.

4.6 Micro-pumping

The increasing interest in micro-fabricated fluidic devices has introduced many engineering challenges in developing predictive capabilities for single and multi-phase transport in micro passages. Scaling laws indicate that surface tension dominates over other forces, such as pressure or inertia with decrease in size. This scaling of surface tension force can be used favourably in the development of micro-devices in order to generate or control flow in micro-channels. The complications arising due to moving mechanical parts at micro-scale are eliminated in these devices.

Liquid handling and actuation by controlling surface tension has many advantages in micro scale applications because of the dominance and effectiveness of the surface tension forces. It has been well known for many years in the field of heat transfer and fluid mechanics that heat can be used to change surface tension (i.e., thermo-capillary) and flow can be induced as a result (e.g., Marangoni effect). It has also been known that electric potential can be used to change surface tension (i.e., electro-capillary). Using chemical methods such as use of surfactant can also change surface tension. However this is not of interest in this study. Taking into account the requirement for a heater for Caesium propellant liquefy, thermal method is more attractive and perhaps the easiest to implement. Thermo-capillary based micro-actuation uses micro fabricated heaters to create a temperature gradient in a column of liquid (or bubble), generating a surface tension gradient that causes liquid to flow. As shown in Figure 4-22 with proper current pulses to the heaters, the Caesium vapour bubble simply grows at one end (right side in the figure) and shrinks at the other end (left side in the figure). This causes the Caesium liquid to flow from left to right according to the figure. The initial bubble can be created either from liquid Caesium vapour itself or by introducing an inert gas similar to that needs to push the Caesium from its storage ampoule illustrated in Figure 4-21.

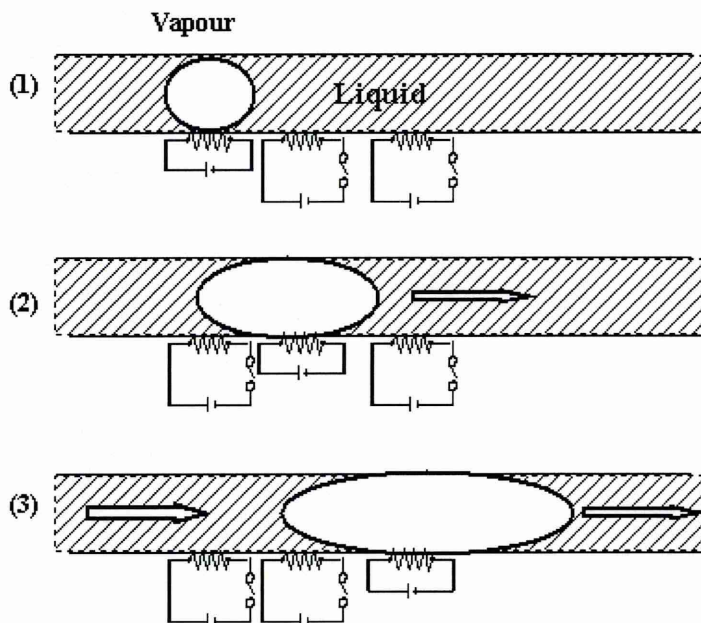


Figure 4-22 The thermal gradient is used pump the liquid Caesium by moving the vapour bubble (Chang-Jin. and Kim C. J., 2001).

4.7 FEEP Test Experience Summary

FEEP thruster testing takes place currently at Centrospazio, Italy as well as at ESA/ESTEC electrical propulsion facility in Holland. During the past thirty years FEEP technology has been extensively studied and it is presently undergoing a final technological validation at the above named two facilities. Both of these centres were visited during the research study, from 1998- 2001, to understand the test procedures and gain first hand knowledge of its working principals.

The author spent approximately three months at the Centrospazio test facility, in Pisa, Italy. This opportunity was granted through the ‘Erasmus Socrates’ project and was an immense help for the success of this study. During this time the author had the opportunity to observe the FEEP test procedures and to discuss the progress of this technology with the relevant experts. Much of the work at the test facility was centred on gaining knowledge about the test procedures and finding the relevant literature, mostly in form of internal reports. In this chapter the author takes the opportunity to give an in-sight of the Centrospazio facility and its test procedures. Later when it came to further development and trouble free operation of the FEEP thruster, the author had drawn conclusions from the observations done at the test facility together with the literature survey.

The Cranfield University awarded a scholarship for the author to visit the ESA/ESTEC FEEP test facility in Noordwijk, Holland, for a short period in early 2000. This visit was

combined together with the Satellite Tool Kit two-day seminar at ESTEC. This visit was mainly a fact-finding mission as during that time the FEEP thruster ground testing had fewer problems at ESTEC as compared to Centropazio. Further, this visit gave an opportunity to meet the electric propulsion and attitude control experts at ESTEC. With respect to the attitude control simulations, the author received important advise and help from Mr. Ton van Overbeek from the ESTEC, AOCS department. During this visit the author had a short discussion with Dr. Barry Zandbergen at the Technical University of Delft (TUD) with regard to the objective of this study. The comparisons and conclusions drawn in this thesis is a result of all these meetings and visits to these two main test sites.

Centropazio Test Facility

An outline description of the Centropazio test facility will be given here in order to provide a ‘mental picture’ of the FEEP ground testing procedures at the Centropazio test facility in Pisa, Italy. Centropazio is a space technology experimental laboratory under the Consorzio Pisa Ricerche (Universita delgi studi di Pisa, CNR, IRI). At Centropazio during 1999 three vacuum chamber facilities were available for electric propulsion tests. From these three, one facility (referred as facility-1 hereafter) was completely devoted to FEEP thruster testing. Even though facility-1 is somewhat smaller than the other two, it provided the best vacuum conditions needed for FEEP testing. The remaining two facilities were used for MPD thruster testing (conventional MPD’s and ‘Lorentz’ force plasma ejection MPD). Vacuum chamber facilities 1 & 2 are built of stainless steel vessels while the other is of fibreglass. The facility-2, which is the biggest, is of two cylindrical sections and facilities 1 & 3 are of single cylindrical sections. A summary of the three facilities is given in Table 4-4.

	Facility1 (FEEP)	Facility2 (MPD- Conventional)	Facility3 (MPD-Lorentz)
Diameter (m)	0.6	1.75 & 1.0	1.0
Length (m)	1.2	1.25 & 2.3	1.5
Volume (m ³)	0.3	4.0	1.2
Best Vacuum condition (milli-bar)	2.3 x10 ⁻⁸ to 2.3 x10 ⁻⁹	10 ⁻³ to 10 ⁻⁴	10 ⁻⁴ to 10 ⁻⁵

Table 4-4 Description of Centropazio test facility

At Centropazio all the test facilities as well as the vacuum pumps are located in a single room approximately 10 x 10 m² by area. Facility one is easily movable due to its size and facilities 2 & 3 are stationary.

The vacuum chamber for FEEP Testing

This chamber consist of three parts:

- The test chamber, which contains the thruster support.
- The emissions chamber where the emitted particles are caught.
- The flange lid, which closes the chamber with the experimental support and the thruster unit, attached to it.

The vacuum chamber is supported on a steel framework that is mounted on wheels to allow the unit to be flexible in movement. The same framework even supports the vacuum pumps. The test chamber is a cylindrical container of stainless steel with a length 1 meter and an outer diameter about 0.6 meters. A flange lid closes the rear side of the plant with 12 clamps. The lid is sealed to the plant using two Rubber O-rings. The emission chamber is also a cylindrical duct of stainless steel with the same dimensions as the test chamber. This is connected at the front using 48 clamps, while the rear part ends in a semi-elliptical cover, welded to the cylinder. The flange lid consist of the flange itself, closing the chamber by means of 12 clamps and of the support for the cryogenic and rotary pumps, as well for the mass spectrometer and the probes for high vacuum (Ionivac).

FEEP test Procedure

At Centropazio, Italy, the FEEP test facility has four vacuum pumps connected to it. Initially the rotary vacuum pump-1 reduces the chamber pressure to $\sim 10^{-1}$ milli-bar. Then the rotary pump-2 will be switch on to develop the turbo pump to its working condition (taking out the excess water droplets from the chamber). When the turbo pump starts the rotary pump will be shut down and the chamber pressure will be reduced to around 10^{-5} milli-bars. This procedure may take about 24 hrs or more. Finally the cryogenic pump will reduce the chamber pressure to its best possible value of around 5×10^{-8} milli-bars.

If the above pressure conditions were met, then the Caesium feeding system will fill with Argon gas (Inert gas) and the heat bands (see Figure 4-23) will be switch on for about 15 to 18 hrs to allow out-gassing of the chamber walls. In the mean time the cryogenic pump will keep running to reduce the pressure as the out-gassing procedure tends to increase the chamber pressure. This procedure may requires to repeat couple of times to make sure that all the water vapour is evaporated (actually to reduce the total pressure, so that all the trapped gases total partial pressure falls to 5×10^{-8} milli-bars).

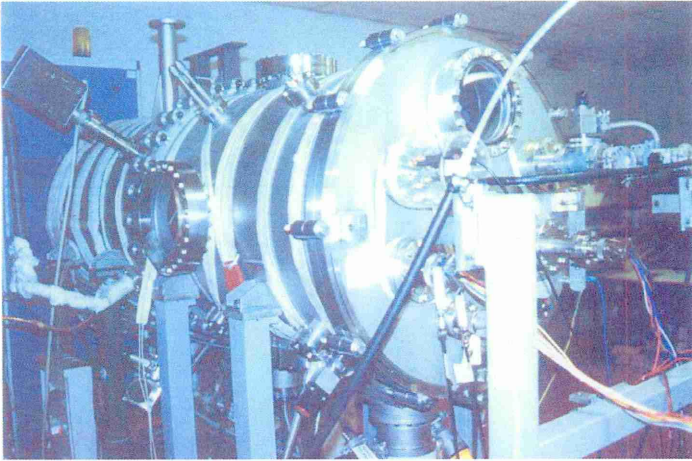


Figure 4-23 FEEP test facility at Centrospazio, Pisa, Italy, with heat bands wrapped around the chamber (white bands)

Finally the emitter ‘baking procedure’ will carry out to evaporate possible trapped water vapour from the emitter. A filament type heater as shown in Figure 4-24 is used for this task. This procedure may take about 12 to 15 hrs and during this time the emitter temperature will rise to about 350° C and overall chamber temperature a lot lower. When the chamber pressure falls down below 5×10^{-8} milli-bars, the system will allow cooling to about 30° to 40° C. This temperature range is important as a higher value may evaporate the Caesium and a lower value will solidify the Caesium.

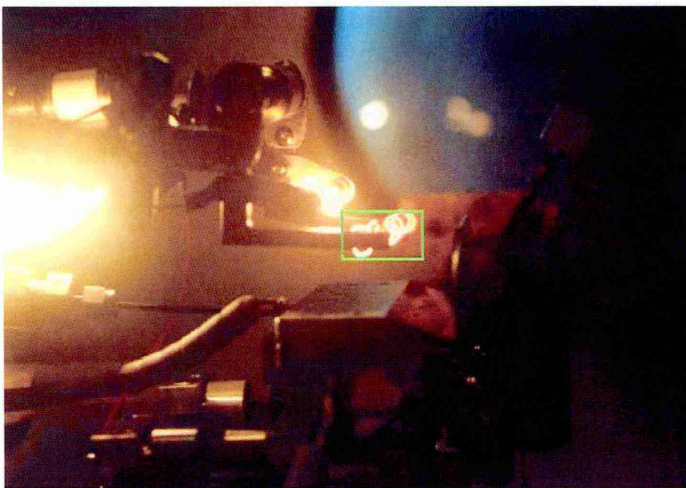


Figure 4-24 Filament type heater (box in middle)

From the test preparation to the commencement and continuation of the above procedures, it will take about 7-10 days. When the pressure and temperature in the vacuum chamber meets the satisfactory conditions, the propellant (Caesium) impeller breaks by a magnetic hammer. The Caesium is pressed gradually through the feeding system due to the argon gas

pressure. Once the Caesium has just arrived over the feeding system the pressurisation from Argon is stopped and the Caesium will be allow to fall over to the reservoir freely. In Figure 4-25 the Caesium feeding system is shown within the box at right top hand corner. As it is important not to spill Caesium, the above procedure requires a skilled operator to handle it (presently during this procedure 3 to 4 people are involved).

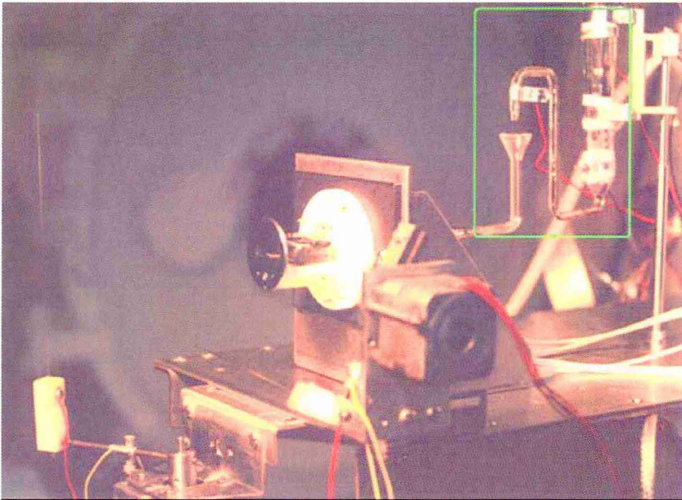


Figure 4-25 Caesium is pressed through the glass container at right over to the FEED emitter feeding system (the feeding system is shown in the box at right top corner).

Baking procedure

The emitter ‘baking’ procedure takes generally 15 to 18 hours. This procedure is important to allow out-gassing of all the trap air from the emitter. During the baking procedure the chamber pressure will rise due to the increasing temperature. At the end of the baking procedure, the emitter will allow to cool down slowly. During this time the chamber pressure will be reduced to required limits (5×10^{-8} milli bars). When the emitter temperature falls to about 40°C , the Caesium impeller will be broken to allow Caesium to flow under small Inert gas pressure to the Reservoir. After that, required voltages will be set for the emitter testing.

Although on a ground base test one can take all the above discussed test procedures this may hard to achieve on a real space mission. Beside the long test setting time, some of the following may find hard to achieve especially on a nano-satellite platform.

- Emitter baking temperature ~ 300 to 350°C
- Feeding system temperature $\sim 100^\circ\text{C}$
- Chamber pressure (water vapour) $< 5 \times 10^{-8}$ milli-bar

Emitter Attachment

The emitter halves are attached with screws (four screws for 5 mm slit type). On three sides of the one emitter half, 'Ni' is sputter deposited to ensure the required slit gap ($\sim 1.67 \mu\text{m}$). A high-resolution microscope is used to monitor if the emitter halves are in alignment (parallel edges) during the screwing procedure. If the slit misalignment is more than $10 \mu\text{m}$ along the slit length, the whole procedure will be repeated. For satisfactory operation the misalignment should be less than $8 \mu\text{m}$ along the length.

It is important to give some attention to the emitter halves attachment, as the precision of the system may deteriorate during its manual attach procedure of the emitter halves. Although microscopes are used to get the required precision, this may allow only focusing with high resolution to a single point, and therefore depict the emitter slit as a line. Precise engineering is required throughout the test procedure, to get the final accuracy.

The smoothness of the slit surface is achieved through a process called 'lapping'. The lapping procedure is an important part in the manufacturing of the emitter. This is similar to fine 'sand papering' of surfaces. In emitter lapping machines, very fine diamond particles in an oil bath is used to give the required precision. This process requires a skilled person to operate it, to provide the required high smoothness.

Neutralizer

During the FEEP tests the ionised beam will be neutralised by the vacuum chamber walls, as they are of ground potential. One of the negative effects of this is the metal sputtering of the wall. This will in return give rise to secondary electron emission from chamber walls. Therefore during the vacuum chamber tests the space charge effect on the FEEP ion beam is negligible due to effective neutralisation of the beam. However space charge effect is great concern during a FEEP mission as it can severely hampered the ion flow. Although thrust and beam current for a FEEP is much smaller than that of an Ion thrusters (eg. T5 thruster on Artemis: 10 cm beam diameter, 18 mN thrust, $\sim 370 \text{ mA}$), thrust and current densities at the source is much higher for FEEP thrusters. Like any other EP device, FEEP thrusters alleviate the spacecraft charging effects. Therefore in order to operate the FEEP system as well as the other S/C subsystems effective neutralisation of the ion beam is of prime importance.

ESA/ESTEC FEEP Test Facility

After visiting the Centropazio, it was of interest to visit the ESTEC FEEP test facility as well to get a first hand insight to their test procedures. Therefore a short visit was arranged and it was combined together with the STK conference at the ESTEC. Importantly the test facility (vacuum chamber) at ESTEC is much bigger than that at Centropazio. Further it

gave better vacuum conditions for FEEP testing. The ESTEC FEEP emitter slit opening had a different size and the purity of the Caesium propellant had some differences too due to different suppliers for these two facilities. Considering the FEEP thrusters unique propellant flow characteristics, it can be argued that the composition of the Caesium (purity) played a vital role during the initial start-up process. This was further strengthened as the FEEP thruster had often encountered initial start up problems at Centropazio, test facility. However, it should be noted that the FEEP thrusters and test conditions between the two facilities varied, and had the following distinct differences:

- Centropazio had a smaller vacuum chamber compared to the ESTEC.
- FEEP emitters had a different slit opening and had different suppliers
- Purity of Caesium was different and had different suppliers.

Test Results

Once the emitter and accelerator voltages were set and the required system conditions were satisfied, ion emission will take place. The presence of the ion beam can be verified by means of measuring the electrical characteristics (current and voltage) of the thruster's power supply lines. Independent to these measurements the strength of the ion beam is measured with the help of electrostatic wire probes. These probes are placed as shown in Figure 4-26 along the vertical and horizontal direction to the ion beam. The strength of the ion beam is measured as an electrical signal. Experimental test results for the 1 cm FEEP emitter with Caesium propellant is shown in Figure 4-27 and Figure 4-28. In this experiment varying the applied emitter voltage generated a pulse wave. The results are shown for 100 Hz.

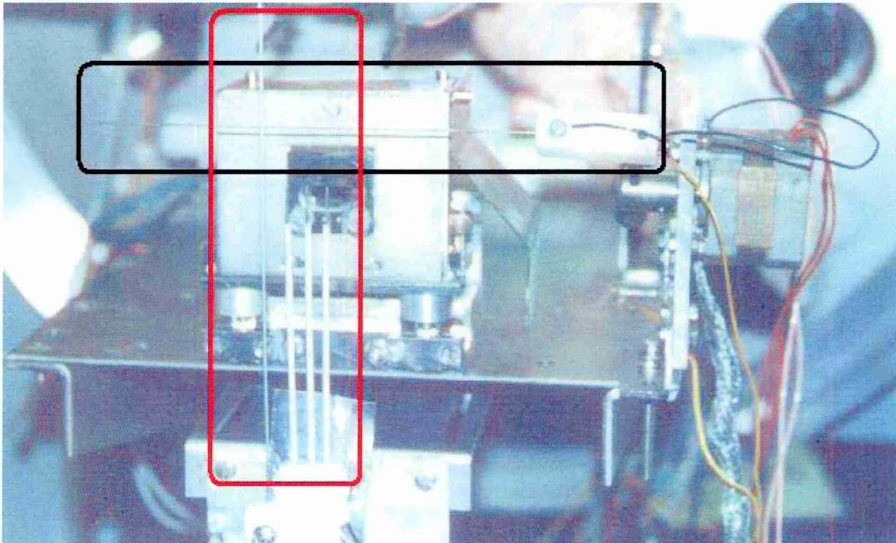


Figure 4-26 Vertical and horizontal probe locations (boxed in vertical and horizontally)

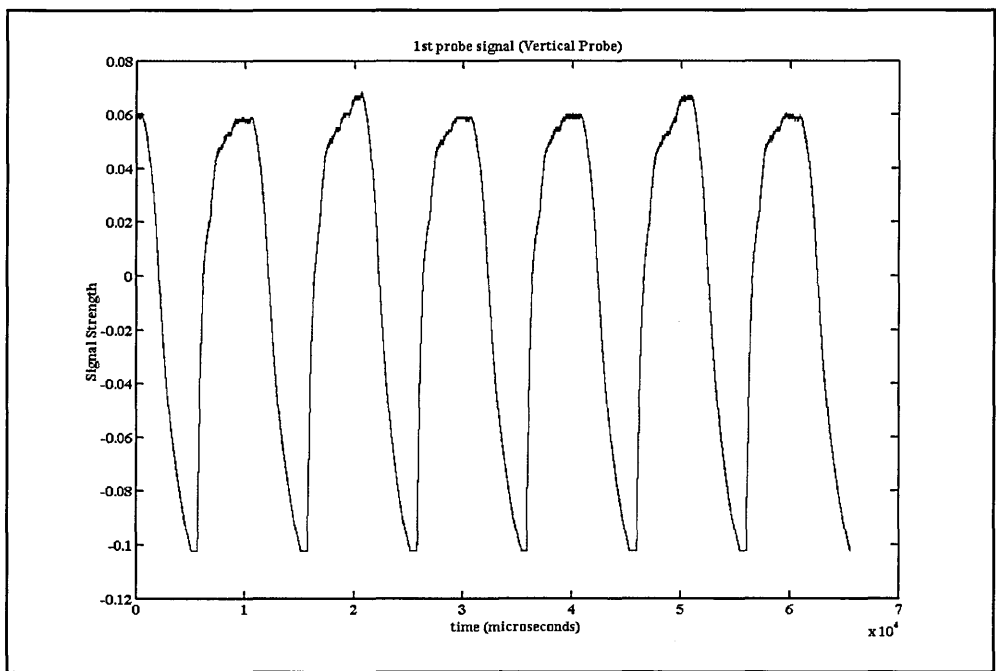


Figure 4-27 The measured vertical probe signal strength (normalized) for 100 Hz operational frequency.

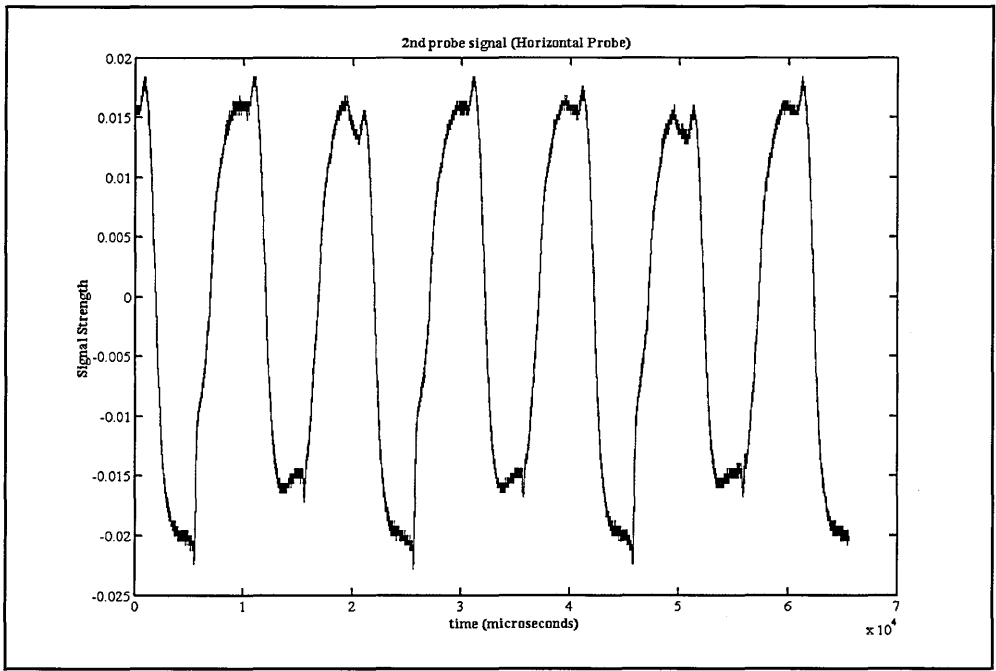


Figure 4-28 The measured horizontal probe signal strength (normalised) for 100 Hz operational frequency.

According to Figure 4-27 and Figure 4-28 the reproducibility of the signal is acceptable. However at the operational frequency of 100 Hz, each pulse with a length of 100 ms produces some fluctuations at its peak strength. This may have been caused due to the back scattering and/or the nature of the electronic components. Importantly secondary emission is one of the major error sources when the tests are confined to vacuum chambers. The secondary emissions are caused due to ion bombardment on chamber walls (wall material sputtering) and Caesium deposits on walls. Secondary emissions are also generated from the neutralizer as well as from the accelerator electrode bombardment.

Table 4-5 shows some experimental test results using ‘Rubidium’ (Rb) as the working propellant. The accelerator voltage (V_A) was held constant (-4 kV) and the emitter voltage (V_E) was varied. The emitter and accelerator currents (I_E and I_A respectively) and the chamber pressure were measured. For an ideal thruster with 100% efficiency, the ion flow should pass through the accelerator aperture without giving rise to an accelerator current flow. However in practise there will always be some amount current flow through the accelerator electrode. The thruster total power demand can be calculated according to the Equation 4-17. Here the power was calculated by measuring the voltage and current at the emitter and accelerator. According to the experimental data average power loss along the accelerator electrode is around 10% of the total power demand. As shown in Figure 4-29 and Figure 4-30 at low operating voltages (at low power), the accelerator power loss increases to as high as 30% of the total. Beside this there are losses taking place at various electronic components as well as during the ionisation and acceleration process. The power requirement will also depends on the type of propellant due to differences in ionisation energies. For example compared to Caesium, the Rubidium propellant requires a higher potential difference to initiate the ion emission, hence higher energy. From the possible FEEP propellants in the alkali metal group Caesium has the lowest power requirement.

$$P_{Tot} = I_E V_E + I_A V_A$$

Equation 4-17

V _E (kV)	I _E (μA)	I _A (μA)	V _A (kV)	Chamber Pressure (milli-bars)
5.4	5	3	-4	6.30 x 10 ⁻⁰⁸
5.5	5	3	-4	5.80 x 10 ⁻⁰⁸
5.6	20	6	-4	5.90 x 10 ⁻⁰⁸
5.7	26	7	-4	6.00 x 10 ⁻⁰⁸
5.8	32	8	-4	6.20 x 10 ⁻⁰⁸
5.9	50	11	-4	6.40 x 10 ⁻⁰⁸
6	57	13	-4	7.00 x 10 ⁻⁰⁸
6.1	77	15	-4	7.50 x 10 ⁻⁰⁸
6.2	105	18	-4	8.40 x 10 ⁻⁰⁸
6.3	118	21	-4	8.90 x 10 ⁻⁰⁸
6.4	160	25	-4	8.80 x 10 ⁻⁰⁸
6.5	181	29	-4	9.20 x 10 ⁻⁰⁸
6.6	200	32	-4	1.00 x 10 ⁻⁰⁷
6.7	279	39	-4	1.20 x 10 ⁻⁰⁷
6.8	309	46	-4	1.30 x 10 ⁻⁰⁷
6.9	345	51	-4	1.50 x 10 ⁻⁰⁷
7	410	57	-4	1.70 x 10 ⁻⁰⁷
7.1	447	65	-4	1.70 x 10 ⁻⁰⁷
7.2	478	70	-4	2.00 x 10 ⁻⁰⁷

Table 4-5 Experimental test results with Rubidium as the propellant (Credit:Centrosazio)

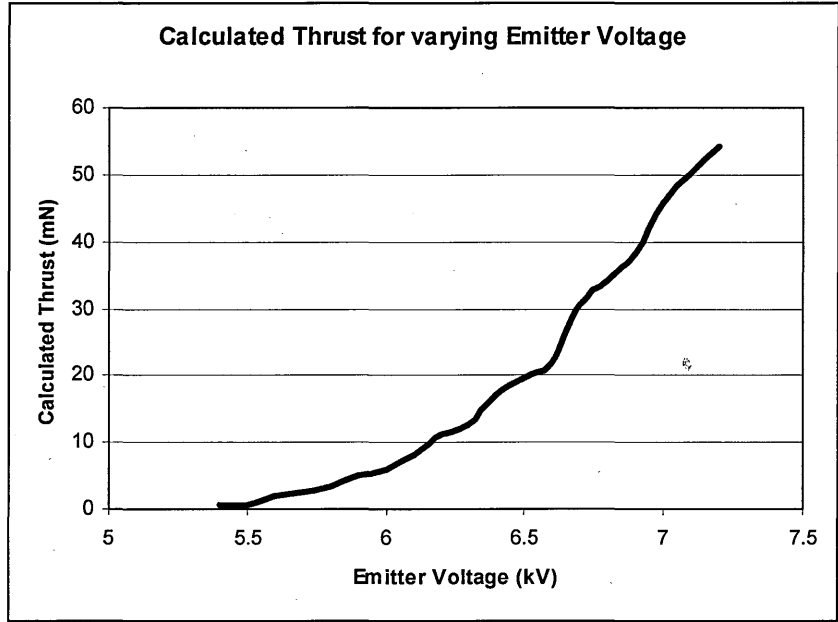


Figure 4-29 FEEP thruster test performance with Rubidium propellant.

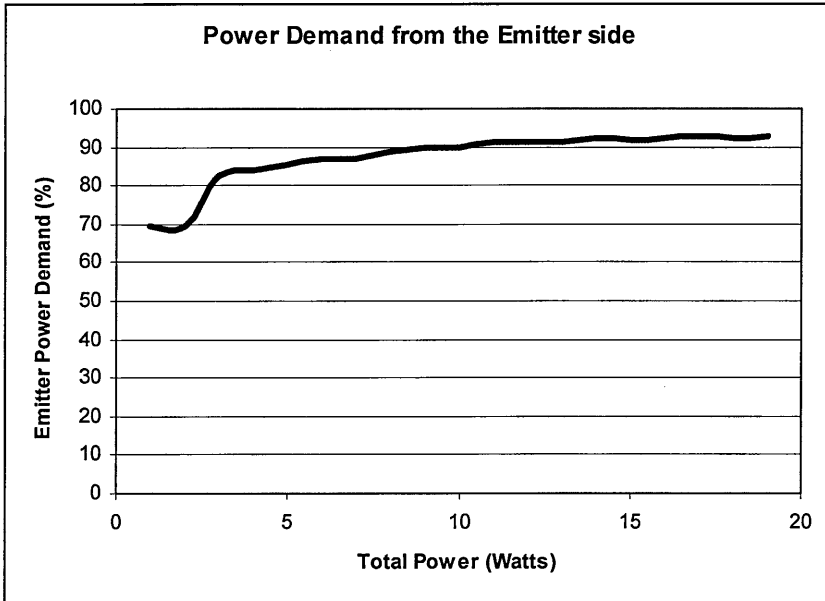


Figure 4-30 Percentage of emitter power demand with respect to total power (Total power = Emitter power + Accelerator power).

4.8 Summary

The development milestones of the FEEP slit technology were explained from the earlier pin emitter. The main components of the FEEP system were identified and discussed in brief. The FEEP thruster operational mechanism was explained to show the role of different components. Referring to the previous section Ion thruster technology, the FEEP thruster main difference is explained as its method of propellant ionisation. In FEEP system the propellant is ionised and accelerated by applying a very high electric field, in the order of 10^9 V/m, to the emitting propellant surfaces. It was explained that this is achieved by applying a high potential difference between two electrodes, which has a distance of few millimetres.

After studying different ionisation methods it was explained that the ion flow consist, mainly singly charged ions and to some extend ‘slow running’ propellant droplets too. Therefore by increasing the potential difference between the electrodes to achieve high thrust, the efficiency of the thrusters dropped. The FEEP thrusters associated flow divergence was explained taking into account the geometry of the slit and the space-charge effects.

Different slit emitters were compared with respect to its thrust and power demand and the FEEP-10 (1 cm slit) was selected for the FEEP demonstrator mission study. The main reason for this is the FEEP-10 low power demand compared to longer slits and its ability to provide adequate thrust (25 to 50 μN) even with partial slit contamination. The requirement for a FEEP neutraliser was explained and the current state-of-the-art neutraliser technology development was investigated.

From the available propellants Caesium was selected due to its low ionisation, low power demand and the data availability from its extensive ground testing. However the propellant contamination was identified as the most serious problem associated with this propellant. To overcome this problem, safe handling and operational techniques were discussed and its usefulness is yet to be identified. To reduce the initial start up problems and to transport the propellant from its safe storage tanks, a unique method, which uses micro heaters to assist the propellant flow, was discussed.

Due to the FEEP system high power demand and the requirement for high voltage components, FEEP thruster electrical power subsystem was investigated in a separate section.

5 Electric Power Subsystem

Small spacecraft power plant requirements and its capability to meet them are investigated in this section. Requirements are developed to meet the basic nano-satellite 'house keeping' and payload demands. It is assumed that the FEEP thrusters might operate either continuously or on demand. As a basic foundation for a possible case study, the main components of a simple spacecraft power subsystem were considered. Much of the emphasis is given to the power generation and storage requirement of a small spacecraft. Technology developments in the areas of solar arrays, batteries and power converters are mentioned. The selected FEEP emitter, FEEP-10, electrical characteristics were analysed.

5.1 General

Like any electrical device a spacecraft need electrical power to operate. Spacecraft subsystems, which consist of computers, radio transmitters, receivers, sensors, instruments etc, need electrical power to operate. For electric propulsion, thruster operation and performance is solely dependent on the electrical power availability. Therefore the electrical power subsystem (EPS) is a vital system that supplies electric power to different spacecraft subsystems. EPS should design to meet the spacecraft power requirement throughout its design lifetime. After a long journey into space and injection into orbit, a spacecraft requires electrical power under extreme conditions to operate. However, operation of the power subsystem may start much earlier than this. The batteries may require to be charged for testing of equipment during storage period. The operations of different subsystems in almost all spacecrafts rely on the electrical power availability. Fortunately the peak power requirements for different subsystems do not coincide, as they are not operating simultaneously. Depending on the mission and orbit, spacecraft power demand may vary considerably. Micro satellites such as UPM Sat-1 (47 kg, 20 W), Kitsat-3 (100 kg, 100 W) or Strv-1A (50 kg, 30.5 W) are equipped with electrical power systems that generate roughly 10 to 100 W (Wilson, 1996). The main components of a typical power subsystem are shown in Figure 5-1 below. The EPS system consists of power generating and storage sources such as solar arrays and batteries, power supply system, power conditioning and distribution unit. The power supply system includes battery charge and discharge regulators and a shunt regulator (to dump unwanted array power).

Almost all Earth orbiting satellites use solar cells to generate electrical power by converting the incident Sunlight. However the generated solar power, the currents and voltages, fluctuate considerably as the incident solar intensity varies. Further when the satellite passes through eclipses the solar power output will fall to zero. Therefore power system consists of batteries to compensate the power requirement. Different subsystems, which are rated for a certain plug in voltage, require power conditioning and control unit for fail free operation.

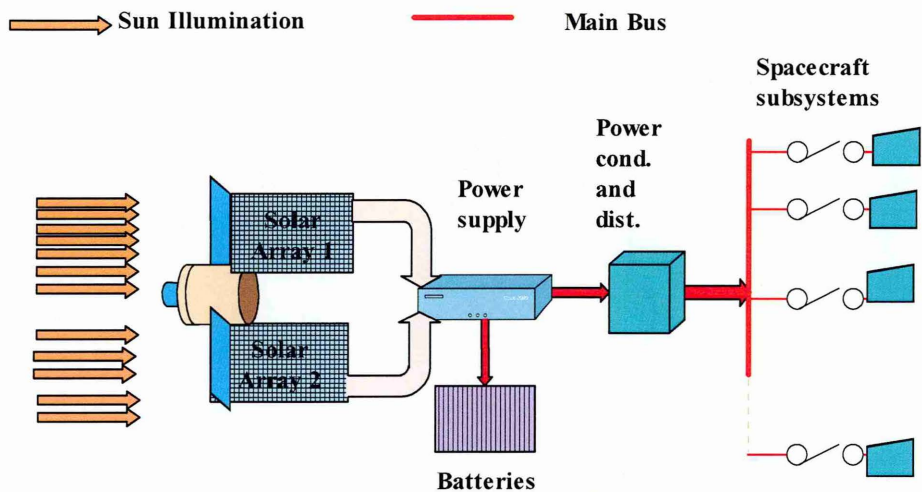


Figure 5-1 Block diagram of a simplified electrical power subsystem

Specific power values for few common thruster types are shown in Table 5-1. The specific electric power requirement per unit thrust is calculated according to {(Koelle and Koelle, 1961), equation-21.82} Equation 5-1.

$$P = \frac{I_{sp}}{\eta_e} g_e \times 10^{-3} \text{ W/mN}$$

Equation 5-1

Where, I_{sp} : Thruster specific impulse.

η_e : Energy conversion efficiency of the thrust device.

g_e : Gravitational acceleration at sea level

Propulsion system	Power/thrust (W/mN)	Isp (sec)	η_e
Resistojet	4- 5	280 - 305	60%
Arcjet	5 - 10	480 - 810	91% - 95%
Pulsed plasma thruster	10 - 15	830 - 1200	80 %
Ion bombardment	27 - 37	2500 - 3310	87% - 91%
Radio-frequency Ion	46 - 48	3000 - 3150	64%
FEED	35 - 100	3000 - 10000	80% - 98%

Table 5-1 Typical power to thrust ratio for some general electric propulsion devices (Moss, 1993), (Jordan, 2000) & (Danzmann et al. 1995).

5.2 Power Generation

Virtually every electrical or electronic component on a spacecraft may be switched on and off many times during the mission. This is accomplished using solid state or

mechanical relays that connect or disconnect the component from a common circuit, called the spacecraft main bus. Its computers, instruments and other sub systems demand precise and constant voltages for their operation. To do this the spacecraft electrical power source is typically designed to produce a slightly higher voltage than needed. The generated power is supplied to the main bus via a regulator to supply a constant voltage to the spacecraft instruments. The excess power dissipates as heat at the regulator. Typically a spacecraft electrical bus supplies a 28V DC voltage. However some instruments or subsystems may operate higher or lower voltage (eg. +5, ± 12 V dc) than this. Small, efficient power and voltage converters are available to supply the correct voltage and current to the attached systems (see Appendix-D.2). In a de-centralised system, converters are employed at each load end to meet different system requirements. This has the advantage that the EPS does not have to be tailor designed for different applications, which may simplify spacecraft design. However, selection of a standard bus type and component interfaces may be useful for nano-satellite design, development and modifications using a ‘modular’ approach.

Spacecraft with long mission times (from couple of weeks to years) are usually equipped with rechargeable batteries. The batteries are charged whenever the solar arrays are in sunlight. During the eclipse or at peak loads, the batteries supply power to the spacecraft main bus. There are many different types of batteries available for space missions today. From those Nickel-Cadmium (NiCd) and Li-Ion batteries (now the standard technology) are frequently used, especially for low cost missions. It is also a safe and economical choice for possible nano-satellite missions. NiCd and Li-Ion batteries can be discharged and charged thousands of times, and they also have a high energy density. Battery lifetime depends on how deeply it discharges and the total number of charge/discharge cycles it experiences. Li-ion batteries are now becoming popular due to their much-improved performance but at a higher cost.

5.3 Solar Array

A solar array consists of solar cells, inter-connectors and panels on which the solar cells are mounted, and for deploying arrays, deployment mechanisms etc. Crystalline ‘Silicon’ (Si) and crystalline ‘Gallium Arsenide’ (Ga-As) are two typical choices of photovoltaic materials for space applications. Photovoltaic material converts the incidence sunlight into electrical energy. The average energy emitted from surface of the Sun per unit time is around 3.85×10^{26} J/s, which is defined as the luminosity, L_0 . The solar constant ($= L_0/(4\pi r^2)$), which is defined as the solar luminosity per unit area at 1 AU has a value about 1367 W/m^2 . Annual solar constant variation is about $\pm 3.5\%$ and is due to the Earth’s elliptical orbit.

The sunlight incidence angle varies by about $\pm 23.5^\circ$ annually for an Earth orbiting equatorial satellite. This is due to the Earth and Sun plane tilt to the equatorial plane. The fraction of time that a spacecraft is in sunlight determines the longest time available for battery recharge. This in turn influences the solar array geometry and the depth of battery discharge. The maximum eclipse period determines the longest time that the battery must

sustain the load, a major factor in defining the battery capacity. The power from the solar cell has a maximum, when the angle of incidence is zero (normal to solar cell surface). The power decreases proportionately to the cosine of the angle of incidence. In flight, solar cells get quite warm, and therefore the solar arrays cement, and substrate have to be thermally conductive to help dissipate the heat. Prolonged exposure to high-energy particles causes photovoltaic performance to degrade in the neighbourhood of a percent or two per year. The cell degradation rate increases at high earth orbits in radiation belts due to exposure to trapped particles. Other problems that need to consider with the solar arrays are the 'hot spots' that cause due to partial shadowing of the array and hence degradation of power.

The spacecraft power generating technology has changed rapidly during the past 40 years to meet the current spacecraft power demands for both Earth orbiting and interplanetary missions. The solar cells have experienced almost a 240% (Society of Automotive Engineers, 1984) improvement in power per unit area during this period. For Earth orbiting missions *GaAs* cells are a better choice compared to *Si* cells. A summary of cell performance for low Earth orbiting missions is presented in Table 5-2. The UoSAT-12 mini-spacecraft Gallium Arsenide (*GaAs*) solar cells, generates about 227-watts of power per square meter (Marconi Applied Technologies Web Site, 1997). The power storage systems using batteries have experienced a 200% (Society of Automotive Engineers, 1984) improvement in power per unit weight. Although the cell development is constantly improving, only a small percentage of the total solar radiation is absorbed and converted to electricity with current technology. The remainder of the energy passes through the cell, reflect back to space or dissipates as heat. Present developments in multi junction cells appear to have increased power and mass efficiencies. A recent research study has recorded multi-junction *GaInP/GaAs* photovoltaic cells with efficiencies as high as 29.5% at 1-AU (Bertness et al. 1994). This is still leaving 35% of the sun's longer wavelength (Peak of the Suns radiation energy, about 65%, is concentrated in this wavelength range) energy uncollected. Additional improvements can be envisioned in ultra-thin solar cells to decrease weight and area to reduce overall cost.

	Efficiency @ 1 AU.	Status	Relative Cell Cost	Relative Array Cost	Array Power. (W) @ 1-AU, 1- m ²
Silicon	14-15%	Production	1	1	130-140
GaAs	18-19%	Production	5	1.6	160-227
GaAs/Ge Dual	22% Est.	Lab. Near Pilot	7	1.9	190-200
Advanced Silicon	20% Est.	Lab. Near Pilot	3	1.5	190-200
GaAs/GaSb	23% Est.	Lab	30	7	195-200
InP	17-18% Est.	Lab	40	9	150-160

Table 5-2 Photovoltaic cell performance & relative cost (*Si*) summary (Detwiler et al. 1995).

With the development of lightweight structures and new materials, electric propulsion for small satellites becomes attractive. As shown in Table 5-2 that ‘*GaAs*’ cells are attractive in terms of efficiency and relative low array cost, specially considered for small satellite applications. To date *GaAs* cells have been successfully used in many space applications. Cells having an area of 4 cm² under AM-0 (Air Mass Zero: In Space environment) conditions are able to produce 8 W (taking 15% efficiency) of power at 25° C (Meier et al. 1988). However cell weight (5.32 g/cm³) is a concern because cell thickness must be appreciable in order for the cell to be durable. In case of small satellites, reducing the array area due to high cell efficiency can compensate this. It would be desirable for nano-satellites to have body-mounted arrays, to avoid spacecraft design and operational complicity and to reduce mass. Table 5-3 summarises associated mass savings by using high specific mass (generated power per unit mass) cells for FEEP thruster applications.

Thrust (μN)	Power Requirement, W (Specific Power = 60W/mN)	Solar cell area (cm^2)		Cell Mass (grams)	
		Efficiency		Specific mass	
		18%	23%	15W/kg	45W/kg
1	0.06	2.5	2.0	4.0	1.34
10	0.6	24.5	19.0	40	13.4
50	3.0	122.0	95.5	200	67
1000	60.0	2438.5	1908.5	4000	1334
10000	600	24384.5	19083.5	40000	13334

Table 5-3 Characteristics of solar cell area and mass distribution for current and future missions.

Note: Calculations were based on taking effective area = 2/3 x Projected Area. Therefore for a 30x30x30-cm spacecraft has an effective solar array area of about 600-cm².

The solar cell efficiency degrades with time due to their susceptibility to particle radiation. The majority of cell degradation is due to the ionisation and atomic displacement. The electrons and protons trapped in the Earth’s magnetic field and electron belt in GTO are the principal causes of solar cell degradation at low earth orbits. The heart of the trapped particles, proton belt, lies at a distance of 2 to 3 Earth radii. Radiation damage due to the Earth proton belt needs attention, if the parking orbit is a GTO. However, this is only significant, if the spacecraft is parked for extensive periods of time in this orbit. Solar flares cause some additional damage during maximum solar activity period. The frequency of solar flare eruption increases to a maximum and then decreases during an 11 year long “solar cycle”. Generally the radiation damage is less important for a nano-satellite study, as it is a low earth orbit application, with short mission times. The main areas of consideration in this study are the particle radiation belt at the South Atlantic anomaly and material erosion effects due to atomic oxygen. The damage coefficient for a solar cell is dependent on the shield thickness of the front and back. The cover glass and adhesive in front and substrate and adhesive from back provide this required shield thickness.

5.4 Power Storage Cells

Two types of energy storage devices are taken into account here: primary and secondary batteries. Although the term battery is commonly used, sometimes it is referred to as cells, and a battery consists of one or more cells. Secondary cells are attractive as they can be recharged using a primary energy source (Solar array). Primary or secondary batteries are selected as power sources, according to the mission length and application.

When used as a secondary power source, the battery supplies power to the load during eclipse or whenever the load exceeds the solar array capability. The important properties of spacecraft batteries are long operational life (in extreme conditions), high energy density, low weight, and stability in a wide range of temperatures.

Primary batteries are mainly used for very short-term missions and applications such as firing pyrotechnic devices. They are attractive due to desirable properties such as good shelf life, high energy density, non-hazardous attributes and a wide range of operating temperatures. From the many different types of primary batteries 'Silver Zinc' and 'Lithium' cells have been widely used for space missions. Their main advantage is high-energy density, high discharge rate and fast response time. They have the lowest source impedance, providing the ability for a high discharge rate, limited mainly by the maximum heat dissipation rate. For small satellites with very short-missions (1-2 days) or as a memory backup, Lithium primary batteries have desirable features as shown in Table 5-4. In case of FEEP thruster demonstration mission, primary batteries could be useful. This is assuming that the FEEP thruster assembly is to fly as a payload, allowing its operation to be limited to few hours only.

	Silver-Zinc	Lithium Sulfur Dioxide	Lithium Carbon Dioxide	Lithium Thionyl Chloride
Energy density (W.hr/kg)	130	220	210	500
Energy density (W.hr/dm ³)	360	300	320	980
Operating temp.(°C)	0 to +40	-50 to +75	0 to +10	0 to 30
Storage temp (°C)	0 to +30	0 to +50	0 to +10	0 to 30
Storage life	30 to 90 days (wet) 5 years (dry)	10 years	2 years	5 years
Open circuit voltage, V/cell	1.6	3.0	3.0	3.6
Discharge voltage. V/cell	1.5	2.7	2.5	3.2
Manufacturer	Eagle-Picher, Yardney Technical Prod.	Honeywell, Power Conversion	Eagle-Picher	Duracell, Electrochem, Altus, ITT

Table 5-4 Characteristics of primary batteries for space use (Pisacane et al. 1994).

For most space applications secondary batteries have been the obvious choice. From the wide range of space qualified rechargeable batteries, 'Nickel-Cadmium' batteries have been widely used. For low Earth orbiting missions with frequent sunlight and eclipse periods, higher discharge and recharge rates are required. They also necessitate more sophisticated controlling electronics for battery charging.

	Silver Zinc (Ag-Zn)	Nickel Cadmium (Ni-Cd)	Nickel Hydrogen (Ni-H₂)
<i>Energy density (W. hr/kg)</i>	90	35	75
<i>Energy density (W. hr/dm³)</i>	245	90	60
<i>Operating temp. (°C)</i>	0 to 20	0 to 20	0 to 40
<i>Storage temp (°C)</i>	0 to 30	0 to 30	0 to 30
<i>Dry storage life</i>	5 years	5 years	5 years
<i>Wet storage life</i>	30 to 90 days	2 years	2 years
<i>Maximum cycle life (approx.)</i>	200	20000	20000
<i>Open circuit (V/cell)</i>	1.9	1.35	1.55
<i>Discharge (V/cell)</i>	1.8 to 1.5	1.25	1.25
<i>Charge (V/cell)</i>	2.0	1.45	1.5
<i>Manufacturer</i>	Eagle-Picher, Yardney Technical Products	Eagle-Picher, Gates Aerospace Batteries	Eagle-Picher, Yardney Tech. Prod. , Gates Aerospace Batteries, Hughes

Table 5-5 Basic characteristics of typical s/c secondary cells (Pisacane et al. 1994).

A comparison study was performed taking three types of secondary cells and the results are summarised in Table 5-5. According to this study 'Nickel Hydrogen' batteries are attractive mainly because of their high energy density and extended operating temperature range. However they are more cumbersome than other batteries and especially true of space-qualified batteries. For small satellites with limited mass and volume 'NiCd' cells may provide a suitable option for current missions. These cells can be packed into many different configurations to suite the specific design. Generally in space missions, batteries are stored in a metal casing to avoid possible contamination hazards during battery charge. Selection of batteries requires a prior knowledge of power requirement, length of the mission, type of orbit etc.

For illustration purposes, it was considered a mission with a continuous power requirement of 10 W. If the mission last for 6 hours and the above power demand to be meet with batteries only, then:

Battery energy requirement = 10 W x 6 hr. = 60 W.hr.

With a DOD of 90% and 95% transmission efficiency, required battery capacity, C_r :

$$C_r = \frac{60 \text{ W.hr.}}{90\% \times 95\%} = 70.2 \text{ W.hr.}$$

If use NiCd batteries that have an energy density of 35 W.hr/kg, average mass of the battery pack will be 2 kg. With increase power demand the battery mass will increase proportionately. However, if used Lithium-Ion batteries with an average energy density of 140 W.hr/kg, battery mass will reduce to about 0.5 kg to meet the set power demand. As shown in Figure 5-2 use of Lithium-Ion batteries compared to conventional NiCd batteries will reduce the system mass and volume.

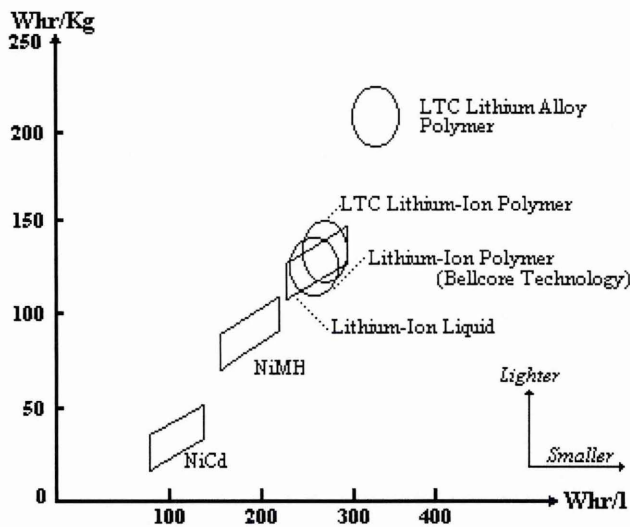


Figure 5-2 The technology development in Lithium alloy Polymer batteries (Detwiler et al. 1995) & (Lithium Technology Corporation Web Site, 2000).

As shown in Figure 5-2, secondary lithium alloy polymer batteries have several intrinsic and potential advantages including higher energy density, longer active shelf life and lower self-discharge over conventional *Ni-Cd* and *Ni-H₂* batteries. Successful development of these batteries will yield success in small satellite power storage development as well as in future interplanetary missions. However, as this technology is still in the stage of laboratory testing, near term applications are doubtful. For nano-satellites with minimum redundancies, application of complete new technologies may endanger the mission or cause costly developmental delays. On the other hand nano-satellites will pave the way for 'low cost space system tests'.

For a FEEP demonstrator mission using 'Mustang' as a platform, mission duration of one month may of interest. This will allow adequate time to perform the control manoeuvres and to allow enough time to validate the test data. For an 800 km circular orbit, it requires at least 0.06 m² solar panel area to meet the specification given in Table 5-6 below.

However, the solar panel must over sized in order to meet the losses and allow battery charging and payload demand simultaneously. This can be easily met with the ‘Mustang’ model, which has a projected body area of 0.09 m².

Orbit	800km
Power demand	10W continuous
Mission duration	30 days
Worst case sunlight angle	30 deg
Battery type	NiCd
DOD	80%
Transmission efficiency	95%
Solar panel efficiency	25%

Table 5-6 Typical mission specifications for solar panel sizing

5.5 Power and Voltage Converters

The spacecraft power distribution system supplies power to its main bus. Different subsystems (loads) that are connected to the main bus may require different power levels, low or high voltage etc, from the main bus. Power converters are electronic units that are fed by a direct current (the "mains supply" of the spacecraft) typically at 28 V dc. They connect different loads to the spacecraft main bus. These converters typically isolate the load from the bus and provide on-off control to the desired load. In case of a FEEP thruster demonstration mission, thrusters may demand an emitter voltage around 5 kV from the spacecraft bus, which is at 28 V dc. The applied converters must provide this low-to-high dc conversion and need to isolate the bus from possible thruster noise. In case of thruster failure and short-circuiting, it should protect the main bus to avoid any power losses.

When selecting power or voltage converters it is important to consider the power efficiency of the system to reduce losses while achieving reliability, overloading survivability and cost. For nano-satellites solid-state converters are attractive in terms of its small size and low mass. For illustration purpose Table 5-7 shows the IRF, Sweden (see Appendix-D) developed 28 V to 3 KV voltage converter general specifications. This has been space qualified and is possible to further develop to meet the FEEP thruster high voltage demands.

Input voltage	28 V, dc
Output voltage	3 kV
Weight	50 grams
Size	8 x 5 x 2 cm
Maximum switching frequency	30 kHz

Table 5-7 IRF high voltage converter current specifications (Courtesy of IRF, Sweden)

5.6 FEEP-10 Electrical Characteristics

In order to succeed as a nano-satellite propulsion system, it is important that the power demand of the FEEP thrusters is within the limits of nano-satellite power budget. Compared with the conventional electrical thrusters, FEEP has the highest power requirement per unit thrust (approximately 65 W per 1 mN thrust). Nano-satellites with its small surface area and mass are highly restricted for power generation and storing. For an application of FEEP thrusters together with reaction wheels the system mass and power demand will further increase. Although the FEEP power requirement per unit thrust is high, most of the attitude and orbit control requires a very small thrust. Therefore FEEP thrusters can be operated in conjunction with nano-satellites within their power budget. For successful application of FEEP thruster with respect to nano-satellites, it is important to restrict the number of operating thrusters at a given time to be within the limits of power demand and availability. Therefore the thrust manoeuvres needs to be organised to meet the control demand as well as the power availability.

The attitude control accuracy of FEEP thrusters depends on its minimum impulse bit and the frequency of the control pulses. FEEP thrusters had ground tested in pulse mode up to about 50 Hz at ‘Centrosazio’ test facility. However at low frequencies ion emission becomes stable as it gives enough time for the ion beam to settle down. With the current technology in electronics and power systems it is possible to achieve operating frequencies as high as 50 Hz. For the FEEP demonstration mission it is identified a frequency less than 0.1 Hz for most of the control modes. Between two on-times, the attitude control system operates in open loop and the S/C attitude will change according to the disturbance torques. Attitude control accuracy of FEEP thrusters can be calculated for a thruster pulse as given shown in Figure 5-3.

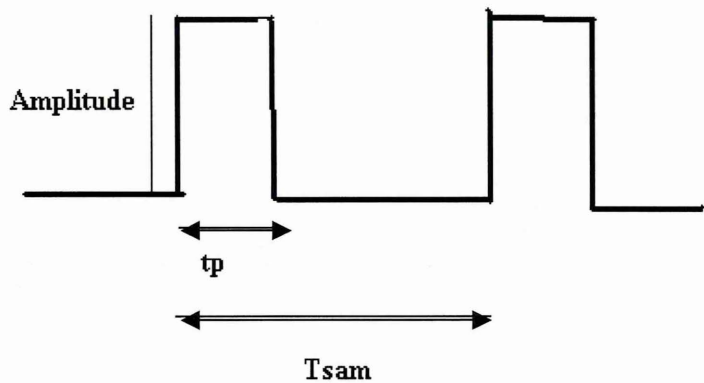


Figure 5-3 Thrust Pulse wave form

The simplified torque equation can be written as

$$T = I \ddot{\theta}$$

Equation 5-2

The minimum impulse bit that the thrusters can deliver to the satellite is thrust, F , multiplied by the minimum pulse duration, t_p . Hence the minimum torque bit is the minimum impulse bit multiplied by the torque arm, L . During the period that the torque exists, angular velocity will increase by

$$\Delta\dot{\theta} = \frac{FLt_p}{I}$$

Equation 5-3

Until the next sampling time the attitude will increase by

$$\Delta\theta = \frac{FLt_p}{I} (T_{sam} - t_p) = \frac{\tau}{I} (T_{sam} - t_p)$$

Equation 5-4

Where;

- θ : Rotational angle
- τ : Minimum torque impulse bit ($F \times L \times t_p$)
- L : Moment arm length
- F : Thrust amplitude
- Δt : Pulse width duration
- I : Moment of inertia about the axis of rotation
- T_{sam} : Sampling time (T_{sam}) = Period
- t_p : Minimum pulse duration

According to Figure 5-4 with a sampling frequency of 0.1 Hz, thruster pulse duration of 1 second and a minimum torque impulse bit of 1.5 $\mu\text{N.m.s}$, control accuracy about 0.01 deg (moment of inertia = 0.1 kg.m^2) can be achieved. The power requirement to provide this control accuracy will be as little as 650 mW. However it should be noted that FEEP thrusters are capable of delivering 'impulse bit' as low as 10^{-8} N.s. In this case the accuracy will increase by a factor of 100 to 1000 of what was predicted here.

Varying the voltage to the emitter, accelerator or both, controls FEEP thruster operation. In order to keep the regulator electronics and power supply unit simple, it is advantageous to keep the accelerator voltage at a constant level and to vary the emitter voltage according to the control demand. With the availability of multiple thrusters, eight thrusters in this case, the AOCS 'controller' must select the best possible thruster (or thrusters) pair and the associated beam neutralizer. In order to meet high attitude control accuracy, it requires a higher operating frequency. This in turn requires all thrusters to be in 'standby mode' at all times, in order to operate when demanded. Further to maintain the Caesium propellant in the liquid phase, the propellant temperature must be within 28°C to 35°C. This requires propellant temperature control system to operate without interruption. In order to avoid temperature above 35°C, the FEEP thrusters should be in a

temperature-controlled environment. Therefore in order to achieve high control accuracies, the thrusters need continuous power supply to the system.

Electrical characteristics of the FEEP thrusters were analysed for an application in a ‘disturbance rejection’ mode. Though each individual FEEP thruster has a unique electrical characteristic, the analysis was performed here using the test results of the ESA/ESTEC FEEP-10 emitter. In Figure 5-4(a), test results from the ESTEC vacuum chamber for FEEP-10 thruster is shown. Using polynomial curve fitting method, a relationship for the emitter voltage and current was derived and plotted as shown in Figure 5-4(b). It shows that the selected polynomial equation predicts higher emitter current when emitter voltages are above 6 kV. However for operational voltages below 6 kV, the two curves match fairly well and hence the Equation 5-5 predicts the current voltage relation.

$$I_e = 3.25521 \times 10^{-3} \times V_e^3 + 0.042187484 \times V_e^2 - 0.1008333 \times V_e - 0.006$$

Equation 5-5

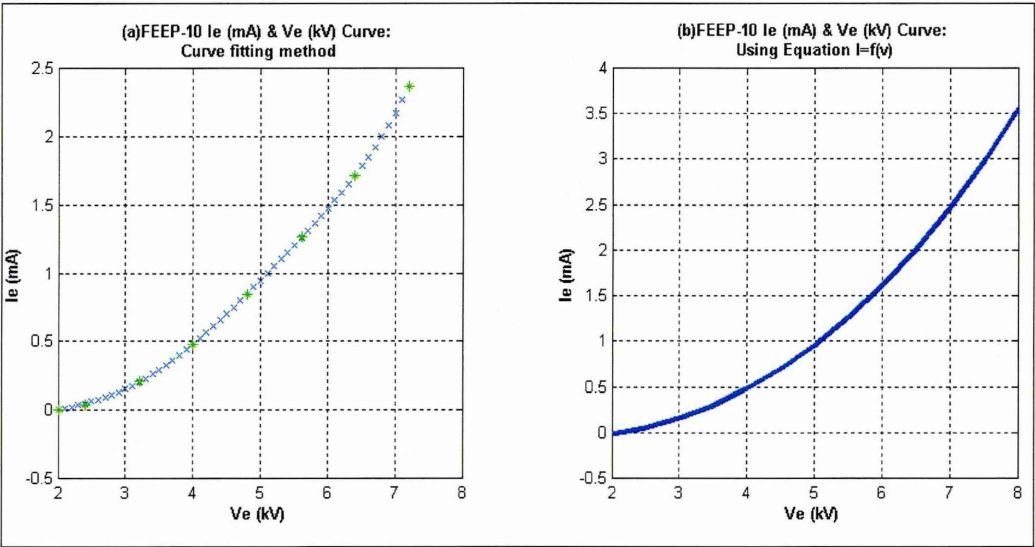


Figure 5-4 (a) FEEP 1-cm Emitter voltage (kV) & current (mA) relationship from ESTEC test results; (b) derived relationship to predict the test results using polynomial interpolation.

The Control torque requirement was calculated using ESA/ADS attitude control simulation tool for a simple ‘disturbance-counteracting’ mission with ‘Mustang’ satellite characteristics. From the pitch control torques, FEEP thrust requirement was calculated for the selected mode. According to section-4 FEEP thrust is calculated using the Equation 5-6 (Stuhlinger, 1964).

$$F = \left(2 \frac{m}{q}\right)^{1/2} I_e V_e^{1/2}$$

Equation 5-6

Where m/q is the selected propellant mass (m) to charge (q) ratio. For singly charged Caesium ions, the mass to charge ratio can be calculated as:

$$\frac{m}{q} = \frac{M \times m_p}{e} = \frac{132.9 \times 1.67252 \times 10^{-27} \text{ kg}}{1.6021 \times 10^{-19} \text{ C}} = 1.3787416 \times 10^{-6} \frac{\text{kg}}{\text{C}}$$

Equation 5-7

Where,

M = Atomic weight (132.9 for Caesium)

m_p = Proton rest mass ($1.67252 \times 10^{-27} \text{ kg}$)

e = Elementary charge of electron ($= 1.6021 \times 10^{-19} \text{ C}$)

Substituting the results of Equation 5-7 in Equation 5-6 gives for Caesium, thrust as a function of emitter voltage and emitter current as of Equation 5-8.

$$F_{CS} = 1.666 \times 10^{-3} I_e V_e^{1/2}$$

Equation 5-8

Equation 5-8 is solved together with the polynomial relationship (Equation 5-5) developed to plot Figure 5-4 (b), for emitter current and voltages. From these results, minimum emitter voltage was selected taking into account 2 kV minimum excitation voltage. The emitter voltage and current demand for disturbance (Aero, Solar pressure, gravity gradient and Magnetic torques) counteraction in pitch mode is shown in Figure 5-5 in (a) and (b) respectively as a function of time.

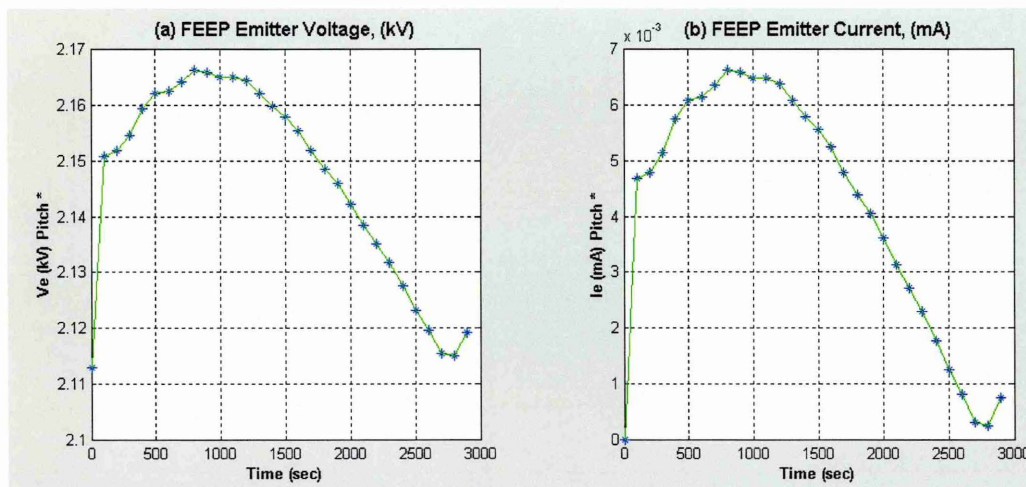


Figure 5-5 FEEP Emitter voltage (at left) & current (at right) in disturbance rejection mode (Including magnetic disturbances)

FEEP power requirement was calculated from the emitter voltage and current neglecting the accelerator power consumption, which is relatively very low. Typical power consumption for pitch, roll and yaw control torques are shown in Figure 5-6. From Figure 5-6 it is clear that the power requirement in attitude control mode is much lower than the set 3 W upper limit. For the FEEP control torques, the total power demand is less than 50 mW to provide roll, pitch and yaw control. If calculated using a linear model considering the average power demands of 60 W/mN, the maximum power requirement will still not exceed more than 200 mW. The main reason for these deviations is that, during the linear approximation method a straight line was best fitted at the higher emitter voltage levels in figure 4-9. By inspecting figure 4-9 it is clear that at high emitter voltages, the curve has a greater gradient than at low voltages. Therefore at high emitter voltages, a small increase in emitter voltage results a much greater thrust than at low voltage levels.

However it is important to note that for the above discussion power consumption from the beam neutralizer, propellant temperature control, electronics, high voltage power plant etc, were not taken into account. With all these factors taken into account, the total power consumption will be less than the set 3 W maximum ceiling. In order to provide maximum power for the propulsion system, it requires a battery capacity of approximately 20 W.hr per orbit (800 km Altitude and 25% DOD). This power demand can be easily met for a couple of orbits with the current primary battery technology. However for a longer duration mission, solar arrays together with secondary batteries are required to provide adequate power throughout the mission lifetime.

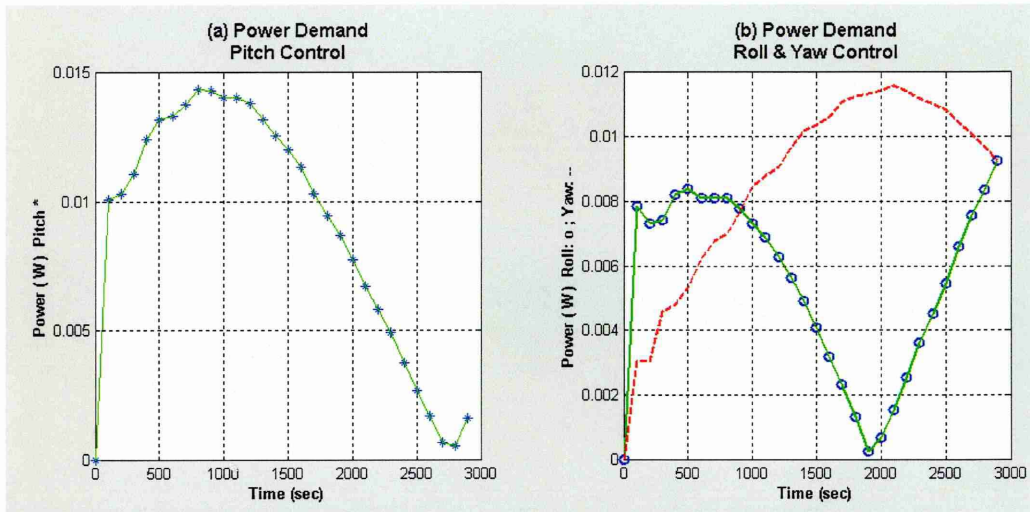


Figure 5-6 FEEP-10 Power demand in disturbance rejection mode. (a): Power requirement for pitch control thrusters. (b): Power requirement for roll and yaw control thrusters.

5.7 Summary

The selection of an electric propulsion system increases the demand for a reliable electrical power subsystem. If not studied in detail, the FEEP thruster application with nano-satellite may find unreliable. However with the current technology and its continuous development, high power demands could be met within a small mass and size. The availability of these electrical components were studied and it had shown that the FEEP power demand could be easily met within a Mustang class nano-satellite.

Further investigations with the FEEP-10 emitter showed that the required power demand for many applications with nano-satellites are much below than its power generation capacity for this category of satellites. It had also shown for a FEEP demonstrator mission that it is possible to use primary or secondary batteries in conjunction with body mounted solar cells to meet the required power demand. However the selection is depending on the length of the mission. Available voltage converters were discussed, as FEEP thrusters require a very high potential field for ion generation and emission. It had showed that the available voltage converters meet the demands of the FEEP thrusters as the operational frequencies are much below than their maximum.

A detail study of the FEEP performance is given in the next section. The result of this study will show that with the set maximum power for the FEEP propulsion system is adequate for most of the applications.

6 Simulation Analysis

Simulation tools that are used for analysis and visualisation of the orbit propagation and attitude control are discussed here. A brief description of the applied simulation tools is given here. The STK and ESA/ADS are the two simulation packages that were used extensively during this study. The Satellite Tool Kit (STK) package was used mainly to demonstrate the capability of FEEP thrusters to maintain a given constellation. The European Space Agency attitude determination software package (ESA/ADS) was used to identify the attitude control capability of FEEP thrusters. Standard packages such as Matlab and Simulink are used for various control analysis and feedback control design. Attitude development due to external torques were analysed using feedback control loops. A PID controller was designed to meet various demands of the control system.

6.1 Distance Measurement

During the FEEP thruster collaborative control study different phrases were used for distance measurements. In order to minimise any misunderstandings, a short descriptions of them are given below. Importantly the origin of relative distance measurement can be either at the target or interceptor satellite. In this study Interceptor satellite was used as the origin of distance measurements if not otherwise stated.

Cross Track Range

Cross Track Range represent the distance, measured perpendicular to the orbit plane: from the target location to the interceptor plane or vice versa (see Figure 6-1).

In Track Range

In Track or Along Track Range represent the distance, measured along the orbit track, from the target satellite point to the interceptor satellite or vice versa (see Figure 6-1).

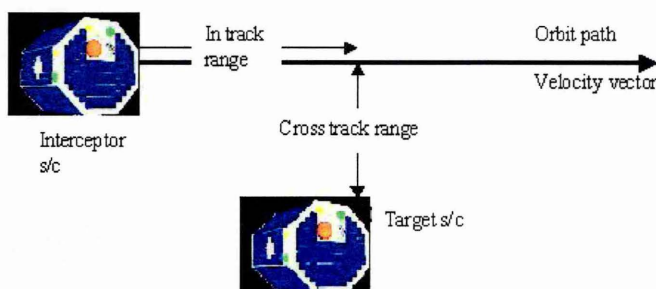


Figure 6-1 Cross track and In track range

Radial

Distance measured from target to interceptor orbit along the radial vector.

Tangential

Distance is measure from the origin along the velocity vector. Positive if the reference s/c is in front of the origin (see Figure 6-2).

Normal

Distance is measured from the origin, perpendicular to the velocity vector. Positive if the reference s/c is above the origin (see Figure 6-2).

Range

Vector distance measured from origin to Target (see Figure 6-2).

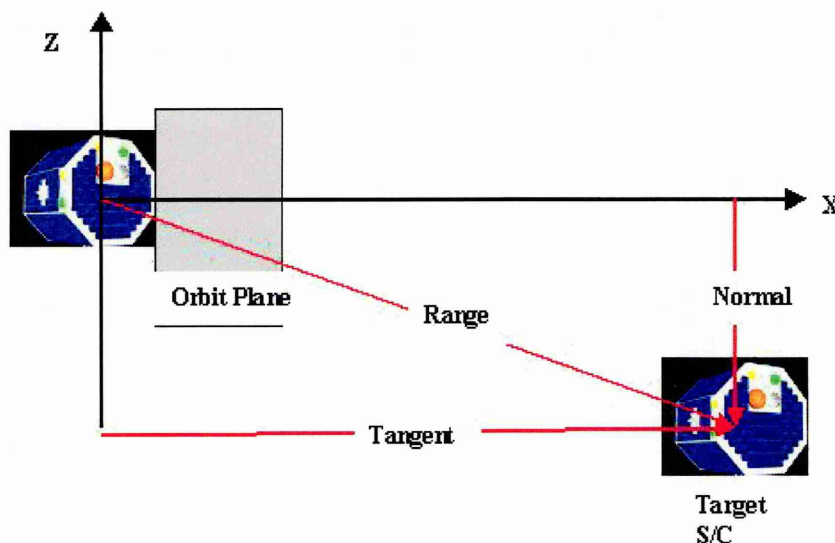


Figure 6-2 Orbit plane distance measurement

6.2 Satellite Tool Kit (STK) Simulation

The Satellite Tool Kit (STK) Educational version (4.2 and 4.3) was used to analyse the FEEP thruster formation-flying capabilities. STK offers a variety of analytical and numerical propagators to predict the orbit of a satellite. The Two-Body, or Keplerian motion, propagator considers only the force of gravity from the Earth, which is modelled

as a point mass. J2 Perturbation (first-order) and J4 Perturbation (second-order) propagators account for secular variations in the orbit elements due to Earth oblateness. These propagators do not model atmospheric drag or third body gravitational forces. The Merged Simplified General Perturbations (MSGP4) propagator, a standard NORAD propagator, is used with two-line mean element (TLE) sets. It considers secular and periodic variations due to Earth oblateness, solar and lunar gravitational effects, gravitational resonance effects and orbital decay using a simple drag model. The Astrogator, the state of the art STK propagator, provides trajectory and manoeuvre planning and includes targeting capabilities. The Astrogator allows the user to select different built in propagators such as Earth point mass, Earth J2, Earth full, Earth full gauss Jackson etc. A full force model, such as the Earth full model used in this study, takes into account all major perturbations (Earth oblateness, Atmospheric, solar pressure, point mass effects from Sun and Moon) acting on the satellite. Following models are used in the full force propagator for calculation of different perturbation forces.

- Central body: Earth
 - Defines the Earth gravitational parameter (μ)
 - Size (the distance from the centre of mass of the central body to its surface) and shape of the Earth (oblate spheroid)
- Atmospheric model
 - Jacchia-Roberts (Boland D. E. Jr. and Taesul Lee, 1982)
 - Computes atmospheric density based on the composition of the atmosphere, which depends on altitude as well as seasonal variation. Valid range is 100-2500 km.
- Point mass effect:
 - Gravitational attraction from the Sun and Moon
- Gravity field: JGM2 (Analytical Graphics Inc. 2003)
- Solar radiation with apparent Sun position (Flux at 1AU = 1358 W/m²)
 - Acceleration due to solar radiation pressure
- Numerical integrator: RKV8(9)
 - An eight order Runge-Kutta-Verner integrator with ninth order error control (Analytical Graphics Inc. 2003)

STK Full force model (Astrogator) was used for the close proximity formation flying simulations in order to meet the required high accuracy due to the distance between the two satellites as well as very low thrust levels from the FEEP thrusters. The ‘Mustang’ satellite physical properties (Roberts et al. 2002) were used to represent a nano-satellite. An 800-km circular polar orbit was selected as the initial reference orbit. The ‘Jacchia-Roberts’ atmospheric model used with ‘Earth Full’ propagator takes into account the atmospheric density based on the composition of the atmosphere, which depends on the altitude as well as the Earth climate seasonal variation. The selected reference orbit (800 km) is within the valid range, which is 100 – 2500 km, of this model. Table 6-1 shows the model parameters that were used for STK simulations.

Parameter	Model	Comment
Gravitational constant (μ)	398600.44180 km ³ /sec ²	$\mu = MG$ M = Central body (Earth) mass G = Universal gravitational constant
Gravity model	JGM2	NASA Joint Gravity Model 2 (Nerem R.S. et al. 1994)
Shape	Oblate Spheroid $r_{\min} = 6356.752314240$ km $r_{\max} = 6378.1370$ km $f = 0.00335281$	$r_{\min}$: minimum radius $r_{\max}$: maximum radius f : flattening coefficient $f = 1 - \frac{r_{\min}}{r_{\max}}$
Atmospheric model	Jacchia-Roberts	Atmospheric density model

Table 6-1 Selected model parameters

FEEP thrusters operation was simulated considering a continuous constant thrust model. A pre selected ‘ ΔV ’ was used as a stopping condition, for the selected thruster operation segment. The ‘ ΔV ’ requirement was calculated using the Hill’s equation to propagate to a required distance from the target satellite. A specific impulse value of 3000 seconds ($V_{\text{equivalent}} = I_{sp} \times g_E = 29,400$ m/s), which is at the low end for FEEP thrusters, was used. Mass flow rate was calculated from the applied thrust and specific impulse using Equation 6-1. Simulations were performed for a separation ‘ ΔV ’ of impulsive and finite manoeuvre type. In contrast to Hill’s equation, where the target follows an unperturbed reference orbit, STK simulations take into account perturbations effects on both target and interceptor satellites. Another limitation in Hill’s equation is that it is not valid for finite manoeuvres; hence FEEP performance cannot be modelled with the required accuracy using the simplified Hill’s equations. This limitation was overcome in this study by using STK for such analysis.

$$m = \frac{F}{I_{sp} g_E}$$

Equation 6-1

Where;

- m : Mass flow rate (kg/s)
- I_{sp} : Specific impulse (seconds)
- g_E : Earth gravitational acceleration at sea level (≈ 9.8 m/s²)

6.3 STK Quantification Study

Propagators

The STK-Astrogator ‘Earth Full’ propagator results are compared with respect to the ‘Earth point mass’ propagator for a circular, polar, 800 km altitude orbit (Eccentricity, $e = 0$; inclination, $i = 90^\circ$; semi-major axis, $a = 7178.137$ km). As shown in Figure 6-3 semi-major axis has a short-term (oscillations on the order of satellite period or less) periodic variation for the earth full propagator model.

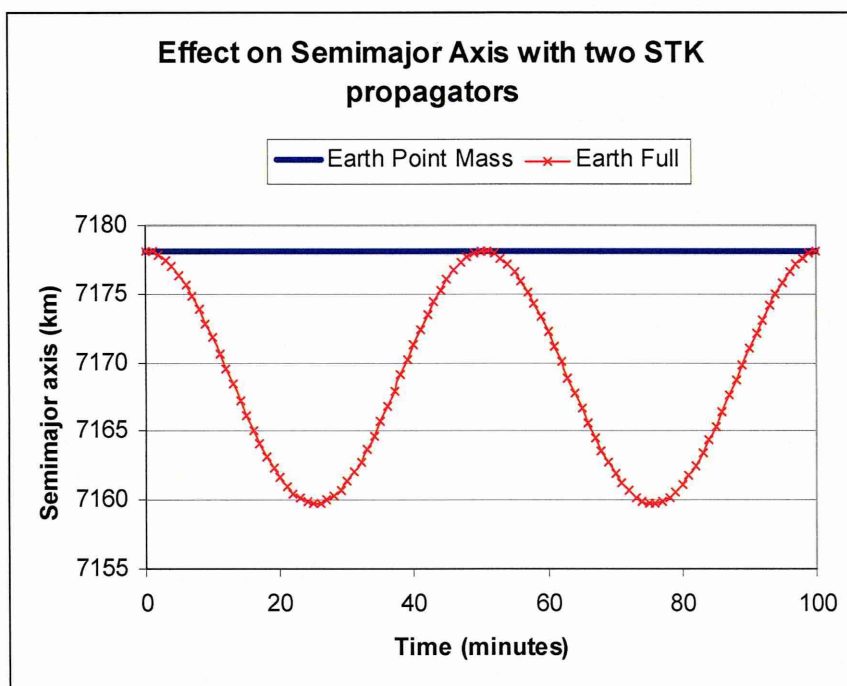


Figure 6-3 Semi-major axis variation with two different propagators using STK (initial conditions: $i = 90^\circ$, $\Omega = \omega = v = 0^\circ$, $a = 7178.137$ km, $e = 0$)

Where,

- i: Inclination
- a: Semi-major axis
- Ω : Right ascension of the ascending node
- ω : Perigee
- v: True anomaly
- e: Eccentricity

The ‘Earth Point Mass’ propagator calculates accelerations due to central body as if all the mass is concentrated at its centre of mass. The J2 (first gravitational zonal harmonic coefficient, which takes into account first order Earth oblateness effects) and J4 (takes

into account first and second order Earth oblateness effects) propagators take into account secular variations in the orbital elements due to Earth oblateness. However both J2 and J4 propagators do not take into account short-term orbital perturbations. Therefore all of the above three propagators (two body, J2 and J4), do not show any short-term periodic variation as shown in Figure 6-3 for Earth full model with accurate propagators such as ‘HPOP’ and ‘Astrogator’.

For the central-body motion, orbital energy, ζ , the sum of kinetic and potential energy, is defined in terms of semi-major axis, a , according to Equation 6-2. For a conservative force field, such as Earth gravitational potential, orbital energy remains constant if there are no external forces, such as air drag, acting on the satellite.

$$\zeta = -\frac{\mu}{2a} = \frac{v^2}{2} - \frac{\mu}{r}$$

Equation 6-2

Where,

ζ : Orbital energy

μ : Gravitational parameter, = GM

G: Universal gravitational constant, $6.67 \times 10^{-11} \text{ N m}^2/\text{kg}^2$

M: Mass of the central body

v : Orbital velocity

r : Orbital radius

When more precise propagators such as ‘High Precision Orbit Propagator’ (HPOP) or ‘STK Astrogator Earth Full’ propagator are used for analysis, it was observed a sinusoidal variation of the semi-major axis with half the orbital period as of Figure 6-3. This sinusoidal variation is caused due to ‘zonal harmonics’, which takes into account latitudinal Earth’s gravitational departure from a perfect sphere. J2 is by far the strongest perturbation due to Earth’s shape and is almost 1000 times larger than the next largest coefficient (J3). The earth’s departure from a perfect sphere can be visualised as bands of mass concentration, symmetrically around the polar axis (latitudinal mass change) and the highest concentration along the equator. If considered the instantaneous conditions, the satellite feels more mass at its start (above the equator) and consequently speeds up (instantaneous increase in mean motion) towards the poles-the added attraction of the Earth’s equatorial bulge introduces a force component toward the equator. Consequently the instantaneous semi-major axis reduces and mean motion increases according to Equation 6-3. From poles to the equator the opposite happens: satellite mean motion decreases and the semi-major axis increases. As the gravitation is a conservative force field, the orbital energy remains constant for a given orbit. This is true for both models as the orbital energy is a function of instantaneous velocity and radius, which varies in sinusoidal form in the ‘Earth full’ propagator. Although classical orbital parameters: semi-major axis, eccentricity and inclination do have a short-term periodic change, their average change per orbit is null. That is there are no secular variations for semi-major axis, eccentricity and inclination as gravity is a conservative force field, a net change of the above orbital parameters is not possible. Importantly these short-term periodic

variations have a considerable effect for a close range formation-flying mission at LEO. The amplitude (peak to peak) of these short-term periodic variations will reduce for high altitude missions (i.e. semi-major axis variation is about 8 km for altitudes 12,000 km and about 2 km for altitudes around geosynchronous orbits).

$$n = \sqrt{\frac{GM}{a^3}}$$

Equation 6-3

The Earth Full model uses the JGM-2 gravity model for trajectory calculations. In this gravity model the departure from the point mass model is represented using the gravitational potential function given in Equation 6-4 with measured data (satellite tracking data) for accurate analysis (Nerem R.S. et al. 1994). The central body (in this case the Earth) mass distribution both in latitudinal and longitudinal takes into account here. A more detail discussion of the periodic orbital element variation due to Earth mass distribution is given by Sidi (in section 2.8.2) (Sidi, 1997).

$$U(r, \phi, \lambda) = \frac{GM}{r} \left(1 + \sum_{n=2}^{\infty} \sum_{m=0}^n \left(\frac{a}{r} \right)^n P_{nm}(\sin \phi) [C_{nm} \cos(m\lambda) + S_{nm} \sin(m\lambda)] \right)$$

Equation 6-4

Where,

- G: Universal gravitational constant
- M: Mass of the Earth (or central body)
- r: Radial distance from Earth's centre of mass
- a: Semi-major axis of the reference ellipsoid (equatorial radius of the Earth)
- n: Degree of the harmonic term
- m: Order of the harmonic term
- ϕ : Geocentric latitude
- λ : Geocentric longitude
- C_{nm}, S_{nm} : Spherical harmonic coefficients
- P_{nm} : Associated Legendre polynomials

JGM-2 (Joint Gravity Model) was developed from a combination of satellite tracking data, surface gravimetric and satellite altimetry observations. It is complete to degree and order 70 in spherical harmonics. The principal cause of uncertainty in the JGM-2 model is the limitation of tracking data from satellites in different inclination bands and the lack of precise tracking data for satellites in very low orbits. However these models provide enough accuracy (error below 1 cm) for FEEP thrusters' simulation studies.

STK quantification was used to analyse the effect on orbit due to an impulsive and continuous ' ΔV ' application. Simulations were performed with similar initial conditions as in the earlier case but with a propagation time of 100 minutes prior to and after the applied ' ΔV '. The corresponding semi-major axis variation after an impulsive ' ΔV ' is shown in Figure 6-4 using 'Earth point mass' and 'Earth full' propagators. Importantly

‘Earth point mass’ propagator gives an easy to compare results with analytic calculations after the applied ‘ ΔV ’. However in order to achieve a good accuracy for a close proximity formation flying mission analysis, the ‘Earth full’ propagator is important to use as it takes into account periodic variations.

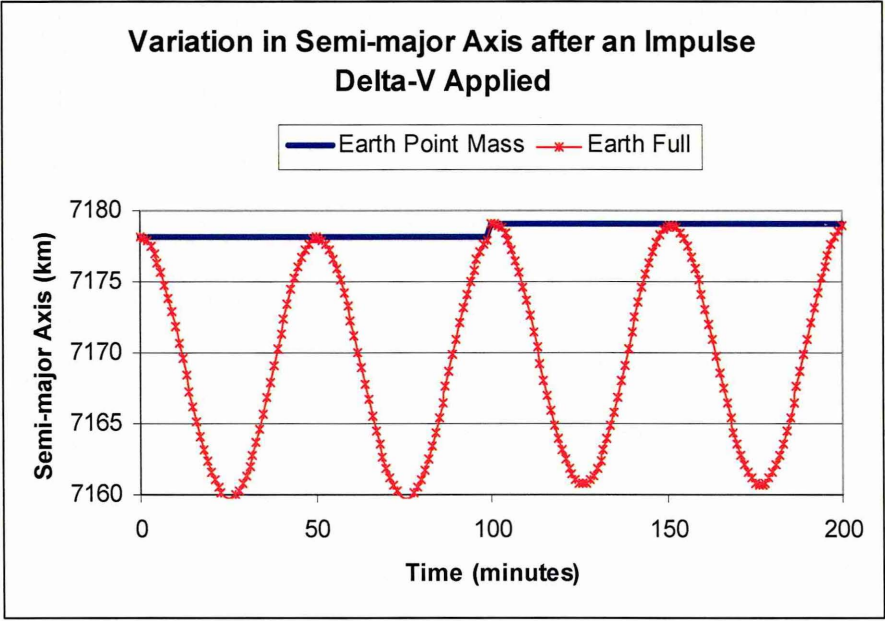


Figure 6-4 Variation in Semi-major axis after an impulse ΔV ($= 0.5$ m/s) application along the velocity vector using ‘Earth point’ and ‘Earth full’ propagators (initial conditions: $i = 90^\circ$, $\Omega = \omega = \nu = 0^\circ$, $a = 7178.137$ km, $e = 0$)

Prior to the applied ‘ ΔV ’ the orbital velocity for an 800 km circular orbit can be calculated using Equation 6-5.

$$V = \sqrt{\frac{\mu}{r}}$$

Equation 6-5

Where,

V : Orbital velocity

μ : Gravitational constant ($= 398600.4418$ km³/sec² for Earth)

r : Orbital radius ($= 7178.137$ km for a 800 km circular orbit)

Initial orbital velocity prior to ‘ ΔV ’:

$$V_i = 7.45183 \text{ km/sec}$$

Applied ΔV (impulse) $= 0.50$ m/sec along velocity vector

Orbital velocity just after the applied ΔV :

$$V_f = 7.45233 \text{ km/sec}$$

Orbital energy, ϵ :

$$\epsilon = \frac{V^2}{2} - \frac{\mu}{r} = -\frac{\mu}{2a}$$

Equation 6-6

Where,

- ϵ : Orbital energy
- V : Orbital velocity
- μ : Gravitational constant
- r : Orbital radius
- a : Orbit semi-major axis

For $V = 7.45233 \text{ km/sec}$ and $r = 7178.137 \text{ km}$, orbital energy using Equation 6-6:
 $\epsilon = -27.76117 \text{ km}^2/\text{sec}^2$

And semi-major axis, a , after the applied ΔV :
 $a = 7179.10043 \text{ km}$.

	Analytical (Two body)	STK (Earth Point Mass)	STK (Earth Full)
r_{initial} (km)	7178.137	7178.13700	7178.13699
V_{initial} (km/sec)	7.451831	7.45183133	7.45183125
a_{initial} (km)	7178.137	7178.13700	7178.13683
$\epsilon_{\text{initial}}$ (km ² /s ²)	-27.76489511	-27.764895	-27.764895
$\Delta V_{\text{(impulse)}}$ (km/sec)	0.0005	0.0005	0.0005
r_{final} (km)	7178.137	7178.13700	7178.13698
V_{final} (km/sec)	7.452331333	7.45233133	7.45233116
a_{final} (km)	7179.100433	7179.10043	7179.10005
ϵ_{final} (km ² /s ²)	-27.76116907	-27.76117	-27.76117

Table 6-2 Comparison of semi-major axis derived with analytical and the STK propagators, round to cm precision (Earth point and Earth full: 1 second propagation before and after ΔV)

According to Figure 6-3 and Figure 6-4 the two STK propagators (Earth point mass and Earth full) instantaneous results varied considerably with time. For example the instantaneous difference in semi-major axis between the two STK propagators can be as much as 20 km within a single orbit period (6050 seconds). This shows the importance of using an accurate propagator such as ‘Astrogator Earth Full’, for close proximity (less than 100 meters) formation flying analysis. However as shown in Table 6-2, a simple propagator like ‘Earth Point Mass’ is adequate for short-term (few seconds) orbital variation analysis. Further a simple propagator gives a good understanding of the problem, such as in the above case the impact on semi-major axis due to the applied ΔV .

Drag Effects

Orbital perturbations due to drag forces were studied using STK High Precision Orbit Propagator (HPOP). This was used as it allows to switch on and off between ‘Aero’ and ‘Solar Pressure’ drag forces. The orbital perturbations due to aero drag forces are shown in Figure 6-5 using Table 6-3 properties in the HPOP propagator.

	Properties
Orbit	Semi-major axis = 7178.137 km; Inclination = 90°; Eccentricity, $e = 0$; RAAN, $\Omega = 0^\circ$; Argument of perigee, $\omega = 0^\circ$; True anomaly, $v = 0^\circ$
Earth Gravity field	JGM-2 model
Third body gravity	Solar and Lunar gravity
Aero Drag	Drag coefficient, $C_D = 2.0$ Area/Mass ratio = $0.0075 \text{ m}^2/\text{kg}$ Atmospheric density model: Jacchia-Roberts

Table 6-3 Selected values for aero drag effect study with STK-HPOP

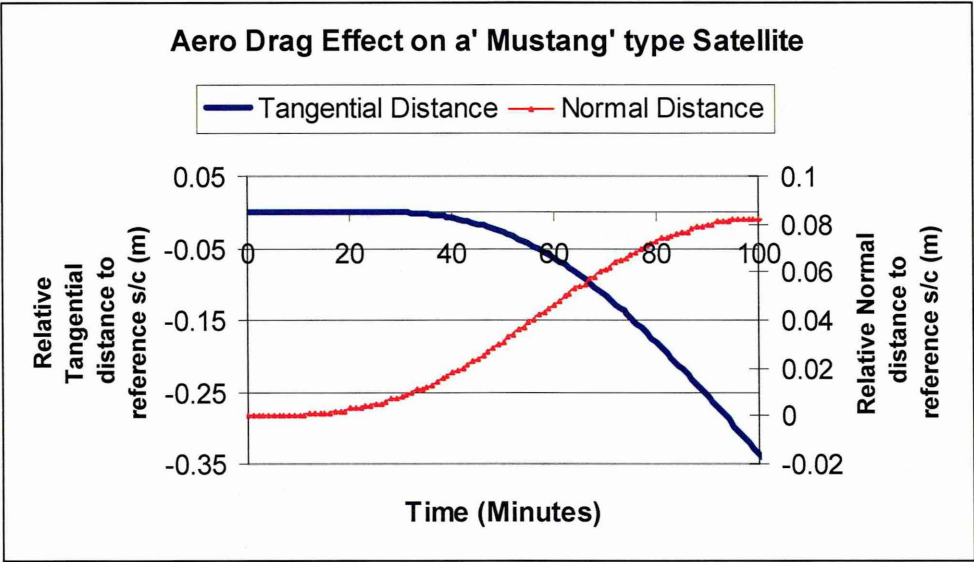


Figure 6-5 Relative distances from a reference s/c due to ‘aero drag’ forces.

The aero drag forces acting on a spacecraft will eventually decrease the semi-major axis, which in turn decreases the orbital period. The drag effects were analysed with respect to a reference spacecraft, which propagates in an un-perturbed reference orbit (reference s/c does not experience aero or solar radiation drag forces). Therefore with respect to the reference s/c, the drag-induced s/c will propagate in front (negative tangential distance) and below (positive normal distance) the reference s/c. As seen from the Figure 6-5, at

the end of one orbital period (6000 seconds) the reference and perturbed s/c have a range of 36 cm (35 cm in tangential and 8 cm in normal distance).

Similarly solar pressure drag effect was studied using the STK-HPOP propagator. Table 6-4 summarises the different values used for this study. As shown in Figure 6-6, after a one orbital period (6000 seconds) the relative distance in range between the reference and solar pressure perturbed s/c are about 71 cm (68 cm in tangential and 20 cm in Normal). Unlike the aero drag forces, which reduce the total orbital energy, the solar pressure forces effect can either increase or decrease the instantaneous orbital energy. The main reason for this is that the solar pressure force effect is depending on the solar angle as well as the selected orbit. Depending on the Sun angle sometimes the net solar pressure force is opposite to the velocity vector (decreases the orbital energy) and sometimes along the velocity vector (increases the orbital energy).

	Properties
Orbit	Semi-major axis = 7178.137 km; Inclination = 90°; Eccentricity, $e = 0$; RAAN, $\Omega = 0^\circ$; Argument of perigee, $\omega = 0^\circ$; True anomaly, $\nu = 0^\circ$
Earth gravity field	JGM-2 model
Third body gravity	Solar and Lunar gravity
Solar pressure drag	Reflection coefficient, $Cr = 1.6$ Area/Mass ratio = $0.0095 \text{ m}^2/\text{kg}$ Shadow model: Dual cone

Table 6-4 Selected values for solar pressure drag effect study with STK-HPOP

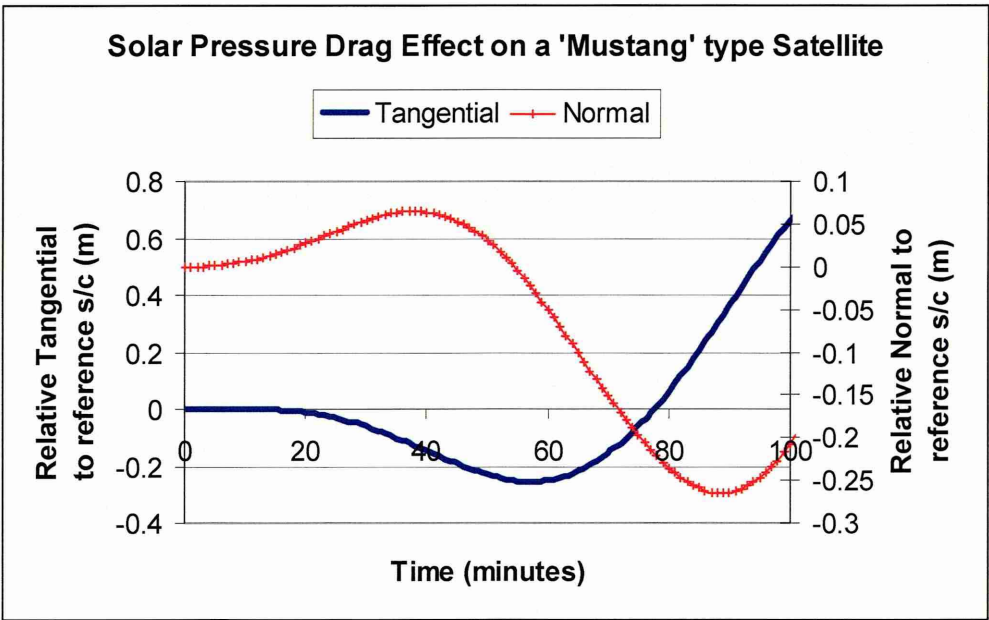


Figure 6-6 Relative distances from a reference s/c due to ‘solar pressure drag’ forces

The models for atmospheric drag and solar radiation pressure calculations use the following analytic expressions:
Acceleration caused by atmospheric drag is:

$$a_D = C_D \left(\frac{A_D}{m} \right) \left(\frac{\rho V^2}{2} \right)$$

Equation 6-7

- Where,
- a_D : Atmospheric drag acceleration (deceleration)
 - C_D : Drag coefficient
 - A_D : Satellite cross sectional area perpendicular to the velocity vector
 - m : Satellite mass
 - ρ : Atmospheric density
 - V : satellite speed relative to the atmosphere

For the values given in Table 6-3 and $\rho = 9.41 \times 10^{-14} \text{ kg/m}^3$ (mean atmospheric density at 800 km altitude), $V = 7.45183 \text{ km/s}$ (800 km circular orbit)
 $a_D = 3.92 \times 10^{-8} \text{ m/s}^2$

The worst-case ΔV_D change per orbit (6000 s) due to the above atmospheric drag:

$$\Delta V_D = \int a_D dt = 2.35 \times 10^{-4} \text{ m/s}$$

Similarly acceleration caused by solar radiation pressure is:

$$a_p = (\kappa)(C_r) \left(\frac{A_R}{m} \right) \left(\frac{L_s}{4\pi cr^2} \right)$$

Equation 6-8

Where,

- a_p : Solar radiation pressure acceleration (deceleration)
 - κ : Fraction of the solar disk visible at the satellite location
 - C_r : Reflection coefficient
 - A_R : Satellite cross sectional area presented to the Sun
 - m : Satellite mass
 - L_s : Luminosity of the Sun ($=3.823 \times 10^{26} \text{ W}$)
 - c : Speed of light ($= 299792.458 \text{ km/s}$)
 - r : Distance of satellite from Sun
- (Distance from Earth to Sun = 1 AU; 1 AU = $1.49597870691 \times 10^{11} \text{ m}$)

For the values given in Table 6-4 and $\kappa = 1$, $\left(\frac{L_s}{4\pi cr^2} \right) = 4.5344 \times 10^{-6} \text{ N/m}^2$

$$a_p = 6.89 \times 10^{-8} \text{ m/s}^2$$

The worst-case ΔV_p change per orbit due to the above solar pressure force (assumed acting opposite to the velocity vector throughout the orbital period of 6000 s):

$$\Delta V_p = \int a_p dt = 4.14 \times 10^{-4} \text{ m/s}$$

6.4 ESA/ADS Simulation Tool

The ESA Attitude Determination Software (*ADS*) program is developed and distributed freely by the European Space Agency to evaluate its usefulness and user friendliness. This tool has the ability to perform attitude control using wheels, magnetic torque rods and thrusters. It handles attitude control during wheel off-loading using magnetic torque rods or thrusters. The disturbance torques are calculated both with external and internal disturbance sources. However, the internal disturbance effects such as solar array flutter, sensors and actuator noises were omitted during this preliminary analysis. ESA/ADS offers the following tools for AOCS designer:

- Modelling tools
- Disturbance calculation
- Attitude time simulation
- Automatic analysis
- Stability budgets

In ADS attitude simulations requires the following inputs from the user:

- Spacecraft design
- Its operational orbit
- The corresponding mode of operation

Orbits are limited for Earth oriented satellites either with circular or elliptical orbital parameters. From the orbital control point of view, the major limitation in *ADS* tool is its inability to handle orbital decay due to perturbations. Therefore attitude simulations were limited for few orbital periods, in order to keep the orbit parameters within the satisfactory limits of initial conditions. However, simulations can be performed for decaying orbits by analysing disturbance forces at different orbital altitudes. Recalling that the main application of this software tool is attitude determination and control, the different orbit segments should correctly represent the magnitude of perturbations. Disturbance torques are calculated from the user selected disturbance sources (external and internal) and from the selected satellite physical properties. The following external disturbance sources were taken into account during the attitude determination analysis in this study.

- Aerodynamic
- Solar Pressure
- Gravity Gradient
- Magnetic

The disturbance torques acting on a given satellite are calculated in the ESA/ADS simulation tool taking into account the above mentioned disturbance sources with the following input parameters.

- Spacecraft physical parameters
- Orbit
- Mode of operation
- Seasonal effects

In the design window the satellite is built in subsystem level. The final satellite configuration is used to compute the disturbance torques. The dynamics of the design satellite are used for the simulations and local budget analysis. Orbit type is selected at the orbit selection window. The orbits are considered inertial and propagate inertially except for 'helio-synchronous' and 'free' orbits. The free orbit propagator is the J2 orbit propagator (takes into account the Earth oblateness). According to the selected time period for the simulation, solar pressure force is calculated from the Sun vector direction. The ADS database object mode is very similar to a standard AOCS mode definition. In ADS a mode is defined by:

- A set of AOCS equipments positioned and localised with respect to the satellite body frame
- A well-defined feedback control loop

- A target attitude given in a reference frame
- The requirements that needs to be fulfilled in that mode
- Local budgets, which consists of pointing and stability budgets

ADS attitude control feedback loop is shown in a block diagram form in Figure 6-7. The dynamics of the satellite, which is defined as a state matrix, are derived from the physical properties of the design. The design is performed in a modular format using the components characteristics. Each module or the possible subsystem is defined with complementary information such as mass, size and power budget. The attitude determination and control is performed taking into account the selected equipments, control laws, body pointing, fulfilled requirements and mode stability. However the user is restricted to fill the control loop boxes without altering the control loop. The user is supposed to fill the following boxes in the control loop shown in Figure 6-7.

- Reference signals
 - It is possible define linear or sinusoidal evolution targets (roll, pitch and yaw) for the signals
- Estimator
 - Defined by transfer functions or state space matrices
- Measurements
 - User select the sensors used for the mode of operation
- Magnetic control
- Thruster control
- Wheel control
- Inertia (wheel)
- Thruster control (off-loading)

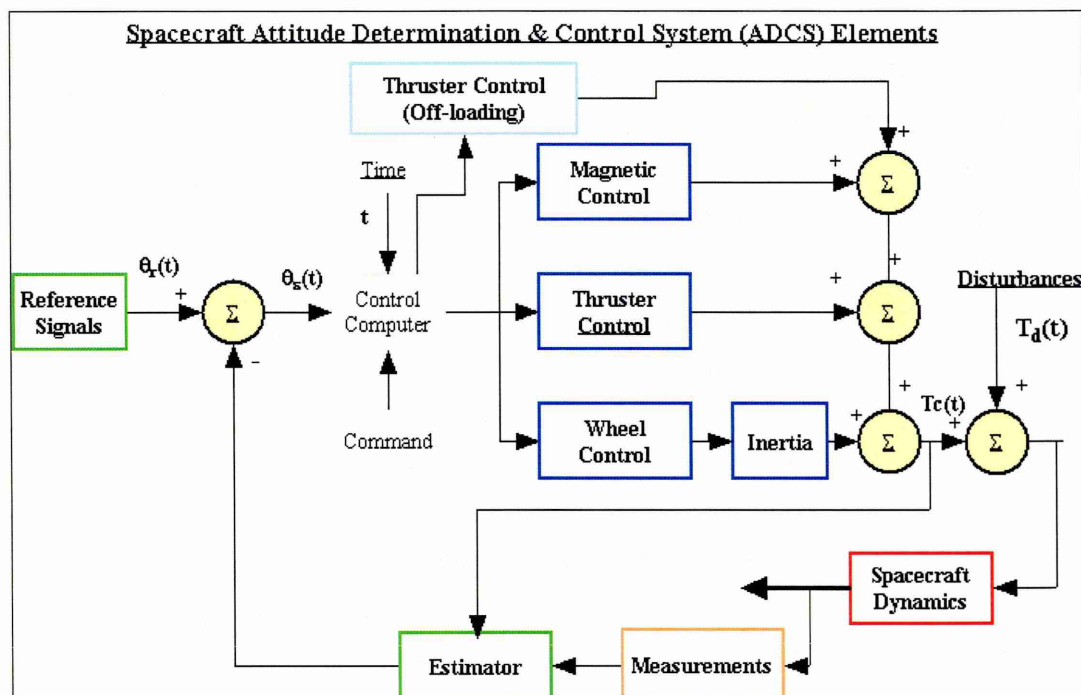


Figure 6-7 Block diagram of ESA/ADS attitude control system

Where,

$\theta_x(t)$: Spacecraft angle (pitch, roll or Yaw) as a function of time or as a constant

t : Time

$T_d(t)$: Disturbance torque (as a function of time)

$T_c(t)$: Control torque (as a function of time)

Σ : Sum

The control torques are calculated from the measured sensor data, estimator transfer function and from the reference signal. Attitude measurements for a given mode are performed with respect to the selected sensors in the 'Measurement window'. It displays equipment list and sensors. The properties of the measurements are saved in the equipment database. ADS control loop allows defining each on board instrument separately with its physical, electrical and sensor properties in the equipment database. The 'Estimator' defines the connection of different sensors, filtering and the required frame changes depending on the orientation of the sensors. The output of the 'Estimator' transfer function should be in consistent with the reference signal. The control torques are vector added with disturbance torques to derive the net torque acting on the satellite. From this knowledge spacecraft position and its rates are calculated according to Euler's moment equations. The reference signal block constitutes the angles and angular rates, which completely defines the attitude of the satellite with respect to a selected reference frame. The reference signal is possible to define using either linear or sinusoidal signals or a combination of both. Therefore ADS feedback loop is a fully defined AOCS system.

In ADS it is not possible to directly import a design developed in another platform such as in CAD. Therefore a satellite model was built in ADS to represent the ‘Mustang’ satellite. Figure 6-8 shows the model developed using ADS design tool. A cylindrical model with dimensions 300 x 300 x 314 mm was selected to represent the ‘Mustang’ satellite. Simulations were performed for an 800 km, circular orbits with J2 propagator for different orbit inclinations (Mustang is planned for a polar orbit). The satellite is assumed to be in an Earth pointing mode. Attitude control capability of FEEP thrusters were analysed with eight thrusters distributed between two clusters. These FEEP performance results were compared with respect to micro reaction wheels and magnetic torque rods. According to the analysis, the combination of FEEP and Reaction wheels gave attractive and precise attitude and orbit control results within the limits of nano-satellites size, mass and power limits.

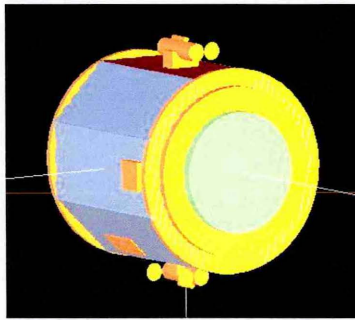


Figure 6-8 Reference satellite type ‘Mustang’

The stability of the feedback loop (Figure 6-7) is important for successful simulation of the AOCS system without getting any divergence errors. The control capability in Figure 6-7 is mainly depends on the selected transfer functions for the Estimator and the Controller. Therefore derivation of the relevant transfer functions and the feedback loop stability was analysed using the ‘Matlab’ and its control loop analysis ‘Simulink’ tool. However for the control loop analysis in Matlab/Simulink, the ADS feedback loop was highly simplified, considering torques along each axis (roll, pitch and yaw) separately. Disturbance torques were calculated from ADS and applied to the Matlab/Simulink as a function of Fourier coefficients. However dynamic changes such as the variations in disturbance torques due to spacecraft rotation were not taken into consideration in Matlab/Simulink, due to its complexity. The disturbance torques along the roll axis was calculated using Equation 6-9 (pitch and yaw torques can be derived in a similar way), where A_i and B_i are known i^{th} Fourier coefficients from ADS.

$$T_x = \sum \left\{ A_i \cos(\omega \cdot i \cdot t) + B_i \sin(\omega \cdot i \cdot t) \right\}$$

Equation 6-9

Where;

T_x : Torque vector along roll axis (= along the s/c velocity vector or the x-axis)

ω : Angular velocity
 t : Time
 $i = 0$ to n (integer)

With increase number of Fourier coefficients in Equation 6-9 (default value 16), the small variations in the disturbance torques will be included in the calculations. The disturbance torque profile for a free orbiting (without control torques) 'Mustang' type satellite was calculated using Equation 6-9. The corresponding disturbance torque profile is shown in Figure 6-9 due to disturbances from Aero, Solar, magnetic and gravity gradient torques. Here the magnitude of disturbance torque varies between 0 to $0.13 \mu\text{N.m}$ (0.06 to $-0.13 \mu\text{N.m}$). For simplicity the disturbance torque profile was calculated by limiting the Fourier expansion to eight coefficients.

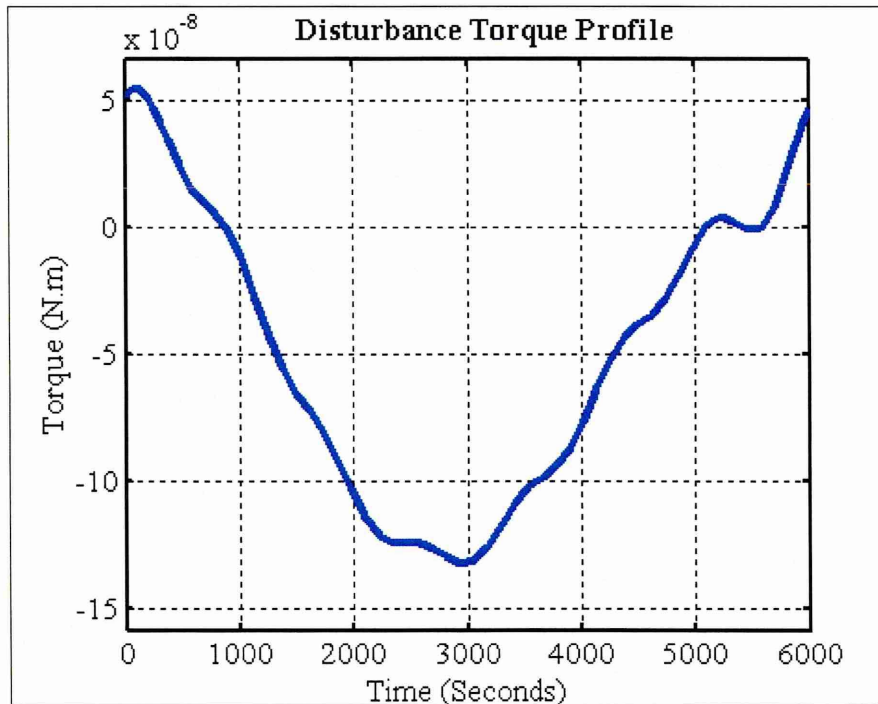


Figure 6-9 Disturbance torque profile derived from ADS Fourier coefficients for used in Matlab simulations

The disturbance torque profile derived from ADS was then used in a Matlab/Simulink feedback control loop shown in Figure 6-10, in the disturbance subsystem block (Dist-Subsystem) to generate the time varying disturbance torque. By solving the 'Euler Moment' equations for zero initial conditions (it is possible to define non zero initial conditions), the roll angle was computed using Matlab/Simulink. The corresponding roll angle variation with time due to disturbance torques (with no control torques) is shown in Figure 6-11.

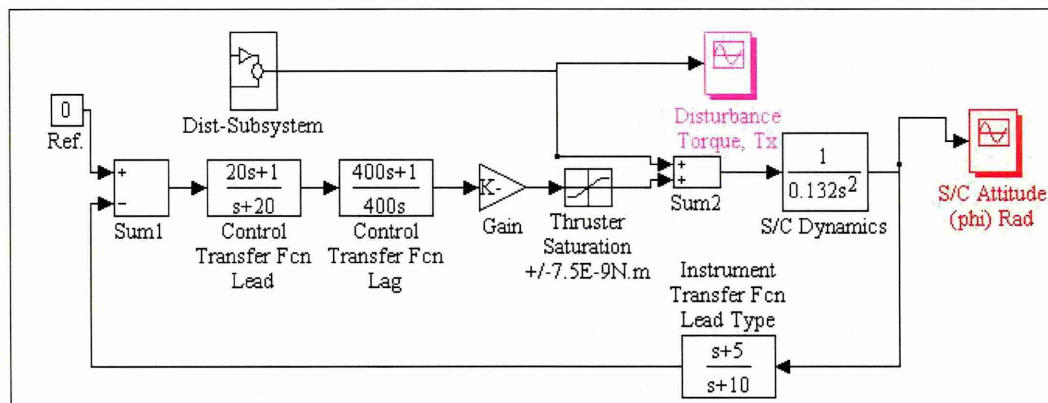


Figure 6-10 Matlab Simulink feedback loop for 'Mustang' roll axis attitude simulation.

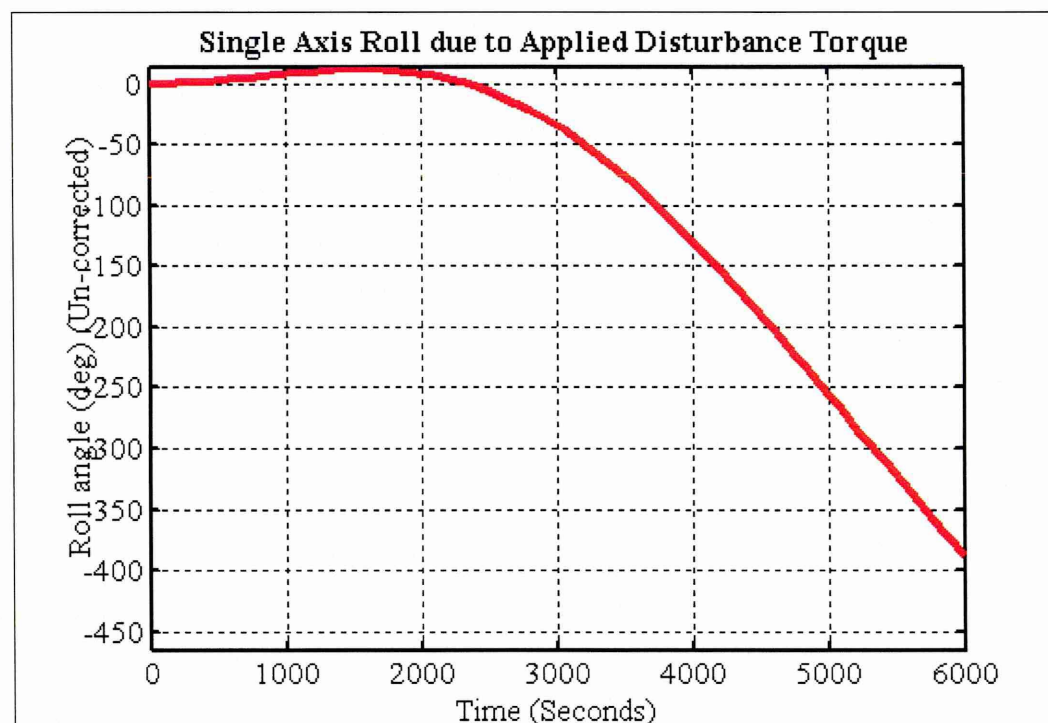


Figure 6-11 Single axes roll due to the applied disturbance torques with no correction torques applied

The s/c transfer function for roll angle determination is shown under the 's/c dynamics' block in Figure 6-10. It was assumed here that the rotational axis (I_x) is a principal axes and the body is symmetric around this axis ($I_y = I_z$). As shown in Figure 6-11 the roll angle (ϕ) will rotate one revolution and a 40° at the end of a one orbital period (~ 6000 sec.). For zero initial conditions the roll angle will change proportionate to the disturbance torques as a quadratic function of time according to Equation 6-10.

$$\phi = \frac{T}{I} \frac{t^2}{2} + \omega_0 t + \phi_0$$

Equation 6-10

Where,

- ϕ : Rotational angle (rad)
- T : Total torque (N.m)
- I : Rotational axes moment of inertia (kg.m²)
- t : Time (sec)
- ω_0 : Initial angular velocity (rad/sec)
- ϕ_0 : Initial angular displacement (rad)

In order to switch off the control torques, the gain constant was set to zero in the feedback control loop in Figure 6-10. This was done to study the disturbance torque effects on a free orbiting satellite without any control torques. However the used torque profile, which was derived from the *ADS* Fourier coefficients, is a *static* one, as it does not take into account the dynamic changes to satellite when it propagates in a disturbance force field. As shown in Figure 6-12 the *ADS* generated static disturbance profile and the disturbance profile used in Matlab shown in Figure 6-9, are not identical due to the selected calculation time steps. In the *ADS* static disturbance profile, magnitude of disturbance torque varies from 0 to 0.2 mN.m. (+150 to -200 μ N.m). Figure 6-13 shows the dynamic disturbance torque profile calculated with the *ADS* propagation tool for 800 km circular polar orbit. In a static calculation model, it does not take into account any attitude changes that affect the model due to these disturbance torques. This error is rectified in the dynamic propagator by taking into account the attitude changes to the satellite, where in each time step the torque profile and attitude profile gets updated due to variations in the surface area. This can be clearly seen by comparing the *ADS* static and dynamic disturbance torque profiles shown in Figure 6-12 and Figure 6-13 respectively.

Initially, the first 500 seconds, the static and dynamic torque profiles are very much similar, as the satellite attitude variation (cross sectional area variation) in both models are small. This is in line with the roll angle variation shown in Figure 6-11. However with time, the selected satellite model attitude starts to deviate considerably from its initial state and hence the dynamic torque profile varies as shown in Figure 6-13. The magnitude of the dynamic torque profile varies between 0 and 0.175 μ N.m and is in line with the static model. Importantly, both the static and dynamic torque magnitudes fall within a common range. Therefore the static torque model derived from the *ADS* Fourier coefficients can be still use to predict short-term attitude variations with the considered Matlab/Simulink feedback control loops.

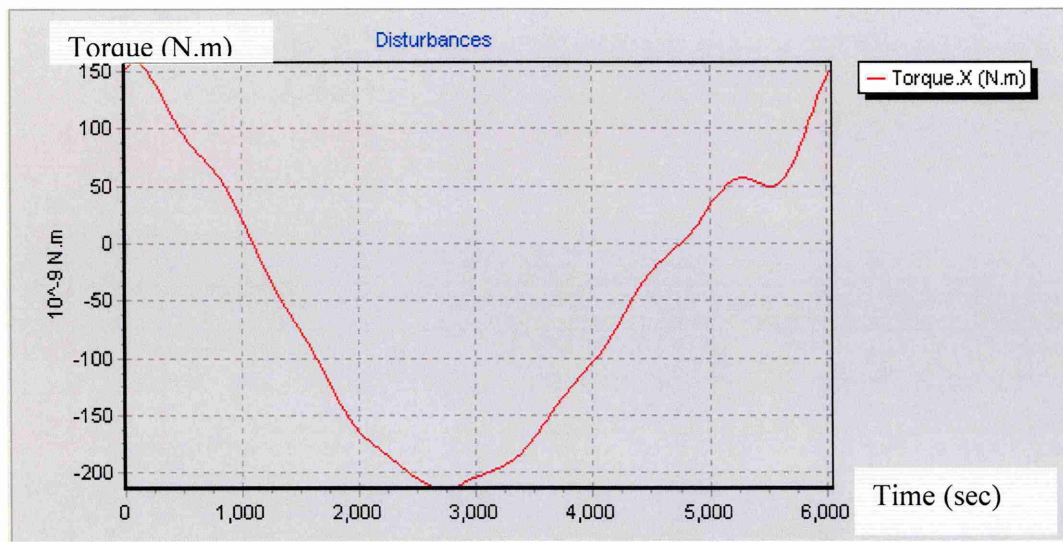


Figure 6-12 Disturbance torque profile generated from ADS with 8 Fourier coefficients

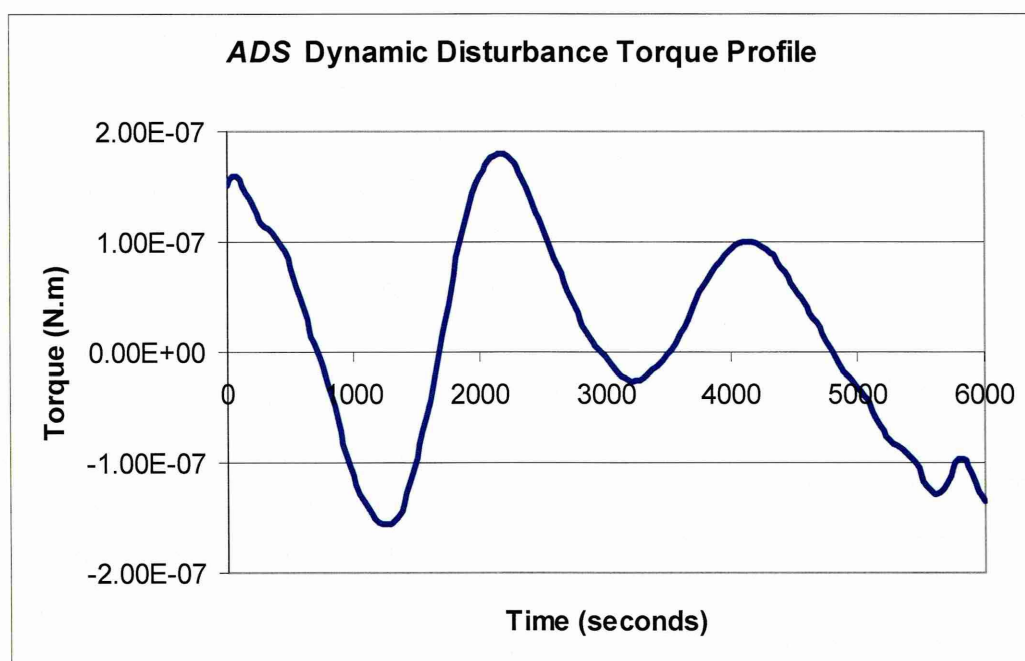


Figure 6-13 ADS dynamic disturbance torque profile with no corrections applied

Figure 6-10 shows the simple feedback loop used in 'Matlab/Simulink' to analyse the spacecraft attitude variation due to disturbance torques along the roll axis. This simple feedback loop consists of s/c and controller transfer functions, thruster saturation limits and disturbance torques. If the interest is to predict the disturbance torque affects for an uncontrolled s/c, the control torques can be set to zero by selecting a zero gain in the control transfer function. With this type of a highly simple feedback loop only single axis

control could be analysed and any possible ‘Euler Moment’ coupling effects were neglected in the derivation of s/c transfer function.

For a propagating s/c without any correction torques, the attitude variation in roll is shown in Figure 6-14 due to the disturbance torque profile given in Figure 6-13. Here the initial (the first 500 seconds) roll angle variation is similar to the one developed using a static disturbance torque profile (neglecting the variation in disturbance torques due to attitude variation with respect to time) with Matlab/Simulink feedback loop shown in Figure 6-11. This shows that the ESA/ADS simulation tool gives a more dynamic response (takes into account attitude variation effect) than analysis performed with a simple feedback loop in Matlab/Simulink environment. However the ‘Matlab/Simulink’ feedback loop shown in Figure 6-10 can be used to predict and analyse the short-term developments. Importantly both models show, although the disturbances torques are small, that if no correction torques are present, the selected model will soon be unstable.

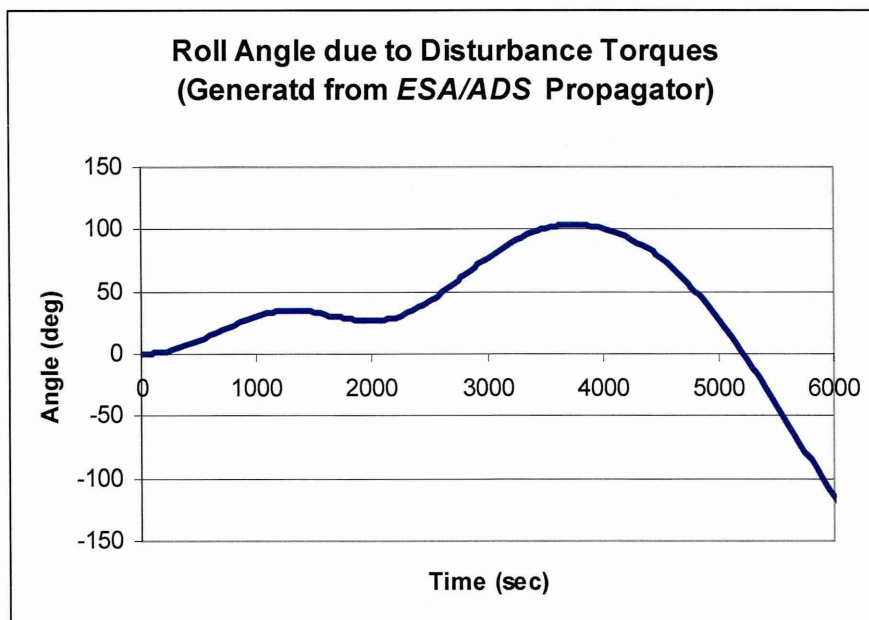


Figure 6-14 Spacecraft roll angle (angle in radians) due to disturbance torques (ESA/ADS Propagator)

The above analysis shows that the selected spacecraft (Mustang model) is prone attitude variation due to disturbance torques although the spacecraft is symmetrical along the roll axis (X-axis): the shift in centre of mass along the roll, pitch and yaw is reasonably small (2, 5 and 6 mm respectively). The major disturbance force acting on the spacecraft is due to the solar radiation and is about 2.5 times greater than aerodynamic force calculated at the 500 seconds from start. However the corresponding solar torque is 90 times greater than the aerodynamic torque for the corresponding time. This shows that the torque arm of the solar pressure is about 36 times greater than the corresponding aerodynamic torque arm. According to Equation 6-8, a low value of inertia (0.132 kg.m^2 for the selected ‘Mustang’ spacecraft roll axis) is too a contributing factor for high attitude variation.

AOCS Feedback Loop study

A spacecraft attitude control system consists of attitude measuring sensors, actuators, electronics etc, in the hardware side and control algorithms, reference database etc, in the software side. A feedback control loop requires all these hardware and software for automatic control of the AOCS system. A stable feedback control loop is important for a successful operation of the attitude control system. The reference spacecraft orientation in ADS is defined by angles and angular rates with respect to time. The instantaneous attitude (angles with respect to an inertial reference frame and angular rates) is measured using sensors and instruments, defined in the measurements window. From these measured values and the reference data the required control torques are generated with correct magnitude and direction. The sensor-measured data are compared with the onboard reference data to measure the relative error. This signal is then passed on to the controller to operate different actuators. Possible sensor delays are embedded in the estimator algorithm and passed through filters like 'Kalman Filters' to improve the estimator output results. A block diagram of a simplified feedback loop is shown in Figure 6-15. As the spacecraft dynamics are already loaded to the control loop from the selected design, the AOCS designer needs to develop a stable control algorithms to the Controller and Estimator transfer functions.

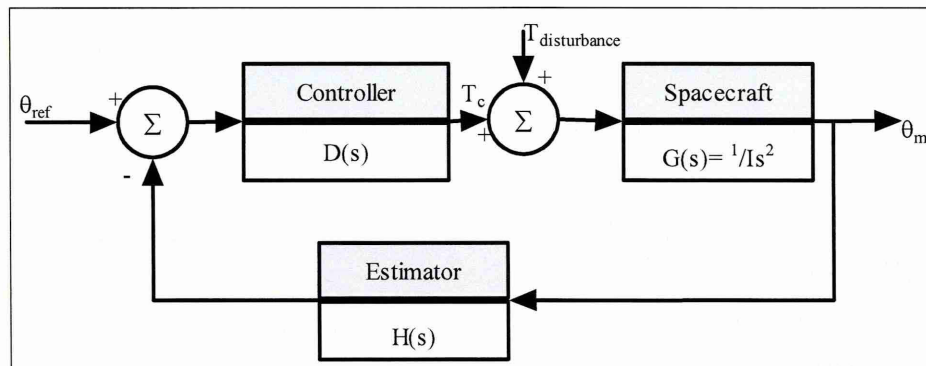


Figure 6-15 Simplified 'Simulink' attitude control feedback loop

Where,

θ_{ref} :	Reference attitude, angle for example
θ_m :	Measured attitude, angle for example
T_c :	Control torque
$T_{disturbance}$:	Disturbance torque
$D(s)$:	Controller transfer function
$G(s)$:	Spacecraft dynamics transfer function
$H(s)$:	Estimator transfer function
I :	Rotational axis moment of inertia
Σ :	Sum

The feedback loop shown in Figure 6-10 was designed using the classical frequency response method during this study. Each control loop (pitch, roll and yaw) was analysed separately for its stability. A simple PID (Proportional Integral & Derivative) controller was designed to generate the required control commands at the controller. A reasonably high damping ratio with a bandwidth of 0.2 rad/sec was selected as the main design criteria. The considered estimator transfer function was derived to simulate an operational and sensor measuring delay in angle and angular rate around 2 to 3 seconds. A simple feedback loop as shown in Figure 6-15 was used to derive the transfer functions for the ADS loop shown in Figure 6-7. The corresponding frequency responses for the satellite alone and for the satellite and sensor combination are shown in Figure 6-16. According to the magnitude of frequency response none of the systems (satellite with or without sensor lag) would be stable (slope greater than -20 dB per decade) for any proportional feedback control, if no derivative feedback were used. The frequency response diagram for the spacecraft clearly shows that it requires some shaping. The required stability was achieved using a derivative controller to bring the slope to -20 dB per decade at the crossover frequency as shown below (Franklin F. G. et al. 1994)

$$\text{Spacecraft transfer function: } G(s) = \frac{1}{Is^2}$$

Equation 6-11

$$\text{Estimator transfer function: } H(s) = \frac{1}{[(s+2)(s+3)]}$$

Equation 6-12

The crossover frequency was selected to 0.2 rad/sec and a value of ω_1 ($=1/T$, where T is a time constant) was chosen so that it is about 4 times lower than the crossover frequency. This gave the slope around -1 or -20dB in the vicinity of the crossover. The derived proportional, derivative (PD) controller is:

$$PD(s) = (T_D s + 1)$$

$$\omega_1 = \frac{\omega_c}{4} = \frac{0.2 \text{ rad/sec}}{4} = 0.05$$

$$T_D = \frac{1}{0.05} = 20 \text{ sec}$$

$$PD(s) = (T_D s + 1) = (20s + 1)$$

Equation 6-13

Where,

T_D : Derivative time constant

ω_c : Crossover frequency

ω_1 : Design frequency

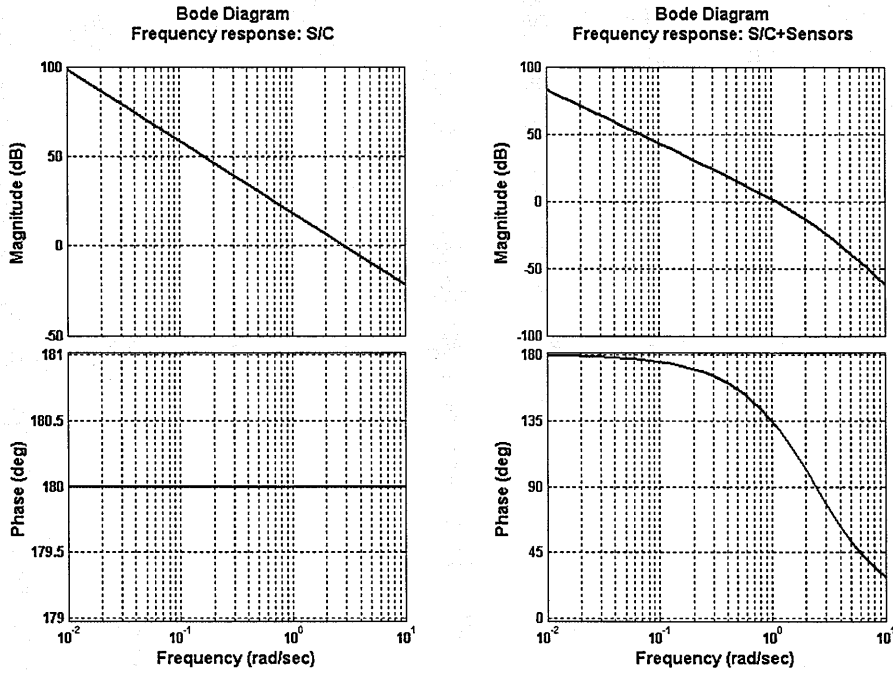


Figure 6-16 Frequency response only for the satellite (left figure) and satellite+ sensor system (right figure)

The transfer function denominator should be greater or equal to the numerator for satisfactory operation in the *ADS* control loop. Therefore a pole was added to the above PD controller at a frequency much higher than the breakpoint ($=1/T_D=0.05$ rad/sec.) of the PD compensator. The modified PD compensator transfer function given in Equation 6-14 is also called as the lead compensator.

$$PD(s) = \frac{20s + 1}{0.2s + 1}$$

Equation 6-14

The integral component of the PID regulator was designed by choosing $1/T_I$ (where T_I is the integral time constant), a factor of 20 lower than the $1/T_D$; that is, $1/T_I=2.5 \times 10^{-3}$. This left the only remaining task to determine was the proportional part of the PID controller, or K_p . In Figure 6-17, frequency response of the PID feedback with $K_p=1$ is shown. The final value for the proportional part was calculated from the magnitude of the frequency response in Figure 6-17-(right) at the crossover frequency, $\omega_c=0.2$ rad/sec.

$$|DIGH|_{\omega=0.2} = 41.5dB$$

$$20 \log M = 41.5dB$$

$$M \approx 120$$

$$K_p = \frac{1}{|M|_{\omega=0.2}} = \frac{1}{120} \approx 0.008$$

The final PID compensator is given in Equation 6-15 as a 'Laplace' function.

$$PID(s) = \frac{0.008}{s} \left(\frac{20s+1}{0.2s+1} \right) (s + 2.5 \times 10^{-3})$$

Equation 6-15

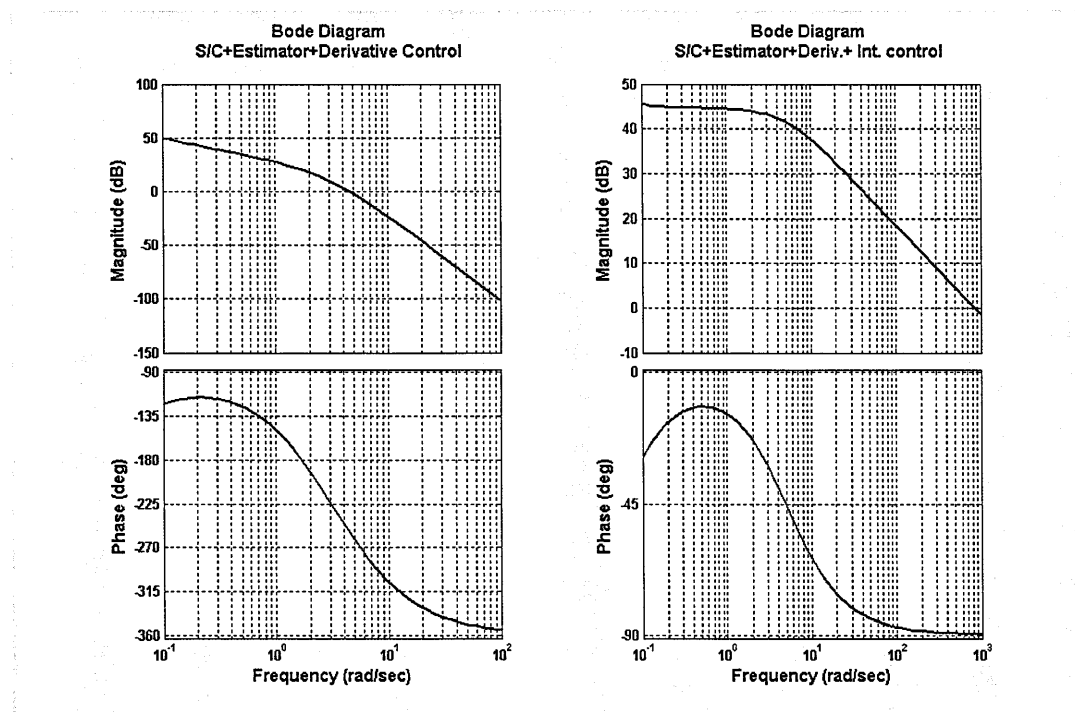


Figure 6-17 Derivative and Integral control frequency response with $K_p = 1$.

The corresponding frequency response of the final PID feedback loop is shown in Figure 6-18-(right). The open loop frequency response graph in Figure 6-18-(left) shows a 65° phase margin (PM) at the crossover frequency, which is 1 rad/sec in this case. However, at the designed crossover frequency of 0.2 rad/sec the PM is even somewhat higher than this. According to the step and impulse response test shown in Figure 6-19, the designed control system should provide a good stability and a damping to the system.

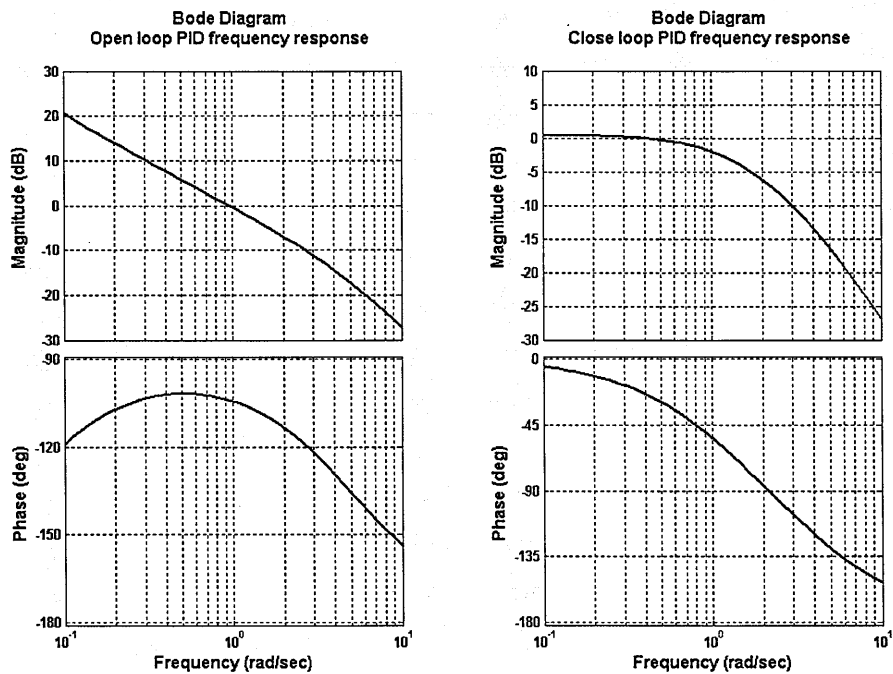


Figure 6-18 Frequency response of the PID feedback loop; $K_P=0.0084$.

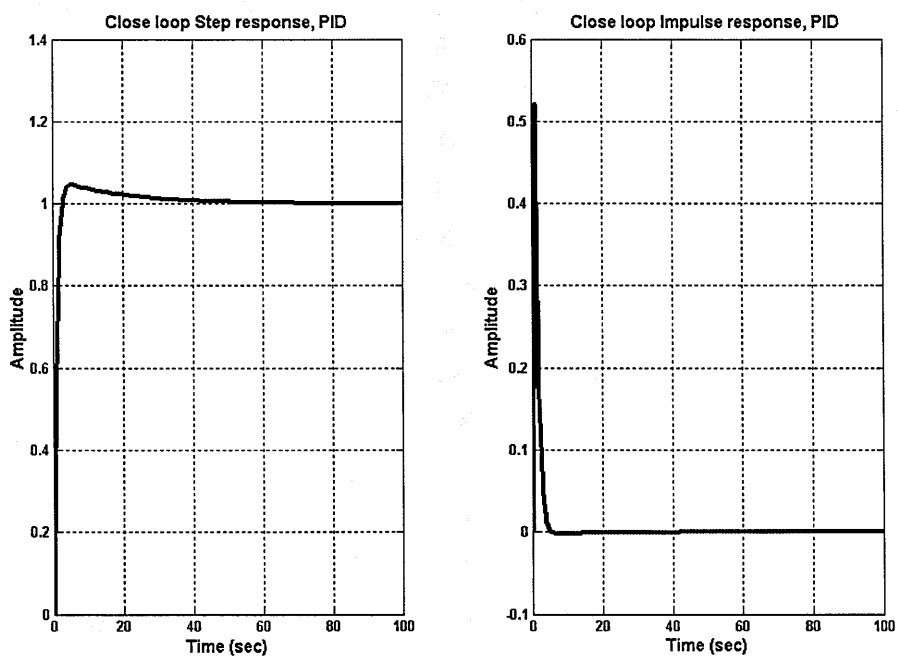


Figure 6-19 Step (left figure) and impulse (right figure) response for the PID feedback loop

The designed PID loop was tested using the Matlab/Simulink tool in order to understand its behaviour. The disturbance torques were derived from Equation 6-9 and ADS Fourier coefficients. The ‘Simulink’ control loop with the Mustang satellite inertia along the yaw axis is shown in Figure 6-20. Actuator saturation limit was set to 7.5 $\mu\text{N.m.}$ to keep in line with the ‘Mustang’ project that has a maximum limit for the FEEP thruster control torques.

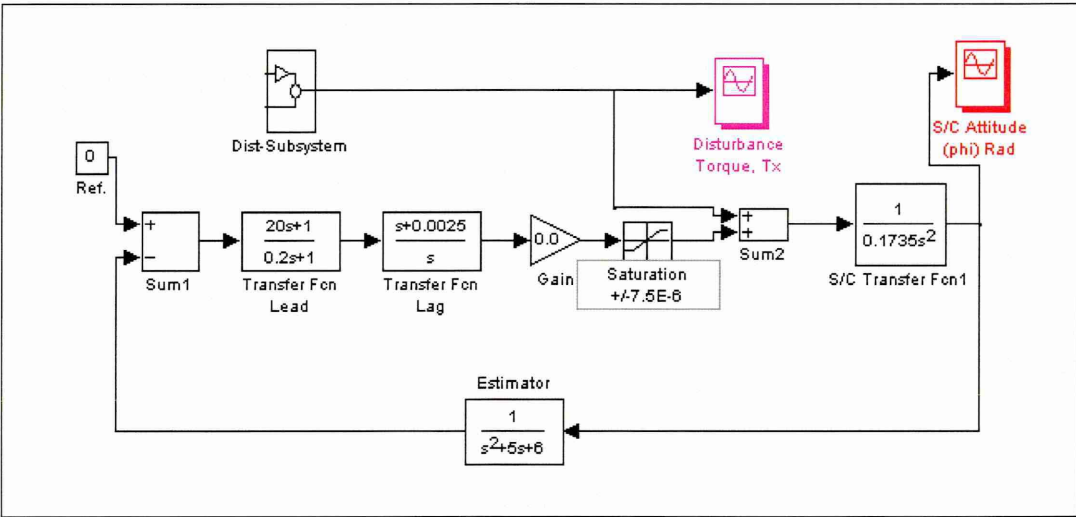


Figure 6-20 Simulink yaw control loop

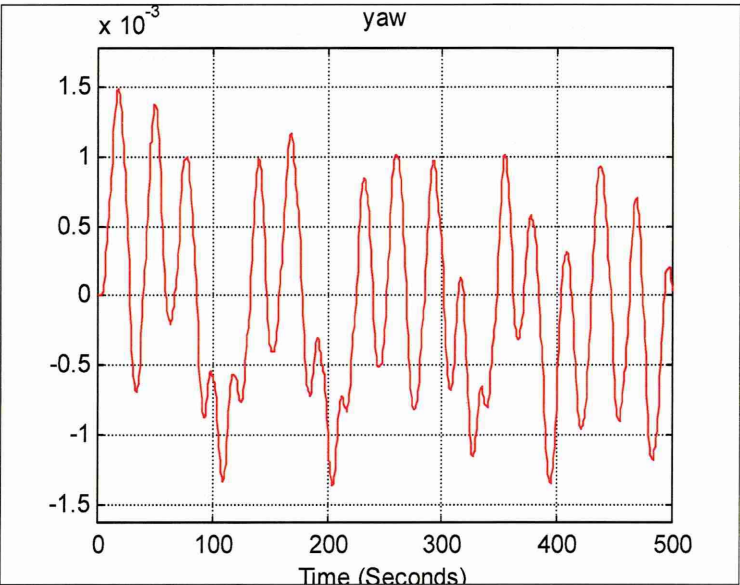


Figure 6-21 Yaw angle controllability using a PID controller in ‘Simulink’ (Angle in rad.)

The simulation results for the yaw angle using the control loop given in Figure 6-20 is shown in Figure 6-21. For the demand reference yaw angle of zero degrees with zero initial conditions, the corresponding yaw angle time distribution is shown in Figure 6-21.

Here the yaw error was limited to be below 0.01° with the designed PID controller. However, this result shows only the stability of the PID controller not the actual performance of a satellite attitude control system. Therefore, the total system attitude performance was analysed by applying the derived control loop in the ADS simulation tool.

The simulations were performed with respect to the 'Mustang' satellite and taking into account a standard set of requirements for the attitude control system. However in order to understand FEEP performance results in a more perspective way, it was compared with respect to 'reaction wheels' and 'magnetic torque-rods'. Though, these selected actuators may not best represent the vast majority of actuators available today, it is assumed here, that they are possible candidates for a nano-satellite missions. More details of these two actuators are given in Appendix-E.

ADS Equipment List

Attitude control simulations were performed considering a three axes stabilized spacecraft in a polar or highly inclined low Earth orbit. It was assumed that the spacecraft is Earth pointing with the actuators and sensor constrained as given in Table 6-5. FEEP thruster attitude control capability was compared with respect to micro reaction wheels and magnetic torque-rods. The performances were analysed in two different modes: disturbance rejection and slewing. The designed PID controller was used in the ADS feedback loop for all three-axis control with minor changes. The magnetic torque-rods and reaction wheels were applied along the roll (X), pitch (Y) and yaw (Z) axes. Eight FEEP thrusters in two clusters were used at the locations shown in the 'Mustang' design in Figure 6-8.

A 'Kalman' filter algorithm was used along with the rate sensors and gyro (GYR) data to optimise the angular rate estimations. The attitude control was performed by sending a series of time-tagged quaternion commands to the control block that in turn simply allows the satellite attitude to follow the reference command. The disturbance rejection mode was simulated assuming zero initial conditions (position, rate and angular momentum of the spacecraft). No internal disturbances were considered, as they are difficult to estimate at this preliminary design study. A residual magnetic moment value of 0.005 A.m^2 was assumed for magnetic moment calculations.

In order to test and verify FEEP thrusters performance in a space mission, the importance was identified of providing a disturbance free platform. It was assumed here that actuators such as magnetic torque-rods and reaction wheels could successfully be used to achieve a disturbance free platform. However in order to distinguish the FEEP thrust performance from the onboard actuators and environmental disturbances, it is important to have a precision attitude control system with very high accuracy.

Requirements	Constraints
Satellite class Inertia Mass	Earth pointing < 0.2 kg.m ² < 15 kg
Equipments Generic rate sensors ARS-12 Two axis Sun sensors Adcole-15486 Adcole-18394 One axis gyros QRS11	Pointing accuracy < 50 nano-radians < 0.5° < 2° < 0.004°/sec
Actuators Magnetorquers 3 x MT-5-2 Reaction wheel 3 x Dynacon Micro Wheel 200 FEEP 8 x FEEP-10 thrusters	Torque Capacity < 4.5 µN.m < 30 mN.m ± 50 µN.m < 4.5 µN.m

Table 6-5 Control mode fulfilled requirement and constrain list

Attitude measurements were performed with sun sensors, star trackers, rate gyros and rate sensors. These equipments were selected to meet various demands of a nano-satellite, such as mass, power and size. Table 6-6 summarises the characteristics of these selected equipments.

Equipment	Accuracy	Mass (kg)	Power (W)
Sun Sensors: 2 x Adcole#15486 1 x Adcole#18394	0.5° 2° at null	2 x 0.44 1 x 0.08	2 x 0.3 0
3 axis Angular rate sensor: 1 x ARS #12	50 nano-radians	0.1	0.5
One axis gyros: 3 x QRS#11	0.004°/sec	0.06	0.3
1 Star tracker LLNL star tracker camera specifications (Praxis Inc Clementine Web Site, 1994)		0.3	2
Micro wheel 1 x Dynacon Micro-wheel	± 50 µN.m	0.77	3
FEEP's 8 x FEEP thrusters in two clusters	Min. pulse: 10 ⁻⁸ N.s	2 to 10	3

Table 6-6 Selected AOCS equipments characteristics.

Star-tracker camera was a star-imaging sensor designed primarily for attitude determination, although it may also use to collect scientific data. The selected star tracker use silicon CCD technology with a 384 x 576 pixel array sensitive between 0.4 and 1.1 micrometer. It has a wide field of view, 29 degrees x 43 degrees, to make more stars available for attitude determination. Although with increase in field of view (FOV) the star tracker accuracy will reduce somewhat, it will decrease the number of stars that must be catalogued for successful operation of the sensor. Its instantaneous field of view (IFOV), which can be defined as the field of view the individual pixel element in the focal plane (see Figure 6-22), is measured to 1.3 mrad. Typically attitude determination could be made within 100 ms using a single star tracker image. The LLNL star tracker and Stellar Compass provided attitude determination of medium accuracy, which is better than 150 micro-rad in pitch and yaw, and 450 micro-rad in roll.

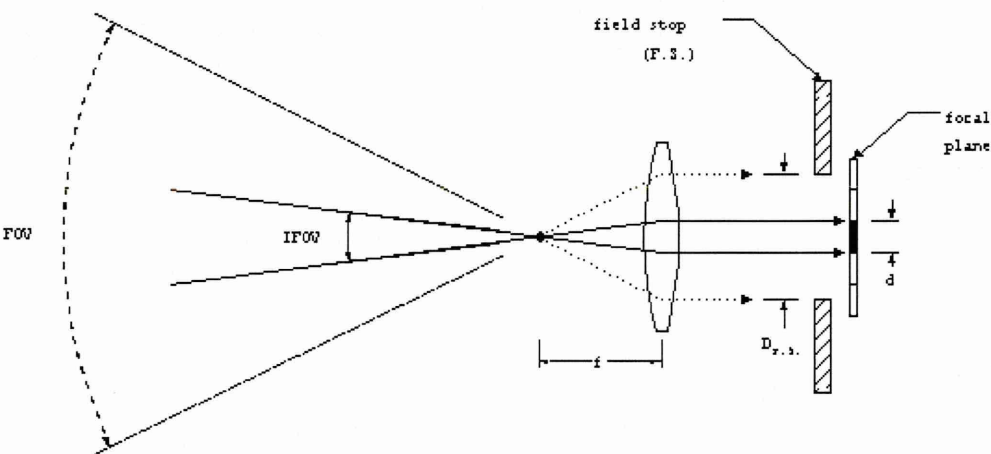


Figure 6-22 A star tracker Field of View (FOV) and Instantaneous Field of View (IFOV)

Mass	290 grams
Size	12 x 12 x 14 cm
Power	4.5 W
Field of view	29 x 43 degrees
Pixel format	384 x 576

Table 6-7 LLNL Star tracker camera specifications (Praxis Inc Clementine Web Site, 1994)

6.5 Summary

Different simulation tools were explained that were used during this study. Formation flying analyses were performed using the *STK*, *Astrogator* for high accuracy propagation analysis. It took into account all major perturbations (Aero, Solar, oblate and third body) acting both on the target and the interceptor satellite. Further *STK*, *Astrogator* facilitates the simulation of FEEP thrusters by selecting a continuous thrust model and to visualize the results using various options. The 3-D visualisation object gave the first insight to

understand a situation and the final results were verified from reports and graphs. This simulation tool gave a faster and reliable method for analysis of a complex situation. Attitude control analyses were better performed using the *ESA/ADS* tool. This had a serious limitation that it was not compatible with other programs and therefore could not link to transfer results from one platform to another. This tool was used to model design, orbit analysis, control mode and to investigate the total system.

The Matlab/Simulink tool was used to derive a stable PID controller for the ADS control loop. A PID controller was designed using frequency response design method. The controllers' stability was analysed using *Bode's gain-phase* relationship. The performance of the *PID* controller was analysed using Matlab *Step* and *Impulse* response methods. The final results were used in the ADS for attitude control analysis. However for stable operation an inbuilt *Kalman filter* was used together with the designed *PID* controller. Simulations were carried out with the selected actuators and instruments characteristics to identify possible limitations in the system. Importantly all these simulation tools have its limitations with respect to FEEP simulations. For example the FEEP minimum impulse of 10^{-5} to 10^{-8} N.s is hard to model with the used simulation tools and the considered nano-satellite is in the preliminary study phase. Further this study has not focused on AOCS but a feasibility study on FEEP thrusters.

7 FEEP Performance Analysis

This section analyses the performance capability of FEEP thrusters, considering a 'Mustang' type nano-satellite as the standard. Although characteristics of the FEEP thruster have been analysed in chapter 4, this research is not complete without the analysis of the total system (Spacecraft/Propulsion system). This is fulfilled here, using a possible FEEP demonstrator mission. FEEP performance was analysed in both attitude and orbit control modes to understand the FEEP thruster efficiency during various sequences of the mission. It is assumed that the work here will pave the way to apply FEEP thrusters more effectively in the future, and to be aware of performance boundaries of this propulsion system (mainly when using in conjunction with nano-satellites). Certainly, some of the limitations of the FEEP thruster operations discussed here, may be relaxed in the future, with increased availability of power per unit system mass, and better development of this thruster.

7.1 General

The FEEP thruster demonstrator mission was analysed in the following three segments: separation from the parent satellite, free propagation to a safe distance, FEEP thrust phase to perform either attitude or orbit control manoeuvres. FEEP thruster demonstrator mission onboard a nano-satellite (Mustang) was used to simulate the thruster attitude and orbit control capabilities. For simplicity an 800 km circular orbit was selected as the mission reference orbit. The satellite physical parameters were selected as of 'Mustang', an octagonal 10 kg nano-satellite, for disturbance force analysis. A detail description of the 'Mustang' satellite is given later in this chapter. FEEP thruster (FEEP-10) attitude control performances were mainly compared with respect to reaction wheels (Dynacon MW-200) although magnetic torquods (Ithaco TRIUPN) were used in this study for comparison due to its high popularity among many university nano-satellite programs, such as Swedish 'Munin' satellite (Strömdal, 2001). However magnetic torquods were not incorporated in the final proposed FEEP demonstrator mission, as it does not meet the required attitude control accuracies (less than 0.5 deg) for a quantitative analysis of FEEP performances.

Separation from the parent vehicle

It is assumed that the FEEP demonstrator satellite (Called 'Interceptor' during constellation analysis) will be launch as an attached entity to a parent satellite (called 'Target' satellite). The required ' ΔV ' for the separation may achieve either through passive means such as spring load detachment or active means such as from FEEP thrusters. However the applied separation ' ΔV ' should be small (few centimetres per second), in order to achieve a close proximity (less than 100 meters in range) formation flying with simple onboard tracking systems.

Free propagation

For the analysis of the considered formation-flying mission, it is assumed that the ‘Interceptor’ satellite must propagate behind the ‘Target’ satellite at a defined distance within an acceptable error (few meters) margin without obstructing the ‘Target’ orbit. For a stand-alone mission, this is the un-powered propagation time or the drift period: Time between separation from the launch vehicle until FEEP thruster operation.

FEEP thrusting phase

It is assumed here that FEEP thruster performances may demonstrate either in attitude or orbit control mode. In attitude control mode FEEP performance is analysed considering simple rotational (roll, yaw or pitch) manoeuvres. It can also be a manoeuvre to spin up or down in conjunction with reaction wheels. The orbit control capability of the FEEP thrusters are best demonstrated using a formation-flying mission scenario. This type of an analysis when both satellites are identical is of interest as most of the planned missions falls in to this category. However the situation when the two satellites are dissimilar is addressed separately.

7.2 FEEP-10 Characteristics

The selected FEEP emitter characteristics are discussed here to provide some background knowledge of its performances. The FEEP-10 emitter, which has a 10 mm long slit, was selected mainly due to its low power demand compared to the longer slits as well as the data availability from ground testing. Slit emitters with slit lengths between 2 mm and 70 mm are being offered, spanning a thrust range from 40 μ N to 1.4 mN. Table 7-1 summarises some of the properties of FEEP-10 together with FEEP-5 and FEEP-70.

Manufacturer	Centrosazio (Italy)	Centrosazio (Italy)	Centrosazio (Italy)
Model name	FEEP-5	FEEP-10	FEEP-70
Propellant	Cs	Cs	Cs
Slit length, mm	5	10	70
Nominal thrust, μ N	100	200	1400
Isp, s	9000	9000	9000
Power at nominal thrust, W	6.6	13.2	93
Specific Power, W/mN	66	66	66
Comments	Under development	Under development	Qualification model

Table 7-1 State-of-the-art FEEP thrusters, (Micci and Ketsdever, 2000)

Caesium was selected as the working propellant despite its high reactivity with residual oxygen, mainly due to its low ionisation potential (see Appendix C-1) and hence low power demand. Considering the possible risk of propellant contamination mainly due to the Caesium reaction with residual Oxygen, it was emphasised the importance of installing the total FEEP system in a sealed container. Different precautionary methods were discussed in section 4.

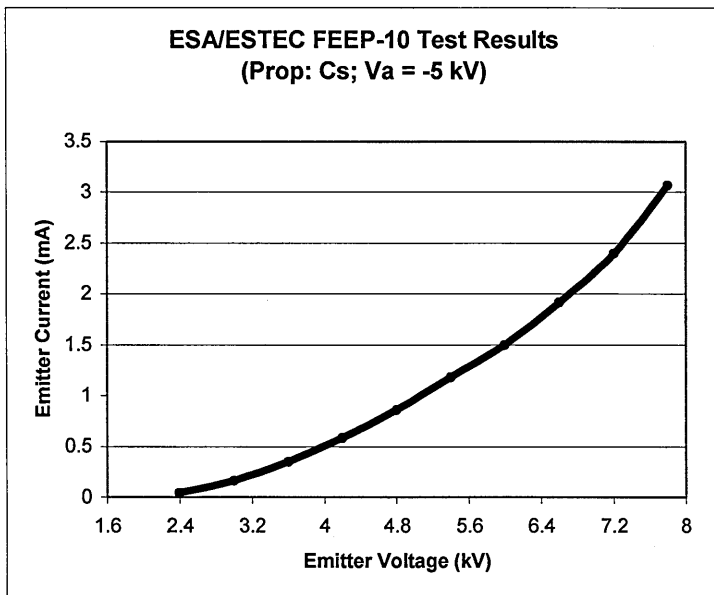


Figure 7-1 FEEP-10 Emitter voltage and current relation {Courtesy of ESA/ESTEC}.

The electrical characteristic of a typical FEEP-10 emitter is given in Figure 7-1, showing drawn out emitter current (I_E) for different emitter voltages (V_E). The FEEP propulsion system has the advantage that it can be operated both in pulse and continuous thrust modes. The required thrust profile is achieved by varying the applied emitter voltage. Operational boundaries of the selected thruster were estimated considering the limits of nano-satellites, such as mass, power, possible areas of applications etc. Considering a general case study, various demands on the propulsion system were analysed to meet different attitude and orbital control manoeuvres satisfactorily. The selected FEEP-10 emitter maximum operational thrust, number of thrusters and their satellite mounting locations were selected in order to:

- Operate the thrusters within the allocated power budget (< 5 W).
- Counteract the disturbing forces acting on the satellite, and maintain its attitude and orbit references.
- Provide 3-axis attitude control authority.
- Operate satisfactorily during the entire mission by avoiding excessive stress.
- To prevent the s/c and other sensitive instruments from propellant contamination.
- Give a free flow passage to the plume.

Thruster orbital control capability was analysed either in order to maintain a given reference orbit or to follow an un-powered reference satellite within a close proximity. The latter one, which is actually a collaborative control mission, will be addressed in detail as many advanced missions are planned to operate in this manner. In order to impart orbital corrections with a low thrust propulsion system, such as FEEPs, require long operational times. While the energy requirement for the correction of spacecraft attitude, depends more or less on the physical properties of the satellite (the size and mass of the spacecraft that determines the s/c moment of inertia), the orbital correction manoeuvres are mainly coupled to the orbital elements (eg. Orbit inclination, semi-major axis). Therefore it seems that FEEP thrusters are much more attractive, (from the nano-satellite point of view) as an attitude control thruster. This does not necessarily mean that the FEEP thrusters should not be used in conjunction with orbital correction manoeuvres. As it is true for many low thrust devices, FEEP is more attractive for fine pointing and collaborative control manoeuvres. It is assumed here that the FEEP propulsion subsystem mainly consists of following components:

- 8- FEEP thrusters in 2 cluster units.
- Each FEEP-10 thruster with a total of 5 grams of propellant (Cs) in its built-in tank.
- Built in heaters to keep the Caesium in liquid form.
- One power control unit attached to each cluster (total of two).
- One neutralizer unit for each cluster (total of two).

The *ESA/ESTEC* FEEP-10 thruster electrical (measured) and thrust (calculated) characteristics are given in Figure 7-1 and Figure 7-2 respectively. Importantly the electrical characteristics are highly depending on the physical properties of the FEEP emitter such as the slit length, width, wetting characteristics etc. Therefore any variation of the FEEP emitter physical properties and the selected propellant characteristics will give different electrical and thrust characteristics curves even though the applied electrical conditions are same. The main reason for possible dissimilarity in FEEP emitter units is that it is hard to manufacture identical thrusters with the current manufacturing technology, as FEEP is a precision thruster. Further any variations in the vacuum chamber test conditions will give rise to deviations in test results. Therefore these characteristic curves are unique for a particular FEEP emitter unit. The selected FEEP emitter in this study, which is the *ESA/ESTEC* FEEP-10, had been tested with a constant accelerator voltage of 5 kV negative, for varying emitter voltages. The amount of current drawn from the emitter and accelerator were measured for different emitter voltages. As the delivered thrust is difficult to measure practically due to its low thrust, μN scale, and various practical reasons with the experiment set up, it was calculated analytically from the measured currents and voltages. In general the current drawn out from the accelerator is much less than the emitter current. Therefore for initial electrical power demand estimations, the power drawn out from the accelerator is generally ignored. However for low emitter voltages, the power demand from the accelerator must be taken into account. The current voltage (I - V) curve in Figure 7-1 shows that it requires a total potential difference of about 7.5 kV ($= V_E - V_A$) between the emitter and accelerator electrodes to initiate the ion emission. According to the laboratory test experience gained from

Centrosazio and ESA/ESTEC, it can be argued that the optimum operational range for this emitter is in its low voltage spectrum. One of the main reasons for this rationalisation apart from the required power and other factors discussed in section 4, is that at higher emitter voltages, the emitter current increases very rapidly. This rapid increase in current may cause damage to the emitter slit and accelerator electrode due to high potential field build up. Further at high potential differences between the emitter and accelerator, the possible danger for electrode short-circuiting increases. If this happens, the emitter and accelerator electrodes could be damaged, causing surface irregularities along the emitter slit. These surface irregularities may disturb or stop the propellant flow towards the emitter slit either partially or completely. However, at low operating voltages it is possible to have some operational irregularities, especially during the initial start up process. This is mainly due to an un-even wetting of the slit and possible irregularities in ion emission sites along the slit. Further the FEEP thrusters' efficiency increases at low power operational range. The main reason for this is that at low power FEEP thrusters can be operated at its optimum, at low temperatures and less neutral propellant evaporation giving rise to high efficiency factors. Further at high powers the flow divergence losses becomes significant as ion density in the flow increases.

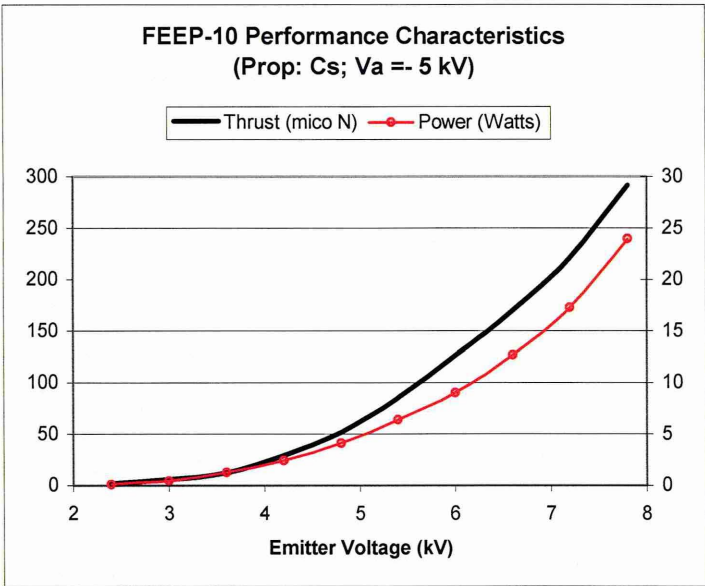


Figure 7-2 Calculated Thrust (in left axis) & Power demand (in right axis) as a function of emitter voltage for the FEEP-10 emitter. {Courtesy of ESA/ESTEC}.

It has been shown that the FEEP thrust scales linearly with emitter length (Marcuccio et al. 1997), with a nominal thrust density of 20 –25 $\mu\text{N/mm}$. Therefore longer slit emitters deliver high thrust. This in turn requires more power from the system. As the thrust varies proportional to the applied power, a longer slit emitter draws more current than a shorter slit, at a given emitter voltage. Generally components carrying high currents are much bulkier and cause excessive power losses ($Loss = I^2 R$). Therefore it is advantageous to use shorter slits, if it serves the purpose. The 10 mm long emitter (FEEP-10) was

preferred here instead of the shorter (5 mm slit length, FEED-5) and longer ones available, mainly due to the following reasons:

- During a possible partial contamination of emitter slit, the FEED-10 may still provide adequate thrust for the mission.
- The FEED-10 emitter had been extensively ground tested both at *ESA/ESTEC* and *Centrosazio*, Italy.
- Fewer complications during manufacturing, cleaning and assembling process.
- The built in propellant tank in the FEED-10 emitter is adequate for most of the proposed missions.
- Have a low power requirement compared to the longer slits, (operation, propellant heating, neutraliser etc.).
- Faster wetting times compared to longer slits.
- Reduced edge effect compared to a smaller slit.
- The maximum achievable thrust limit, which is in the order of 200 to 300 μN (Marcuccio and Genovese, 1997), is more than adequate for the planned FEED demonstrator mission.

7.3 System Study

As a baseline design for mission analysis and simulation, the 'Mustang' model (Figure 7-3), an octagonal structure with its basic dimension of 314 x 300 x 300 mm, was selected to represent a nano-satellite. For analysis, a satellite body fixed reference frame was selected where the X-axis is along the velocity vector, Z-axis towards Earth and the centre at the centre of the body (see Figure 7-3). The total mass of the satellite is assumed 10 kg before launch. An average power level of 10 W was assumed available at all times, from which 5 W is allocated for the FEED propulsion system. However, for an actual mission, these values should be regarded as flexible within the limits of a nano-satellite power budget. Low priority was given to the demonstrator satellite communication subsystem as it was assumed that all communications are relayed through the 'Target' satellite. Importantly the communication architecture onboard the 'Interceptor' satellite will be limited due to the constraints in mass, power and development cost associated with nano-satellites. It is further assumed here that the 'Interceptor' satellite will be launched attached to the 'Target' satellite as a single entity. A mission time of few weeks was selected, as the main interest here is to demonstrate and validate the FEED propulsion system. Although this will seriously reduce the data acquisition and operational experience of FEED, it is within the interest of FEED nano-satellite initial study. A mission lifetime of few weeks was assumed adequate to perform all the mission sequences, download the on-board data and to allow enough time for the validation of the experimental data on ground, prior to mission termination.

Keeping in mind the near term solutions, off-the-shelf components are more feasible as for attitude control hardware. Therefore, as much as possible these components generic data (physical and dynamic characteristics) are used for s/c attitude and orbit control simulation study. It is assumed that the electrical power demand of the spacecraft is met

with primary batteries alone or a combination of solar arrays and secondary batteries, depending on the length of the mission. Importantly the selected power source should ensure the attitude and orbit control system (AOCS) power demand during the entire orbit. A regulated 28 V DC bus is assumed to be available for the components to be plugged in. This modular approach was preferred as it allows different spacecraft subsystems to operate using 'plug and play' type devices. For simplicity and robustness, simple body mounted solar arrays are preferred against the complex deployable arrays. The octagonal shape of the body (Figure 7-3) was chosen to facilitate more surface area and volume for solar cell and hardware mounting. Further it allows better structural integrity and easy access to components through panels.

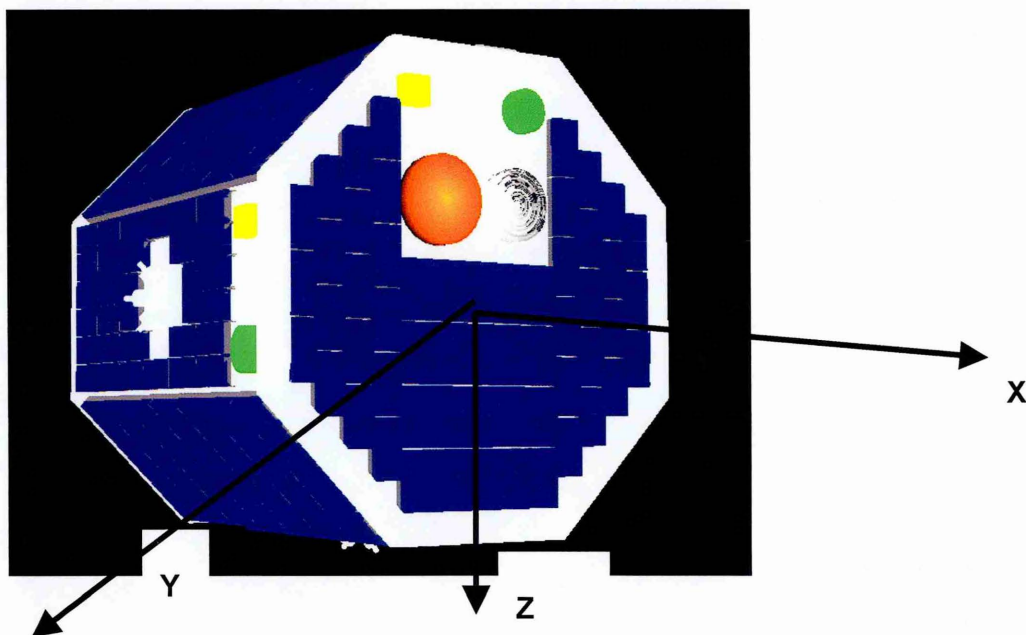


Figure 7-3 Selected nano-satellite model: 'Mustang' (Courtesy of Cranfield University 'Mustang' Group)

It was assumed that the satellite structure to be manufactured as a single unit, allowing access through the front and back panels for test and integration of subsystems. In line with this modular approach, the ASTRIUM designers (Marston, 2000) have already manufactured and tested a standard octagonal shape structure for small satellites around 100 kg mass category. For the 'Mustang' satellite structure, lightweight was achieved by using 'carbon fibre' tubes (filament wound) and 'honeycomb' panels for equipment mounting. By selecting the direction and position of the fibre it is possible to manufacture structures for optimum load handling. This composite manufacturing technology can be used for nano-satellite development with minimal modifications. It is believed that this approach may shorten the time of production and reduce the development costs in the future. Importantly the success of nano-satellites depends on the possibility to manufacture in large quantities, achieve comparatively low cost (25% to 50% savings compared to small satellites), shorter manufacturing time (between 1 to 3 years), and the

ability to adapt for new missions with minimum modifications (within 6 months). Therefore the modular manufacture and integration approach is an important issue. This way of manufacturing and integration reduces the manufacturing and development complexity while giving flexibility to the satellite platform. This is especially true for scientific nano-satellites, which are normally a combination of flight proven off-the-shelf components with new technologies.

Attitude and orbit control simulations were performed in this study using the *ESA/ADS* and *STK* simulation software tools respectively. The validity and accuracy of these simulation results mainly depends on how closely the selected input parameters represent the actual mission data. The *ESA/ADS* software tool is also used to analyse mass and power budgets in subsystem scale. Important parameters such as the spacecraft inertia matrix and spacecraft centre of mass are derived from the subsystem scale analysis with respect to a ‘body fixed’ coordinate system. These results were then used in both *STK* and *ADS* simulations. A typical mass breakdown summary in subsystem scale is given in Table 7-2. A detailed description of the selected, ‘off the shelf’ components and their generic characteristics are given in Appendix-D.

Sub-System	Main Components	Component Mass (grams)	Mass percentage
Structure	Body, Launcher/Target Interface, S/C Separation actuators etc.	900	9%
Electrical Power (EPS)	Solar Array, Battery, Power Control Unit, Power distribution Unit, Switchers and Regulators	2200	22%
Communication	Transmitter & Receiver	500	5%
Command & data Storage	On-board Computer	750	7.5%
AOCS	Star Scanner, I/O Drive Assembly, Reaction Wheels (Dynacon), Sun Sensors and Electronics, Target sensor Rate sensor	3330	33.3%
Propulsion & Thermal	8 x FEEP Thruster Units, Electronics (2 boxes) Neutralizers, Heaters Propellant (Cs = 5 grams) Thruster protection system	2320	23.2%

Table 7-2 Mass breakdown for ‘Mustang’ type FEEP demonstrator spacecraft (All approximate values) with modular subsystems

The mass breakdown for the ‘FEEP demonstrator’ satellite can be compared with the proposed mass breakdown for the actual ‘Mustang’ satellite (Roberts, 2001). According

to the Mustang mass breakdown (structure 19%, power 15.14%, communication 3.2%, command and data storage 4%, AOCS 22.92%), FEEP demonstrator satellite has a comparatively low structural mass. However for all thrusters system, to perform three-axis attitude control, together with the requirement for target tracking (formation flying) and separation interface, the allocated structural mass is a rather ambitious figure. With the current material and structural technology, the development of a nano-satellite to meet those set objectives of this study will be a technological challenge at least for the next couple of years.

As shown in Figure 7-4 a single reaction wheel together with eight FEEP thrusters were selected for attitude and orbit control study. Eight FEEP thrusters are used here to achieve full three-axis control of the spacecraft allowing some redundancy. However it should be noted that it is adequate to have six thrusters for full three-axis control. The extra two thrusters were used here to give redundancy to the mission as well as to generate pure torques for *ADS* attitude simulation purposes. Simulations are also performed for a ‘hybrid system’ with FEEP thrusters and a single reaction wheel placed along the pitch axis (it was assumed that the pitch axis has low coupling effect with respect to roll and yaw axes). The spacecraft inertia matrix and the centre of mass offset (from the coordinate system origin) were derived from the *ESA/ADS* analysis tool. A summary of the FEEP demonstrator satellite and its physical properties are given in Table 7-3. FEEP thrusters were mounted in two clusters, with each cluster unit consists of all the required components including electronics. FEEP system mass was slightly over estimated here to compensate for the required protecting shield around the thrusters. Protecting the FEEP emitter slit and its propellant (Caesium) from contamination are identified as one of the most important issues for a trouble free operation of this thruster.

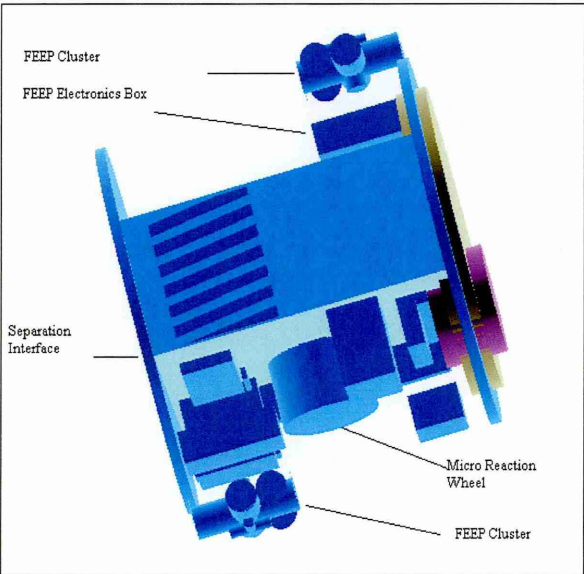


Figure 7-4 Two FEEP cluster units and the single reaction wheel system.

Shape	Octagonal
Size (X,Y,Z)	[314, 300, 300] mm
Launch Mass	10 kg
Dry Mass	9.995 kg
Propellant (Cs)	0.005 kg per thruster
Drag area	0.075 m ²
Drag coefficient	2.0
Solar pressure area	0.095 m ²
Reflection coefficients:	
Specular	0.5
Diffuse	0.2
Absorption	0.3
Residual magnetic moment	0.005 Am ²
Centre of Gravity Position with respect to S/C body frame. Derived using ADS V-2	[-1.745; 4.828; 5.9] mm
Inertia tensor matrix at COG (kg.m ²)	$\begin{bmatrix} 0.132 & & \\ 0.03411 & 0.1174 & \\ 0.004863 & -0.001299 & 0.1735 \end{bmatrix}$

Table 7-3 Physical properties for the ‘Mustang’ nano-satellite

Figure 7-4 shows the attitude control simulation model of the ‘Mustang’ S/C. It also shows the locations of the two FEEP cluster units and the single reaction wheel for 3-axis attitude and orbit control. Taking into account each component location with respect to a reference coordinate system and the mass of the component, the spacecraft inertia tensor was calculated. Table 7-3 summarises these information.

7.4 Estimation of Disturbances

For the sizing of spacecraft propulsion system, it is important to estimate the disturbance forces acting on the s/c, which perturbs its orbit and orientation. The discussion of external forces was limited to atmospheric drag, solar radiation pressure, Earth’s oblateness and magnetic field as these are the major perturbation forces. The third-body gravitational affects causes due to Moon, Sun, planets, etc, were ignored, as these affects are more prone to high altitude orbits.

Disturbance Torque Analysis

Gravity Gradient Torque

A spacecraft body experiences a gravity-gradient torque due to the variation of distances between the spacecraft mass points and the Earth mass centre. A torque arises from the fact that the lower end of the spacecraft subjects to exponentially higher gravity forces than the upper end. This force effect due to gravitational field variation for long extended bodies are only a factor for low earth orbiting (LEO) s/c. The gravity gradient torque produces torque in two axes and calculated as of Equation 7-1.

$$T_{gg} = \frac{3\mu}{r^3} |I_z - I_y| \theta$$

Equation 7-1

Where,

- T_{gg} : gravity gradient torque, (N.m)
- μ : Gravitational parameter, ($\approx 398600.4 \text{ km}^3/\text{s}^2$ for Earth)
- r : Radius from s/c centre of mass to central body centre of mass, (km)
- I_z : Moment of inertia about z axis, (kg.m^2)
- I_y : Moment of inertia about the y axis (for maximum torque use the least of I_y or I_x), (kg.m^2)
- θ : Angle between s/c z-axis and nadir vector, (rad)

-For a 800 km altitude with s/c moment of inertia as: $I_{xx} = 0.132 \text{ kg.m}^2$, $I_{yy} = 0.1174 \text{ kg.m}^2$, $I_{zz} = 0.1735 \text{ kg.m}^2$

-Maximum gravity gradient torque for a 5 deg attitude error: $T_{gg} = 1.6 \cdot 10^{-8} \text{ N.m}$

Aerodynamic Torque

This effect is due to an offset between the centre of mass (CM) and drag centre of pressure (CP). This is only a factor in LEO. The velocity vector is along the body x-axis ($V_{\text{circular}} = 7.45 \text{ km/s}$). The aerodynamic torque is calculated from Equation 7-2, and the greatest uncertainty is in the atmospheric density.

$$T_{aero} = \frac{1}{2} \rho V^2 C_D A L$$

Equation 7-2

Where,

- T_{aero} : Aerodynamic torque, (N.m)
- ρ : Atmospheric density, (kg/m^3)
- V : Spacecraft velocity, (m/s)
- C_D : Drag coefficient depends on shape but usually about 2.0 – 2.5
- A : Area normal to the velocity vector, (m^2)
- L : Distance between the centre of pressure and centre of mass, (m)

-If it is assumed a maximum C.G offset of 0.02 m for each axis and atmospheric density, $\rho = 9.41 \cdot 10^{-14} \text{ kg/m}^3$ (800 km altitude) (Wertz and Larson W. J., 1999) and other parameters as of Table 7-3 then:

-Maximum aerodynamic torque: $T_{\text{acro}} = 7.9 \cdot 10^{-9} \text{ N.m}$

Solar Radiation Torque

This is induced due to the offset between the centre of mass (CM) and solar centre of pressure (CP). The total solar torque on the body caused by a spacecraft surface is calculated as of Equation 7-3.

$$T_s = PAL(1 + q)$$

Equation 7-3

Where,

T_s : Solar torque on the spacecraft caused by a surface A, (N.m)

P: Solar pressure, ($= 4.59 \times 10^{-6} \text{ N/m}^2$, for near Earth orbits)

A: Area of surface projected to sun line normal, (m^2)

L: Distance from the centre of pressure of the surface to the centre of mass of the spacecraft, (m)

q: Reflectance factor between 0 and 1, unit-less. Spacecraft bodies tend to be reflectors, q = 0.5 is representative; Solar panels tend to be absorbers, q = 0.3 is representative (Wertz and Larson W. J., 1999).

-If assumed a CM and CP maximum offset is 0.01 m for each axis (will vary around the orbit), and taking s/c properties as of Table 7-3 then,

-Maximum solar radiation pressure torque: $T_{\text{sol}} = 6.6 \cdot 10^{-9} \text{ N.m}$

Residual Magnetic Torque

-As it is difficult to estimate the residual dipole for the Mustang satellite to calculate for Residual magnetic torque, the following comparison was adopted, as shown in Table 7-4. For example the residual dipole will depend on the internal wiring configuration (eg. Twisted pairs of conductors will have no effect).

$$T_M = NIAB \sin \theta = DB \sin \theta$$

Equation 7-4

Where,

T_M : Magnetic torque, (N.m)

N: Number of loops in the coil

I: Current in the coil, (amp)

A: Coil area, (m^2)

B: Earth's magnetic field, (tesla)

θ : Angle between the magnetic field lines and perpendicular to the coil

-If residual magnetic dipole $D = 0.005 \text{ Am}^2$, $B = 40 \text{ } \mu\text{T}$ and $\theta = 90 \text{ deg}$
-Maximum magnetic torque: $T_M = 2.0 \cdot 10^{-7} \text{ N-m}$

Satellite	Mustang	Picosat	ROCSAT-1	ROCSAT-2
Reference	(Roberts et al. 2002)	(Anderson et al. 2000)	(David, 1998)	(David, 1998)
Average Power (W)	7.5	2.5	200	400
Voltage (V)	28	5	28	28
Current (A)	0.27	0.5	7.2	14
Size (m ³)	0.314x0.3x0.3	0.1x0.1x0.1	1.0x1.0x2.0	1.6x1.6x2.0
Average Area (m ²)	0.09	0.01	1.5	2.9
Current x Area (Am ²)	0.024	0.005	10.8	40.6
Residual dipole (Am ²)	0.005	0.001	1	10
Ratio of the above two items	4.8	5	10.8	4.6

Table 7-4 Residual Magnetic Moment Study

Table 7-4 shows the adopted procedure for estimating the residual magnetic dipole for the ‘Mustang’ model. The analysis was performed here comparing with three other nano-satellites. A low value for the dipole ratio represents high current density (current per unit volume, A/m³), which is in line with the Mustang electrical (FEPP) thrusters.

The magnitude of the earth magnetic field was estimated using the NSSDC (National Space Science Data Centre) simulator (Bilitza, 1995). This disturbance effect is substantial at low altitudes as the earth magnetic field is proportional to $1/r^3$, where r is the spacecraft orbital radius as given in Equation 7-5, (Brown, 2002). As shown in Figure 7-5 the maximum magnetic field strength is approximately 40 μT at 800 km altitude for a polar orbit.

$$B = \frac{B_0 r_0^3}{r^3} (3 \sin^2 L + 1)^{1/2}$$

Equation 7-5

Where,

- B: earth’s magnetic field strength at any altitude or latitude
- B₀: Earth sea level magnetic field strength, (approximately 3 x 10⁻⁵ tesla
- r: Spacecraft orbital radius, (m)
- r₀: Earth surface radius, (=6378.14 km)
- L: Latitude in magnetosphere, (deg)

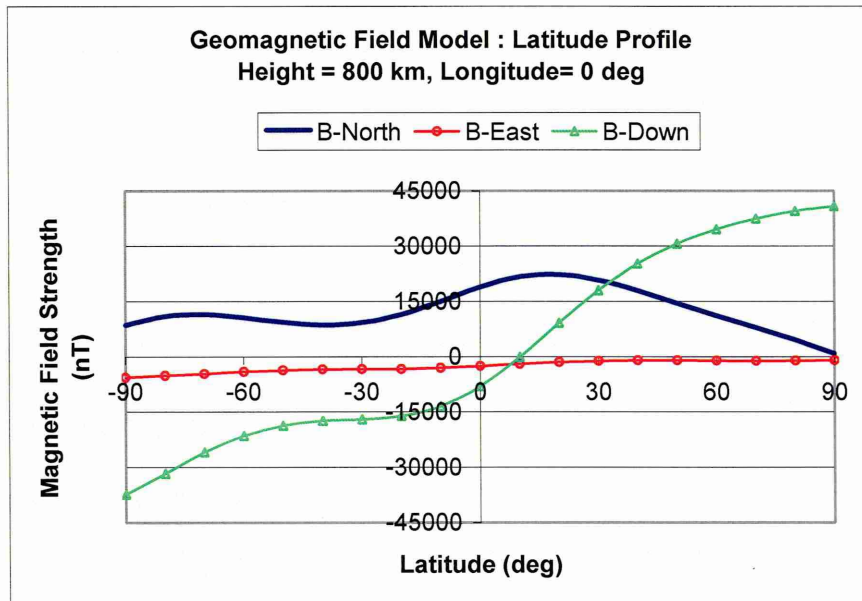


Figure 7-5 Earth magnetic field strength model as a function of latitude (Credit NASA/NSSDC)

Disturbance Force Analysis

At low Earth orbits the earth atmosphere, though it is tiny, imposes a drag force, which in turn alters the reference orbit. Beside this solar radiation pressure force and third body attraction (Sun, Moon, etc.) alters the s/c reference orbit. Although these forces are very small, in the long run they deviate the s/c from its intended orbit and attitude considerably. Therefore, these forces must take into account for any mission design and their effects may have considerable impact on a formation-flying mission. This is especially true if the s/c have different configurations, such as frontal area, mass, material properties etc, in a constellation. However for this study the analysis is limited to air drag and solar radiation pressure forces, as they are the major contributing perturbation forces at the low Earth orbits of interest. As shown below their magnitudes are considerably low and they fall within the selected FEEP thrusters control force capabilities. Importantly if these forces are not corrected for long time period, they will cause high ΔV changes to a satellite in its operational orbit.

Aerodynamic

- Atmospheric density, $\rho = 9.41 \times 10^{-14} \text{ kg/m}^3$ (800 km altitude)
- Maximum Value: $4.8 \times 10^{-7} \text{ N}$

Solar Radiation

- Absorption and reflection coefficient are 0.4 and 0.6 respectively
- Maximum Value: $6.6 \times 10^{-7} \text{ N}$
- (Minimum value = 0 in eclipse)

The *ESA/ADS* tool was used to estimate the disturbance torques. The disturbance torque analysis was limited to aerodynamic, solar, gravity gradient and magnetic forces. The magnitudes of these torques for a Mustang type satellite is shown in Table 7-5. The selected *X*, *Y* and *Z* orthogonal body fixed reference frame has its origin at the spacecraft centre (see Figure 7-3). The vector directions are selected so that the *Z*-axis (yaw) points towards the Earth centre and *X*-axis (roll) pointing along the velocity vector. According to Table 7-5, the magnitudes of the disturbance torques are at least a factor of 10 less than maximum available control torque (7.5 $\mu\text{N.m}$) from the selected FEEP thrusters. However this control torque maximum is only a selected value due to the power limitation set for the ‘Mustang’ satellite, not a limitation from the thruster itself. This shows that the FEEP-10 thrusters are adequate for use as a disturbance counteracting actuator. However, magnitude of these disturbance torques will be considerably high for low perigee altitude orbit with increase surface area. These disturbance torques are compensated at first, as bigger s/c offers longer moment arm providing increase control authority. Importantly when a s/c gets bigger, more powerful FEEP thrusters are required to counter act the disturbance forces acting on it. In practice this is possible to achieve by selecting a longer slit emitters.

As shown in Table 7-5, the source of the maximum disturbing force is due to the magnetic moment. However the magnetic dipole moment depends on the electrical characteristics of the spacecraft. According to the Table 7-4, the ‘Mustang’ dipole moment is well within the range of other similar spacecraft. It should be noted here that as ‘Mustang’ satellite has FEEP thrusters and other electrical components, the s/c current density (A/m^3) would be considerably high as can be expected. The magnetic disturbance effect varies according to the selected orbit as well. Its strength reduces for high altitude and low-inclined orbits (equatorial). Selection of higher altitude orbits further reduces the aero and gravity gradient torques. The gravity gradient and magnetic torques properties can be used to provide control torques to the spacecraft. For nano-satellites with body mounted solar cells it is difficult to reduce the solar pressure torque as the cells have a high reflection coefficient. However for nano-satellites the disturbance effects due to aero and solar torques are considerably small in altitudes of concern. From the design point of view, a well-balanced satellite is less prone to disturbance torques, as the centre of mass and centre of pressure off set is minimal. However for an Earth pointing satellite this statement will not hold, as the interest is to maintain a gravity gradient to meet the pointing demand.

	$T_x \text{ (Nm)}$	$T_y \text{ (Nm)}$	$T_z \text{ (Nm)}$	Percentage
Aero	1.84×10^{-9}	3.29×10^{-9}	2.58×10^{-9}	0.9
Solar	1.97×10^{-8}	2.80×10^{-8}	1.69×10^{-8}	7.7
Gravity Gradient	1.61×10^{-8}	1.57×10^{-8}	1.52×10^{-8}	5.6
Magnetic	1.92×10^{-7}	2.48×10^{-7}	2.82×10^{-7}	85.8

Table 7-5 maximum magnitude of torques along the *X*, *Y* and *Z* axis, based on a ‘Mustang’ type satellite

7.5 FEEP Attitude Control Mode

The control capabilities of FEEP thrusters were studied mainly taking into account two separate modes: Attitude and Orbit control. However, the FEEP thruster demonstration tests could be better performed in attitude control mode (s/c rotation) than the orbit control mode (formation-fly). The main reason for this is that the attitude measurements can be performed with a greater accuracy than the relative orbital parameter measurements. However for the success of the FEEP thruster demonstration mission, accurate angular and relative distance measuring methods are of importance.

Disturbance Rejection Mode

The reference attitude of an orbiting spacecraft changes continuously due to the presence of disturbing torques. These torques are considerably strong at low altitude Earth orbits. For a nano-satellite of type ‘Mustang’, typical attitude development in pitch, roll and yaw axes are shown in Figure 7-6 due to the presence of external disturbance torques. In a low earth orbit (~800-km), magnitude of the total disturbance is in the order of $5.0E^{-7}$ Nm. Though these torques are quite small, they do a considerable attitude change for a nano-satellite, if no counteracting measures are taken. Beside these external torques, any anomalies within the control actuators may have adverse effect during control. Therefore the selected AOCS system should have enough control authority to counteract the disturbing torques while still maintaining its attitude control demands. This is analysed in the rest of the section using FEEP thrusters and other control actuators to provide the required control torques.

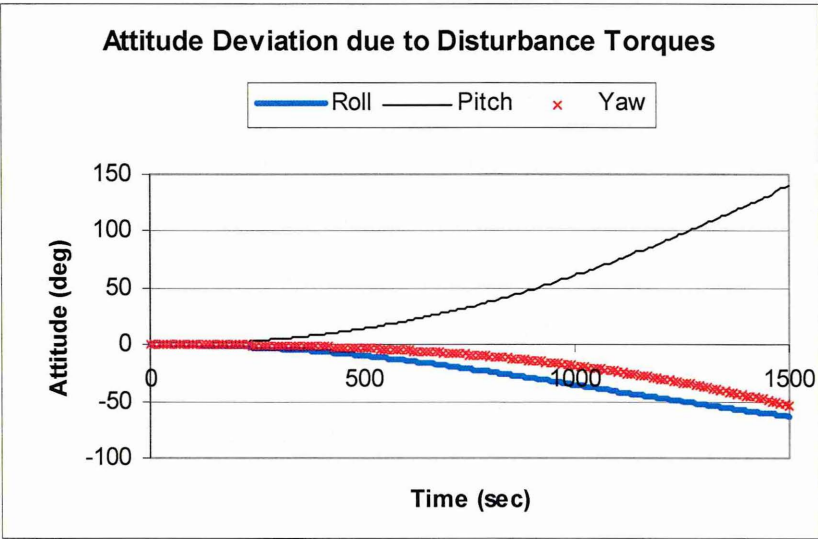


Figure 7-6 S/C attitude variation due to disturbance torques, for zero initial conditions (top to bottom: Pitch, Yaw & Roll) (ESA/ADS Simulations).

For a possible FEEP demonstrator mission, the accuracy and stability of the demonstrator platform will be a decisive factor for the success of the mission. Onboard sensors and instruments should be able to distinguish FEEP control torques from the disturbance environment. Attitude control simulations were performed with respect to magnetic torque rods and reaction wheels in conjunction with FEEP thrusters to meet the defined ACS demands. These control hardware were selected to meet the nano-satellites' general power and mass demands. Simulations were performed with different actuators to identify their control capability. The selected actuators (reaction wheels and magnetic torque rods) were placed along the spacecraft principle axes to control the pitch, roll and yaw. FEEP thrusters were used to provide roll, pitch and yaw axis control on demand. Simulations were performed for each control system: reaction wheels, magnetic torque rods and FEEP thrusters, separately. These actuators were selected in order to enable the validation of FEEP thrusters with respect to different control systems that use in many space missions. A combination of reaction wheels and FEEP thrusters were selected for a possible FEEP demonstrator mission as this may help to perform the FEEP demonstrator mission within a limited time period. For an example, the s/c may allow to rotate using wheels while thrusters can be used to counteract this manoeuvre. Further FEEP thrusters can be tested efficiently during reaction wheel off-loading. The control actuators given below are used to perform s/c attitude control using the 'Mustang' as the reference satellite.

- Three magnetic torqrods (MT) of type TR1UPN 'ITHACO' (see Appendix-E).
- Three reaction wheels (MW) of type 'Dynacon' MW-200 (see Appendix-E).
- Eight FEEP-10 thrusters in two clusters (FEEP).

The main reason to address the torque rods and reaction wheels in this FEEP control study is that in a possible demonstrator mission one or both of the above control hardware may exist. This is especially true due to the operational uncertainties encountered with FEEP thrusters at the present stage of development. Further, these systems give a reference platform for the FEEP test results validation. The *TR1UPN* 'ITHACO' (see Appendix-E) magnetic torque rod, has a maximum attainable torque of 50 $\mu\text{N.m}$ at an altitude of 800 km. Beside this, TR1UPN has a comparatively low weight (100 grams) and power (0.18 W) requirement, which is an added advantage from the nano-satellite design point of view. However in order to generate the maximum torque, the rods residual current must be perpendicular to the Earths' magnetic field. Therefore during most of the orbital period, the attainable torques may be much lower than the theoretical maximum. Torque rods have been used successfully in the past mainly for nadir pointing to Earth. Together with magnetometers (measure Earth magnetic field strength and direction), torque rods (produce magnetic dipole moments) can be used for course control of Low Earth orbiting spacecrafts. However due to the presence of their iron core, magnetometer/torque rod assembly tend to be heavier (IM-102 magnetometer: 220 grams), and their linear output range is limited by the natural magnetic saturation. Importantly there are two basic designs of magnetic torquers. Torque rods are cylindrical rods of ferromagnetic material with many turns of wire wrapped around them. When current is applied to the rod the core becomes magnetised. Similarly some spacecrafts uses 'Torque Coils', that are constructed of many turns of wire without a core. When a

current is applied it creates a dipole as a result of magnetic field it creates in empty space. For example the STRV-1a and STRV-1b (Space Technology Research Vehicle) satellites uses torque coils.

FEEP application as a disturbance rejection thruster, was analysed with respect to micro ‘Reaction Wheels’ and ‘Magnetic torque rods’. Typical pitch control capability with respect to these control actuators are shown in Figure 7-7. Though the spacecraft performance along the pitch axis is illustrated in Figure 7-7, analysis for the roll and yaw axes can be done in a much similar way. For a free propagation without any control torques, the spacecraft attitude deviates quickly from its reference as shown in Figure 7-6 even when the initial attitude rates are zero. This shows the importance for control actuators for attitude corrections. If the FEEP thrusters are to be tested on board a technology validation space mission such as SMART-2, it is important to have a disturbance free platform for this purpose. On the other hand if the FEEP thrusters are to be used as an attitude control thrusters in a stand-alone mission, the selected thrusters should be powerful enough to overcome the disturbance torques and perform the attitude control demands satisfactorily.

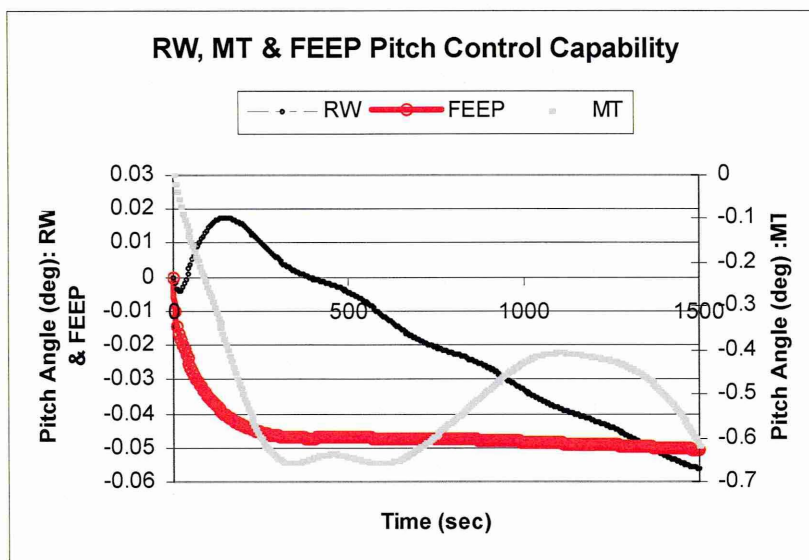


Figure 7-7 Pitch control capability using RW, FEEP (Left Axis) & MT (Right Axis) (zero initial conditions).

For the selected spacecraft (MUSTANG) the motion in pitch axis with respect to time is shown in Figure 7-7. This shows that the FEEP thrusters and reaction wheels provide much finer control capability than the selected magnetic torque rods (TRIUPN). However in order to use torque rods in nano-satellites it is necessary to custom build them according to the need. Therefore TRIUPN torque rods were omitted from the analysis here, as it is not compatible with nano-satellites (MUSTANG) and FEEP control characteristics.

The simulations were performed in a force field that consist of aerodynamic, solar radiation, gravity gradient and magnetic field disturbance torques. However the internal disturbance torques from actuators and sensors were not considered, as they are not defined at this stage of the study. For high accuracy applications, such as disturbance free platforms like LISA, these internal disturbance effects must take into account. This is especially true for reaction wheels that cause internal disturbances due to its flywheel rotation. Disturbances associated with flywheels are caused by the imbalance originated from:

- The offset of the centre of gravity of the flywheel from its rotation axis (Static)
- Angular misalignment of principal inertia and spin axis (Dynamic)
- Induced disturbance at frequency of wheel spin
- Bearing disturbances
- Motor disturbances
- Induce vibrations at higher harmonics of the wheel speed

The considered FEEP thruster system, which has no central propellant tanks, fuel pumps, injectors etc, is ideal for a disturbance free platform as there will be no internal disturbances as above. However attention must be given to the possible thrust vector misalignment that might cause due to many reasons. In order to provide a disturbance free platform, the attitude control actuators must compensate the disturbance torques acting on a satellite. As shown in Figure 7-8, the selected reaction wheels and FEEP thrusters control torques are well below their set maximum (Dynacon Micro Wheel-200: 30 mN.m; FEEP: 7.5 μ N.m). By looking the pattern of the commanded torque in Figure 7-8 it is evident (no high fluctuations of disturbance torque) that few Hz of switching frequencies for disturbance counter action with FEEP thrusters are adequate. Typical values for high voltage regulator's switching frequency limit are around 30 kHz (see Appendix D).

The selected reaction wheel system with an average power requirement of 2.1 W and a minimum power requirement of 0.6 W is well within the set power budget for the propulsion system. However, the internal disturbance effects and the higher power demand will still be a deciding factor for its selection. The total system mass either for the reaction wheel system or FEEP thrusters will be around 2.5 kg to 3.0 kg. For FEEP thrusters with power to thrust ratio about 60 W/mN (without neutraliser power), it requires about 0.5W to operate two thrusters in parallel providing about 4 μ N from each thruster.

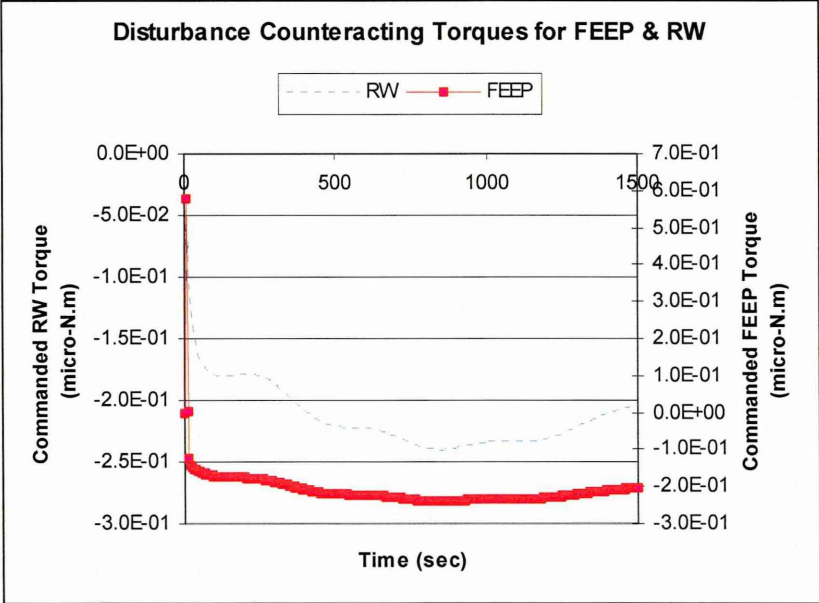


Figure 7-8 Commanded disturbance counter acting control torques from RW (Left Axis) & FEED (Right Axis)

In order to provide a disturbance free platform the control torques should be able to follow the disturbance torques. This needs continuous attitude knowledge and accurate attitude determination techniques. If these demands can be satisfied as shown in Figure 7-9, the commanded control torques ‘follows’ quite well the disturbance torques acting on the satellite. Simulations are performed with zero initial conditions. Control torques are demanded to maintain a zero reference attitude and attitude rates. The thrusters were operated in continuous operation mode. Compared with ‘bang-bang’ attitude control technique (thrusters fire whenever the satellite strays from an angular window), it may be advantageous to use FEED thrusters in a continuous operation mode. The main reason for this rationally is due to the irregular thrusting and start-up delays that might occur due to delay in wetting and high voltage circuit time constant. However continuous thruster operation requires uninterrupted sensor attitude updates and supply of power.

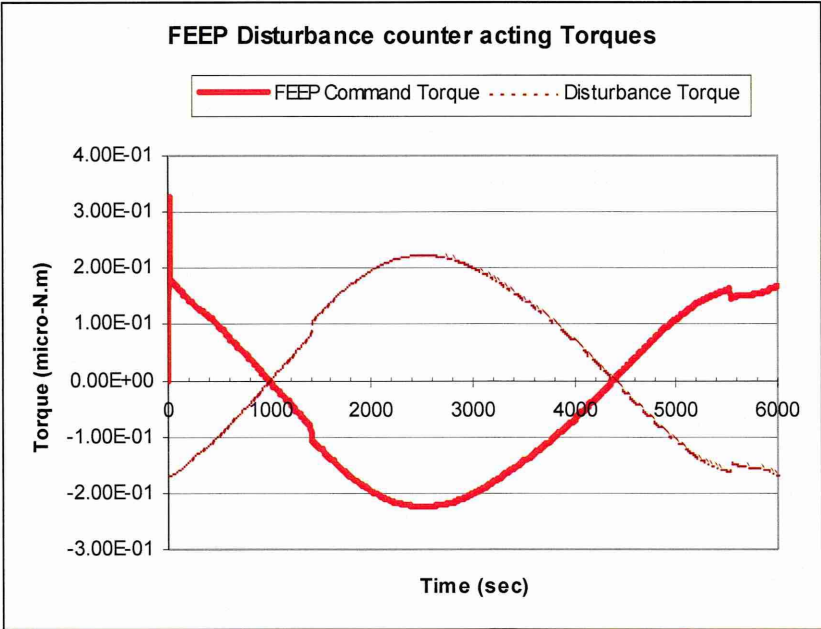


Figure 7-9 FEEP thruster application in disturbance rejection mode in pitch axis

Slew Mode

FEEP thrusters and reaction wheels were analysed for their performances for a typical one-axis rotation. Applying a torque rotates spacecraft, and the torque profile is varied according to the slew requirements. For analysis purpose it is considered an acceleration deceleration torque profile as shown in Figure 7-10. This gives half of the required rotational angle within half of the period according to Equation 7-6 for a simplified one-axis rotation, when there are no other external torques.

$$\left(\frac{\theta}{2}\right) = \frac{T_v}{2I_v} \left(\frac{t_m}{2}\right)^2$$

Equation 7-6

Where,

θ : Slew angle (rad)

T_v : Torque (N.m)

I_v : Spacecraft moment of inertia with respect to the slew axis (rotational axis) (kg.m²)

t_m : Total manoeuvre time (sec)

Actuators such as thrusters (eg. FEEP) can be used to generate a torque profile as that of Figure 7-10, by first operating it to generate a positive torque and then reversing to generate a negative torque. If thrusters are to be used, this requires two different sets of thrusters (for pure rotation it requires thrusters in pairs). The above analysis does not take

into account external disturbance torques and for analysis purpose the applied torque was limited to $0.75 \mu\text{N.m}$. This is exactly one tenth of the available maximum FEEP torque ($7.5 \mu\text{N.m}$). The time dependent slew profile will look as that of Figure 7-11. This shows that a 'Mustang' type spacecraft can be rotated at best about 18 deg within 1500 seconds using one tenth of available torque. However spacecraft are subjected to a number of external torques (aerodynamic, solar pressure, gravity gradient and magnetic). The presence of these external disturbances may either increase (acts along the control torque direction) or decrease (acts against the control torque) the manoeuvre time. In practise the disturbance torque profile varies both in magnitude and direction when the spacecraft travels around its orbit. Therefore in order to understand the manoeuvre requirements it requires a detail 3-D simulation analysis using a tool such as ESA/ADS.

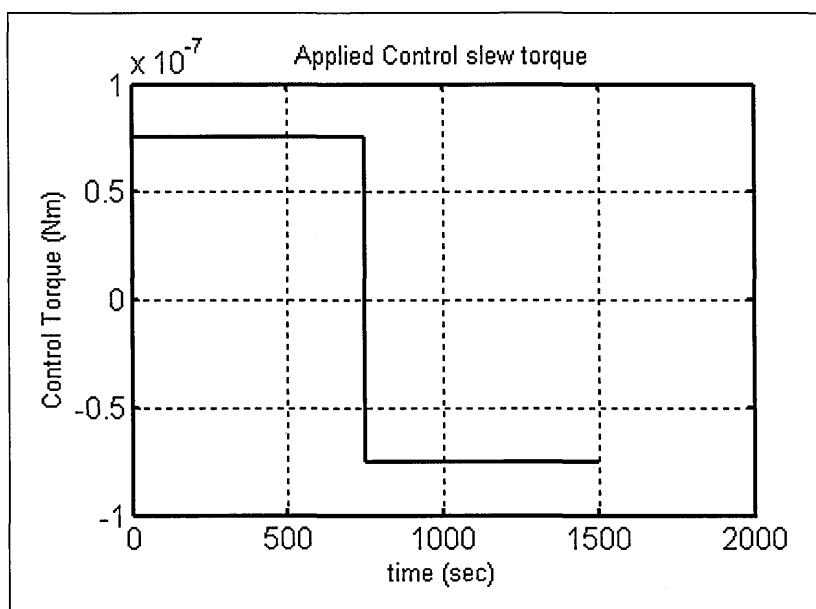


Figure 7-10 Acceleration/deceleration control torque (selected maximum = $0.75 \mu\text{N.m}$)

For many science missions the objective of a slew manoeuvres is to rotate the spacecraft payload through a small angle. This requires precision torque pulses, which is in-line with FEEP characteristics. However the disadvantage of using thrusters in generally for slew manoeuvres of above type is that it requires multiple (two or more) thrusters to generate an acceleration/deceleration torque profile.

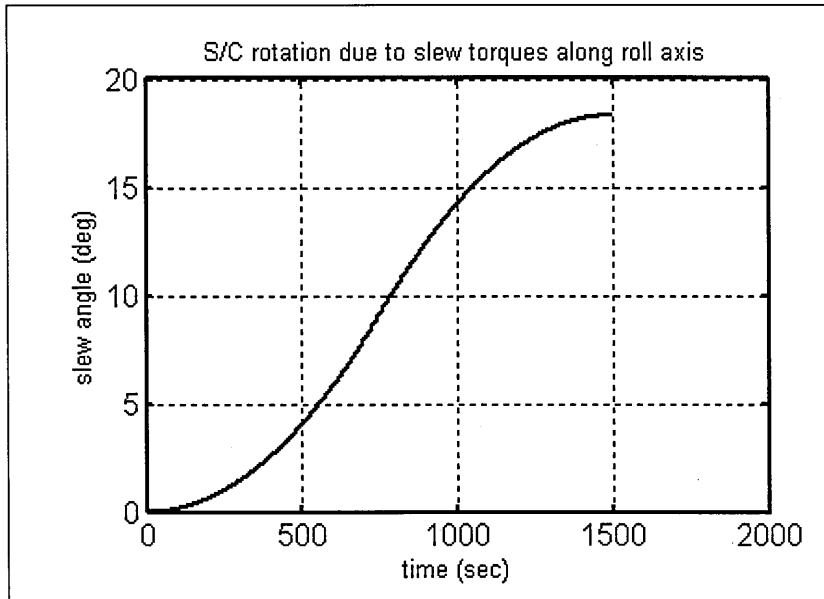


Figure 7-11 Spacecraft rotation due to acceleration/deceleration torque ($I_v = 0.132 \text{ kg.m}^2$)

To perform a manoeuvre of above type using a single reaction wheel (one axis), the flywheel in the reaction wheel must first accelerate in one direction and then decelerate to its initial state. The spacecraft will rotate in the opposite direction of the flywheel rotation. If there are no energy dissipation (eg. Due to external torques) then at the end of the manoeuvre the total wheel angular momentum change will be zero. The magnitude of the wheel-generated torque is a function of the flywheel moment of inertia and its angular acceleration as of Equation 7-7.

$$T_v = \alpha_w I_w$$

Equation 7-7

Where,

T_v : Torque (N.m)

α_w : Wheel angular acceleration (rad/s^2)

I_w : Wheel moment of inertia (kg.m^2)

A reaction wheel of type 'Dynacon Micro Wheel-200' (Mass = 0.77 kg, wheel diameter = 102 mm: $I_w = 0.001 \text{ kg.m}^2$) will generate a torque of magnitude $0.075 \text{ } \mu\text{N.m}$ when accelerated the wheel to about $7.5 \times 10^{-5} \text{ rad/s}^2$. The maximum angular momentum build up will be less than $0.57 \text{ } \mu\text{N.m.s}$. This is much lower than the saturation value (0.05 N.m.s) of the considered reaction wheel. Although a single reaction wheel can perform a one-axis slew manoeuvre, due to energy dissipation (external disturbance torques) the wheel angular momentum will drift to saturation. This requires unloading of the stored wheel angular momentum using for example thrusters (or magnetic torque rods).

FEEP thrusters as well as reaction wheels are suitable for slew manoeuvres and to counteract the disturbance torques. However FEEP thrusters have the attractiveness that it

can be used for precise payload slew operations without requiring a secondary actuator. Further FEEP thrusters generates a much smaller impulse bit (minimum impulse bit) compared to the selected reaction wheel. For example for the considered micro reaction wheel (Dynacon Micro-Wheel 200) control torque pulse is around $\pm 50 \mu\text{N.m}$ averaged over one second. This implies that the reaction wheel will give a control accuracy of 0.016 deg along the rotational axis (pitch axis, averaged over 1 second). If the control accuracies of reaction wheels and FEEP thrusters are compared, the latter provides a 100 to 1000 times improvement. For scientific platforms reaction wheels together with FEEP thrusters provide a good attitude control system as the thrusters can be used for fine control and wheel de-saturation. However, the total system mass and power demand will increase with such a combined system. For the FEEP thruster validation mission, FEEP/RW system will provide a better reference platform.

The above highly simplified analysis considered a pure rotation along an axis and does not takes into account the effects of disturbance torques. In order to understand the effect from external disturbance torques it is necessary to solve the Euler's moment equations given in Equation 7-8 in three scalar forms (a more detail analysis is given in (Sidi, 1997)).

$$T_x = I_x \dot{\omega}_x + \omega_y \omega_z (I_z - I_y)$$

$$T_y = I_y \dot{\omega}_y + \omega_x \omega_z (I_x - I_z)$$

$$T_z = I_z \dot{\omega}_z + \omega_x \omega_y (I_y - I_x)$$

Equation 7-8

Where,

T_x, T_y, T_z : External torques along the x, y and z principle axes

I_x, I_y, I_z : Moment of inertia along the principle axes

$\omega_x, \omega_y, \omega_z$: Angular rotational rates along the x, y and z axes

$\dot{\omega}_x, \dot{\omega}_y, \dot{\omega}_z$: Angular acceleration along x, y and z axes.

As there are coupling terms in Equation 7-8, a pure rotation along one axis without controlling the other axes are not possible for an unsymmetrical spacecraft ($I_x \neq I_y \neq I_z$). Although Equation 7-8 does not have an analytical close form solution (equations are non-linear), it can be solved for one axis as given in Equation 7-6 (assumed no disturbance torques and coupling effects had been neglected).

The performance of different torque profiles in a disturbance free environment is studied for a simplified case assuming that the selected body coordinates are the principal axes. The Euler's moment equation given in Equation 7-8 is solved in one axis to find the time dependent angular motion (eg. pitch angle) and rate (eg. pitch rate) of the body for known initial conditions. The system response is studied for cyclic and pulse control torques as shown in Figure 7-12 and Figure 7-13 with zero initial conditions. The corresponding

changes in pitch angle and pitch rate are shown in Figure 7-12 and Figure 7-13 respectively. This shows with a limited magnitude of control torque availability, a much higher slew angle can be achieved by applying a pulse torque instead of a sinusoidal. However this requires more energy, which may not be a limiting factor with electrical thrusters. With FEEP thrusters it is possible to achieve different control torque profiles either by varying only the emitter voltage or both emitter-accelerator voltages.

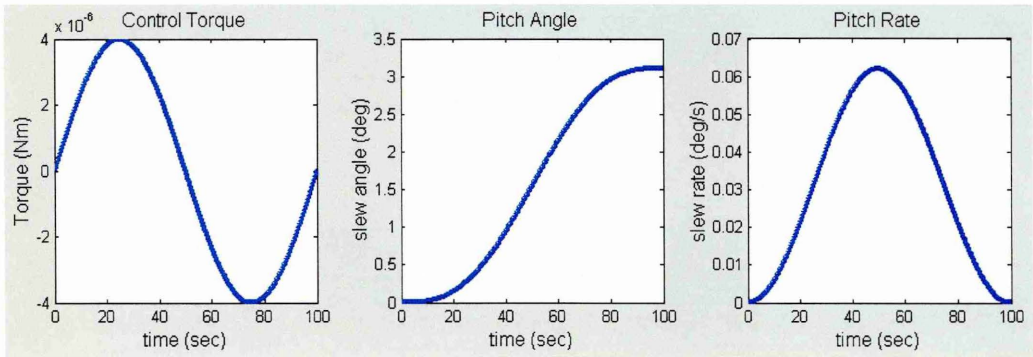


Figure 7-12 Sinusoidal control torque time-domain pitch motion

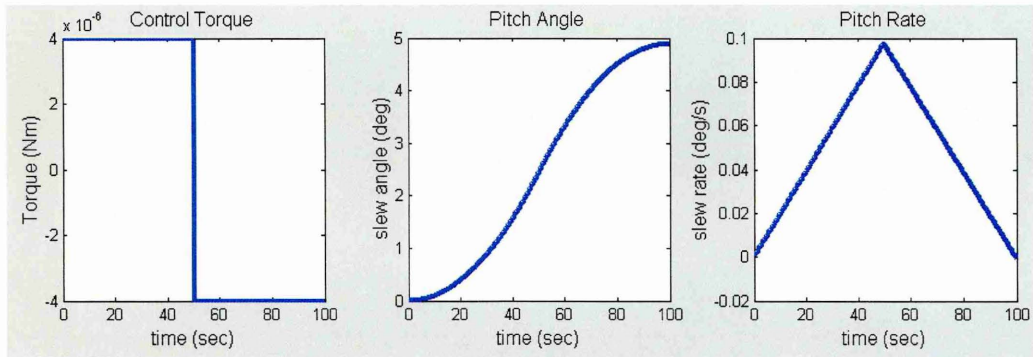


Figure 7-13 Pulse control torque time-domain pitch motion.

To understand the coupling and disturbance torque effects for slew manoeuvres ESA/ADS simulation tool was used. To achieve the min-max torque profile, in the ADS control loop slew angle and slew rate were selected as reference signals. Simulations were performed using a 800 km circular polar (90 deg inclination) orbit. In order to achieve a pure rotation (roll), thrusters locations were selected along the YZ-plane. Figure 7-14 shows the thrusters control and the disturbance torque profile to achieve the required 18 deg (approximately) slew manoeuvre (see Figure 7-15). The control torque and the slew rate distortion visible in Figure 7-14 and Figure 7-15 around 750 seconds is due to the singularity condition that occurs from the slew rate reference signal (slew rate slope changes from positive to negative). Importantly the required control torques to meet the above slew manoeuvres, are well below the maximum torque availability from FEEP thrusters as well as reaction wheels. Therefore the selected actuators (FEEP thrusters and ‘Dynacon-200 Reaction wheel) can be operated below their maximum power limit. However for many practical missions when the slew demands are high, either due to slew angle or time, the control torque demand will increase and hence, the power requirement.

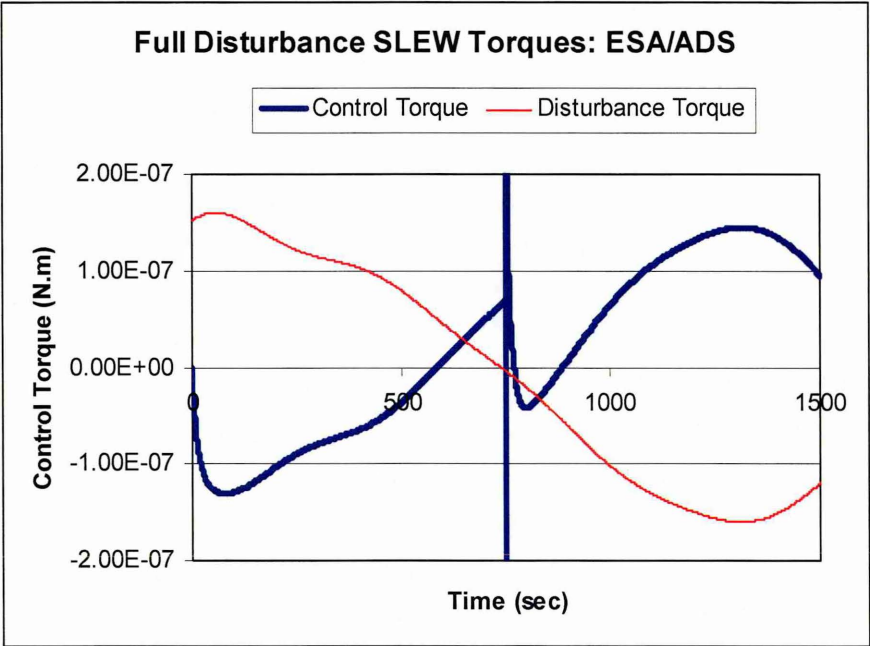


Figure 7-14 Control and disturbance (Aero, magnetic, solar and gravity gradient) torque profile

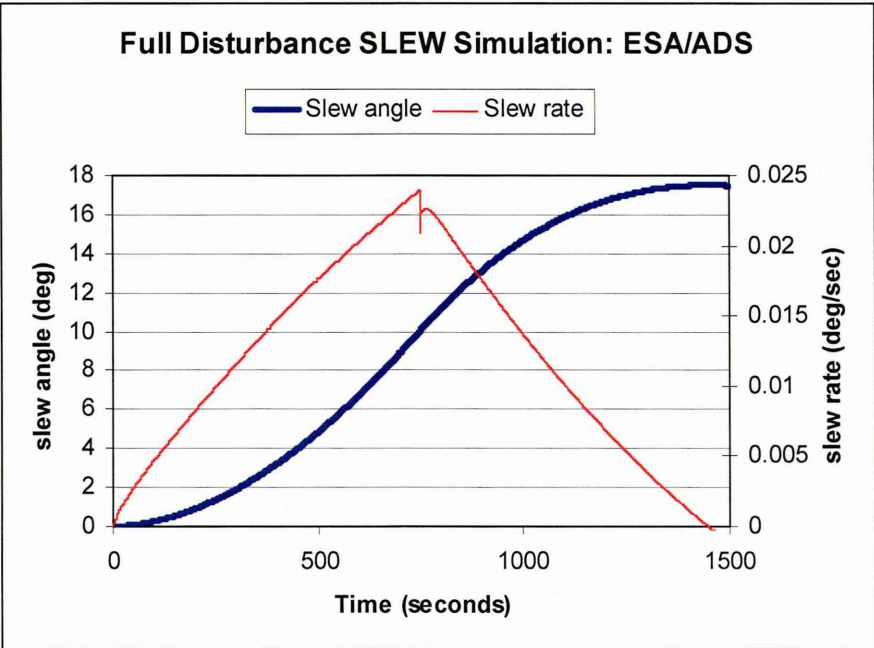


Figure 7-15 Spacecraft attitude dynamics (slew angle and slew rate, $I=0.132 \text{ kg.m}^2$)



Figure 7-16 Minimum slew times in pitch axis ($T_{(max)} = 5 \mu\text{N.m}$; $I = 0.1174 \text{ kg.m}^2$)

The time requirement for a slew manoeuvre depends on the size of the applied torque for a given spacecraft. As shown in Figure 7-16, it is possible to achieve some time saving from the slew manoeuvres taking into account the applied acceleration profile. If the disturbance torques are opposing a slew manoeuvre, the required time for the manoeuvre will be much longer than the predicted values according to Figure 7-16. It may be advantageous to operate the FEEP thrusters in sinusoidal acceleration mode, as this will reduce the stresses on the thruster compared to min-max acceleration mode.

Slew manoeuvre times in practice may vary depending on the payload demand. Therefore it is advantageous to have an attitude control system that can meet a large spectrum of payload demands. For example the FEEP propulsion system can be used to meet both attitude and orbit control demands. For a formation-flying mission, during the separation phase the FEEP demonstrator spacecraft may achieve an initial angular tip off velocity due to separation anomalies. If FEEP thrusters are to be used to stabilise the vehicle during such a transient phenomena, it should have enough torque capacity to stabilise the vehicle within an acceptable error margin and must react promptly without delay in order to lock the tracking system with respect to the ‘Target’ satellite.

Vehicle tip-off stabilisation

A possible FEEP demonstrator mission may design such that it needs to separate from its mother spacecraft prior to its mission (if they are launched in stack configuration). Generally a spacecraft is spin stabilised prior to separation for example from the final booster stage to give the required ‘gyroscopic stiffness’ to the spacecraft. However due to separation anomalies a satellite may achieve a small angular tip-off velocity (in the order of 0.1 to 1 deg/s (Wertz and Larson W. J., 1999) (page 322) with respect to the reference

vehicle. For a formation-flying mission or a stand-alone mission, it is important that the satellite have a built in mechanism to stabilise against these forces. This situation is analysed here considering FEEP thrusters as the stabilising actuators. As this happens at the initial stage of a mission, it is assumed here that FEEP demonstrator spacecraft have about 60% of its maximum torque ($7.5 \mu\text{N.m}$) capability, due to low battery power, initial thruster start up anomalies etc. Analysis was performed using the simplified ‘Euler Moment’ equations (Equation 7-8), with tip-off angular velocity as the initial condition for the one-axis analysis. The maximum excursion angle is considered here as the attitude error that occurs when the S/C tip-off rate is halted using the selected control torque. The required control torque was calculated to keep the tip-off errors within an acceptable limit. However it showed that an initial tip-off rate above 0.1 deg/s is far beyond the set FEEP torque capability. Therefore in Figure 7-17 torque requirement for two different tip-off rates (0.05 deg/s and 0.1 deg/s) are shown with the corresponding maximum error angle (excursion angle). The 0.05 deg/s tip-off rate was selected to show that FEEP thrusters are still capable of correcting small tip-off errors. As shown in Figure 7-17, for high tip off rates the correction torques fall outside the selected FEEP torque capability. Thus tip-off rate higher than 0.05 deg/s for the selected demonstrator s/c falls outside control capability of FEEP thruster’s (mainly due to power constraint as this will be the limiting factor). Therefore with higher tip-off-rates during separation, the FEEP thrusters becomes less effective and the spacecraft final attitude error (horizontal axis in Figure 7-17) becomes significant. Considering that the vehicle tip-off stabilisation arises at the very beginning of the mission and due to the operational behaviour of the FEEP thruster, this type of application may fall far beyond the FEEP thruster’s control capability. One of the main reasons for this is due to the operational uncertainty and delay during the initial start-up process of the FEEP thrusters, as this corrections can be allowed to take an appreciable time.

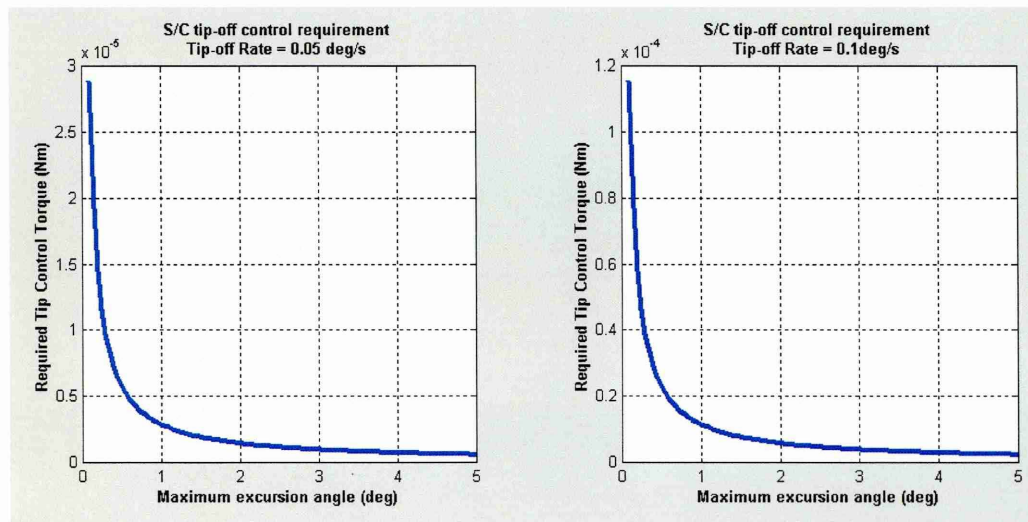


Figure 7-17 Torque requirement (Vertical axis) for a tip-off-rate of 0.05 deg/sec and 0.1 deg/s with the associated maximum excursion angle (horizontal axis)

S/C Spin Stabilisation

For a formation-flying mission it is advantageous to keep the two spacecraft stable during the entire mission: orbit insertion, separation, propagation etc. A non-spinning S/C without any control actuators will quickly become unstable. The simplest way to achieve pointing stability is by spinning S/C about the required axis. During orbit injection, the required spin speed for spin stabilisation varies from about 0.01Hz (Simple orbit stabilisation) to 1.5 Hz (Kick motor firing) (Wertz and Larson W. J., 1999) (Page 327). If FEEP thrusters are to be used at the very instant of initial orbit acquisition mode, it will endanger the mission due to possible operational anomalies or emitter slit blockage. However using FEEP thrusters with reaction wheels or control moment gyros (CMG) will meet this demand as the latter can provide the initial spin stabilisation. Though the FEEP thrusters are less attractive for the initial spin-up process, they can be efficiently applied for spin maintenance and reaction wheel off-loading. For a formation-flying mission, it is advantageous to keep the vehicles velocity vectors pointing along the orbit for tracking and communication link. One way of achieving this is by spinning the spacecraft about its velocity vector. To give an initial spin rate of 1 rpm with the selected reaction wheels requires an angular momentum change by 0.015 N.m.s. For higher spin rates, the angular momentum will increase proportionately.

However FEEP thrusters can be used efficiently to spin-down a given spinning platform. With a control pulse torque of $4.5 \mu\text{Nm}$, it will take approximately 50 minutes to bring the spin rate to zero for a 'Mustang' class satellite with initial spin rate of 1 rpm. Control time will increase proportionately when the effective control torque reduces due to power limitation or due to the presence of disturbance torques. In practice as it is necessary to take into account the disturbance torques, coupling effect between the axes and other deviations from the ideal conditions, the required time would be much longer than the predicted values. In an external disturbance field a spinning S/C may start to 'precess'. Synchronised thruster pulses fired perpendicular to the spin axes are used to 'precess' the spin axis back to its previous position.

Wheel off-loading using FEEP thrusters

Wheels are used to stabilize a spacecraft along its spin axis applying high rotational speeds, similar to the spin stabilization approach. A momentum wheel stabilizes the vehicle along one axis by spinning about that axis. As the wheel spin rate increases, its angular momentum also increases and hence decreases the S/C angular momentum in order to keep the total angular momentum of the system constant. Similarly wheels can be used to counteract disturbance forces acting on a spacecraft by spinning one or more of these wheels. The wheels only change the magnitude of the wheel momentum by spinning faster or slower. If there are no losses (no external torques) the total momentum of the system will be constant according to Equation 7-9. The disturbance forces acting on a spacecraft changes the total momentum of the system, which causes the wheels to spin up and finally to the wheel saturation point (the maximum wheel rotation speed). The momentum saturation condition of a wheel depends on the physical (bearing and

other mechanical components) and electrical (motor) properties of the wheel. Therefore, wheels must be ‘de-saturated’ (off-loaded) periodically. This is performed usually with magnetic torque rods or thrusters to apply a control torque to the s/c. The total angular momentum of the S/C including wheels can be expressed as:

$$\vec{H}_{Tot} = \vec{H}_{S/C} + \vec{H}_W$$

Equation 7-9

Where,

$\vec{H}_{Tot}$: Total angular momentum of the S/C (kg.m²/s)

$\vec{H}_{S/C}$: Angular momentum of the S/C (kg.m²/s)

$\vec{H}_W$: Angular momentum of the wheels (kg.m²/s)

For a wheel/thruster-combined system, wheels can be used effectively to ‘absorb’ the external disturbances. Taking the worst-case scenario, it was assumed here that the spacecraft was subjected to the maximum disturbances simultaneously. The amplitude of the disturbance torque, T_D , is represented as a Sinusoidal input function in Equation 7-10. The amplitude 7×10^{-7} N.m was selected from the previous disturbance torque analysis results section with some safety margin.

$$T_D = 7 \times 10^{-7} \sin\left(\frac{2\pi t}{P}\right)$$

Equation 7-10

In Equation 7-10, t is the time and P , is the orbital period. As this disturbance torque has the accumulated maximum in $\frac{1}{4}$ of an orbit (from minimum to maximum takes $\frac{1}{2}$ orbital period), accumulated angular momentum per orbit (800 km circular orbit with orbital period 6052 seconds), calculated as a root mean square value using Equation 7-11.

$$h = \int T dt$$

Equation 7-11

$$h_w = 0.707 \times 7 \times 10^{-7} \times \frac{P}{4} = 0.707 \times 7 \times 10^{-7} \times \frac{6052}{4} = 7.5 \times 10^{-4} \text{ N.m.s}$$

The wheel momentum storage capacity per day will be:

$$H_w = h_w \times (24 \times 3600 / 6052) \approx 0.01 \text{ N.m.s}$$

This implies a small reaction wheel with angular momentum storage capacity higher than 0.01-N.m.s would be sufficient for a day. For example the ‘TELEDIX RSI-01-5/15’ with angular momentum capacity 0.04 Nms, weights less than 600 grams would meet the required momentum storage demand.

Often attitude control is performed using a feedback loop to send the commands to the actuators. The compensator in the feedback loop controls the wheel speed by controlling the motor current, proportional to the required wheel torque. Achievable accuracies are depending on the feedback control loop electronics, wheel characteristics, used attitude sensors, estimator algorithm etc. This needs detailed knowledge of the different components and continuous knowledge of the attitude through sensors. However for analysis purpose of this study, it was assumed that the selected components have ideal characteristics. A single reaction wheel control was preferred as three-axis control with wheels will certainly increase the system mass, power, cost and complexity. For a FEEP demonstrator mission it can be advantageous to carry a reaction wheel in one of the axes, as this will give a reference value for FEEP test data validation. Besides that the FEEP thrusters can be tested during wheel off-loading if the test has to be carried out within a limited time period. Importantly for a FEEP demonstrator mission a well-stabilised platform is of prime importance in order to validate this ‘ultra’ low thrust actuator. Therefore a reaction wheel/FEEP thrusters combined system would satisfy both attitude and orbit control demands for a FEEP demonstrator mission. Attitude control simulations are performed using ESA/ADS attitude control software tool. ‘Dynacon Micro Wheel 200’ characteristics (see Appendix E) are used for the reaction wheel. As shown in Figure 7-18, wheel angular momentum increases gradually during the operational lifetime of a satellite, if no other actuators are used to stabilise the wheel.

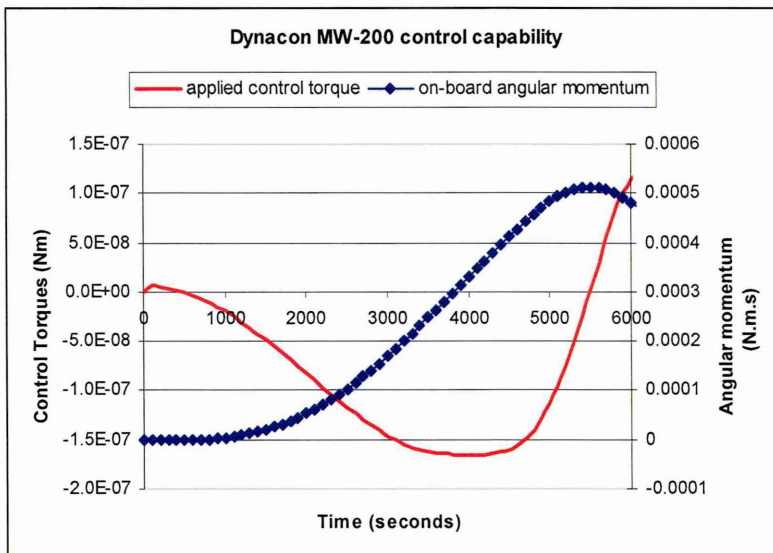


Figure 7-18 Wheel angular momentum develops during S/C pitch control (zero initial conditions)

As shown in Figure 7-19 and Figure 7-20, FEEP thruster performance was analysed for wheel momentum dumping. The simulations were performed considering an initial condition of stored momentum just below the selected wheel saturation point. The analyses were aimed at ‘de-saturating’ the wheel momentum with a certain number of pulses within a limited time period to minimise the disturbance torque effects on the satellite. The control torque magnitudes were selected to be within the range of FEEP

thrusters and wheel maximum torques. The initial conditions and applied simulation constraints are summarised below.

Selected momentum wheel:

- Dynacon Micro-Wheel-200
- Wheel stored momentum: 0.04 Nms
- Wheel angular momentum capacity: 0.05 Nms
- Wheel maximum torque capacity: 30 mN-m

Selected FEED thrusters: FEED-10

- FEED maximum torque capacity: 7.5 μ N-m
- Number of pulses: 50
- Period of the pulse train: 50 sec (25 seconds on and 25 seconds off)
- Mean torque required: 4 μ Nm

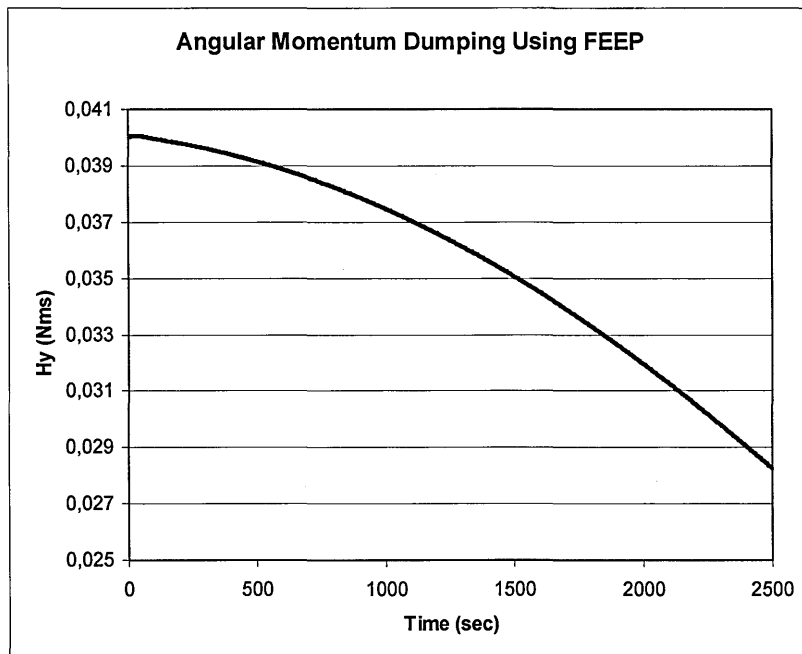


Figure 7-19 Micro reaction wheel angular momentum during wheel ‘off-loading’ using thrusters in a disturbance environment [$H_{y(t=0)} = 0.04$ Nms]

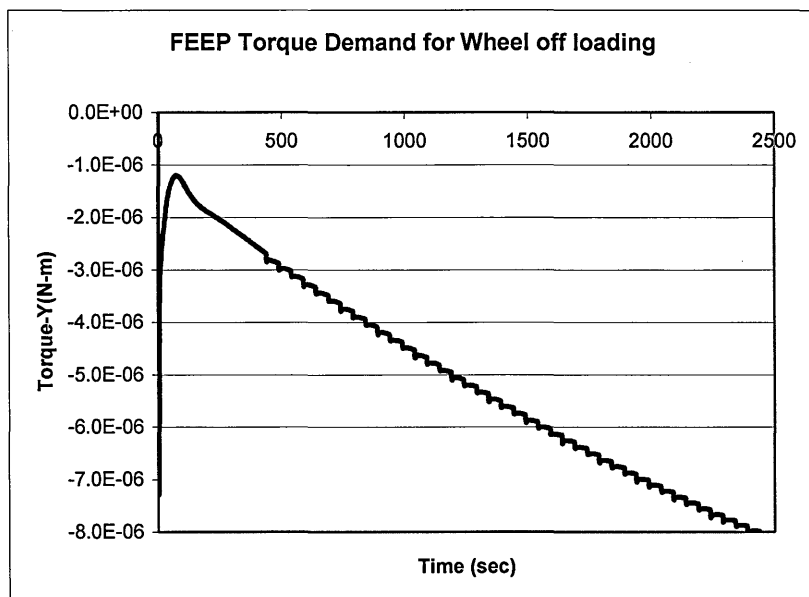


Figure 7-20 Applied FEEP torques (as number of pulses) for wheel off loading while counteracting the disturbances (Pulses are superimposed over the disturbance counteracting torque)

The application of FEEP thrusters for ‘off-loading’ of the wheel was simulated for non-zero roll, pitch and yaw rates. The wheel ‘de-saturation’ and FEEP torque requirements are illustrated in Figure 7-19 and Figure 7-20 respectively. As shown here the angular momentum of the wheel can be brought down to desired operational values successfully with FEEP thrusters. Interestingly the satellite can be kept in a stable attitude with the combined action of wheels and FEEP thrusters. This is confirmed simulating a non-zero initial condition (initial pitch rate of 0.05 deg/sec) during off-loading of the wheel. This shows that it is possible to use FEEP thrusters for accurate attitude control as well as efficient wheel ‘off-loading’ manoeuvres. In Figure 7-20 the thruster pulses can be clearly seen in-bedded on top of the nominal attitude control torques. The ‘off-loading’ pulses were applied at a frequency of 0.02 Hz. Fifty pulses were used with fifty seconds period (25 seconds on and 25 seconds off) to decrease the wheel momentum from 0.04 Nms to 0.028 Nms within 2500 seconds.

7.6 Orbit Control

Spacecraft Separation

It was assumed here that two identical satellites, defined as ‘Target’ and ‘Interceptor’, would be launched in a stacked configuration. The required separation ‘ ΔV ’, which should be small for close proximity formation flying, can be achieved through the separation forces. Although there are number of different separation systems available such as pyrotechnic devices, spring release, clamp bands etc, nano-satellite mission may

require low shock devices. Recently shape memory alloys are being studied as release devices to achieve low shock separation (Peffer, 2000) with shock levels less than 100g. However it should be noted that FEEP thrusters could also be used as a low shock separation actuator and to achieve very small separation ' ΔV '. In this study the conventional separation devices are categorised as 'impulse separation' devices and possible separation using FEEP thrusters are studied as a special case. Importantly whatever the applied spacecraft separation system, it must meet the following conditions:

- The shock limits should be below the material limits of the separation interface and the detachment system
- Separation system must withstand the launch loads
- The oscillation frequencies of the structure, payload, instruments etc., should be within the allowable limits.
- The magnitude of the applied ' ΔV ' should be adequate enough for a collision free separation as well as to overcome any initial friction forces
- Avoid possible tip-off during separation (must select stable separation positions)
- Non-explosive (to achieve low cost piggyback launch opportunities)

From the engineering point of view the two spacecrafts ('Interceptor' and 'Target') can be stacked for launching as one wished. However from the practical point of view, it is important to avoid possible obstruction to the 'Target' orbit propagation. Other important factors that must take into account are the launcher payload bay volume for possible piggyback, magnitude of loads that transfer from the launcher to the spacecrafts, stability, separation method etc. Further for a low cost mission the two s/c should be within 'field-of-view', to enable simple communication and tracking systems. Orbit propagation simulations were initially performed with 'Interceptor' attached behind the 'Target'. Vehicle separation is performed with a small negative ' ΔV ' applied to the 'Interceptor' s/c with respect to the velocity vector. Depending on the magnitude of applied ' ΔV ' and orbital position, the 'Interceptor' spacecraft orbit may alter accordingly (a negative ΔV decreases the semi-major axis and period). Importantly the separation ' ΔV ' should be reasonably high enough to achieve a collision free propagation while not exceeding beyond the correction capability of FEEP thrusters. Clearly the magnitude of ' ΔV ' depends on the separation mechanism employed, as well as the structural constraints of the spacecraft.

As one can expect due to separation ΔV , the two spacecraft's will deviate from their reference orbits. Application of a negative ' ΔV ' will initially slow down the 'Interceptor' s/c resulting a shorter orbital period compared to the 'Target' vehicle ('Interceptor' will lead the 'Target'). As a result the actual orbital velocity of the 'Interceptor' will increase, moving it to a low altitude orbit. However for a formation-flying mission, it is important to maintain their relative distances according to the requirements of the mission. Therefore a negative ' ΔV ' will result a decrease in 'Interceptor' spacecraft perigee altitude. Hence, the 'Interceptor' S/C will propagate into a lower orbit with respect to the 'Target'. This situation is illustrated in Figure 7-21, for two identical satellites ('Mustang' configuration). Here the selected coordinate system origin was set at the 'Interceptor' centre of mass. The relative tangential distance is defined as the relative distance along

the velocity vector measuring from 'Interceptor' to 'Target' satellite. The normal distance is the relative distance between the two satellites perpendicular to the velocity vector (see Figure 7-22).

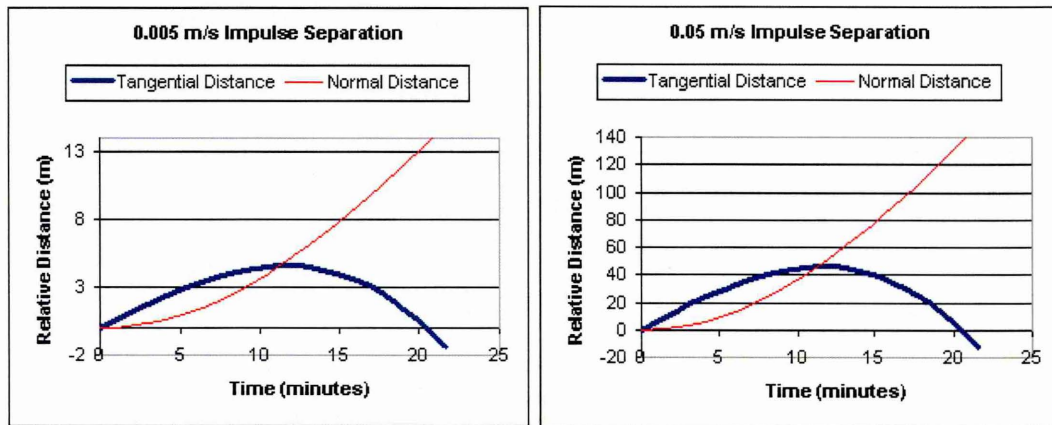


Figure 7-21 'Interceptor' satellite relative distance after a negative ' ΔV ' separation. The coordinate system origin is at 'Interceptor'. (An equal and opposite ΔV is applied to the 'Target')

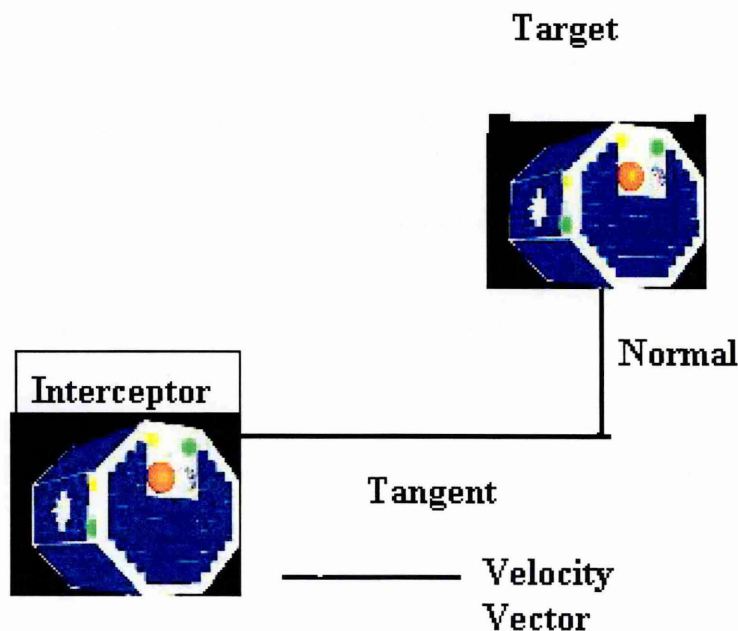


Figure 7-22 Tangent and Normal distance measurement from Interceptor to Target

For simulation purposes, the initial separation distance between the two spacecrafts were taken as zero. However in practice this is not true, as there is a certain distance between the mass centres of the two satellites. As shown in Figure 7-21 for high ' ΔV ' values, the relative tangential and normal distances increases rapidly. However for very low ' ΔV ' values, it may reduce the required safety margins for a collision free propagation. Therefore for the considered orbit (800 km) and the satellites attached configuration, the lowest possible ' ΔV ' for separation is considered as 0.005 m/s. Beside this, it is important to consider the limitations of the employed target tracking system, such as the field of view and operational distance of the s/c instruments. This is especially true for a close proximity formation-flying mission, such as this. As shown in Figure 7-21, within a short period of time, in this case around 20 minutes, the 'Interceptor' starts to propagate in front of the 'Target'. Although at this stage there is a substantial normal distance between the two satellites there will be a severe penalty from tracking point of view. Therefore if 'Interceptor' propagation in front of 'Target' is unacceptable, there exists only a very short time period to apply correction manoeuvres when the Interceptor is stacked behind the Target (negative ΔV to Interceptor and positive ΔV to target). Although the above simulation results assumes that it is possible to select a relative separation velocity, this may be difficult to achieve with conventional separation devices. For example the EUROCKOT launcher with its separation adapter (937 or 591) provides a relative payload separation velocity in a range of 0.1 to 0.8 m/sec using three springs (Stamerjohanns, 1998). This shows that a ΔV of 0.05 m/sec (relative velocity of 0.1 m/sec) is readily achievable using conventional separation techniques. A low separation velocity can be achieved using dampers or the state of the art shape memory alloy separation techniques (Sarafin, 1995). Nano-satellite compatible separation devices are given in Appendix-H.

Formation-flight

FEEP thrusters can be operated either continuously or with varying pulse length and magnitude. This attractive feature, which leads to proportional control do not readily available with other conventional types of reaction jets. The proportional control capability of FEEP thrusters is best demonstrated by simulating a formation-flying scenario. Two 'nano' class satellites are used to model a formation-flying mission. The physical properties of these reference satellites are selected as given in Table 7-3. To keep in line with the interest of the on going projects such as the 'Mustang' project from Cranfield/Southampton Universities (Roberts et al. 2002), a feasibility study was performed for a close proximity formation-flying mission (approximately 10-100 meter range). In order to understand the demands on the propulsion system, the analysis is performed to understand the following situations.

- Two identical satellites are flying in formation, provided that the two satellites have already achieved its reference orbit and separation distance.
- When the two satellites are injected to the orbit a single entity. This requires first to separate and then to achieve the required constellation.

- The differential drag effect on a formation-flying mission when the two satellites are not identical.

The ‘Mustang’ model as shown in Figure 7-3, was used as the base line model for the two satellites. An 800-km circular polar orbit was selected as the reference orbit. The first case was analysed when the two satellites are propagating approximately 10 meters apart in a reference orbit. The relative ‘In-Track’ distance between the ‘Interceptor’ and the ‘Target’ S/C was analysed with STK simulation tool. Analyses were performed with two types of propagators, J-4 and MSGP-4. The corresponding simulation results are shown in Figure 7-23. As seen here the relative distance variation is very little even when all the significant perturbation effects are considered (MSGP4).

In order to keep the two satellites 10 meters apart (in-track) in a 800km circular orbit, the ‘Target’ satellite true anomaly was set to 0.00008 degrees compared to the ‘Interceptor’ which was kept at 0 degrees. The relative distance was measured from the ‘Interceptor’ satellite to the ‘Target’. The propagation was simulated for a polar orbit (90° inclination), with ‘Target’ in front of the ‘Interceptor’. It was assumed here that the ‘Target’ vehicle propagation in front of the ‘Interceptor’ is advantageous for the FEEP demonstrator mission. When this free propagation scenario was simulated for different orbit inclinations, the relative distance variation (initially 10 meters) was within a satisfactory error margin in line with the Figure 7-23 results. Therefore as shown in Figure 7-23, once the constellation has been established, the perturbation effects on relative distance between the two satellites have no long-term (periods higher than one orbital period) effect on the orbital parameters. However this statement will not hold if the two spacecrafts physical design varies significantly, which will give rise to differential drag.

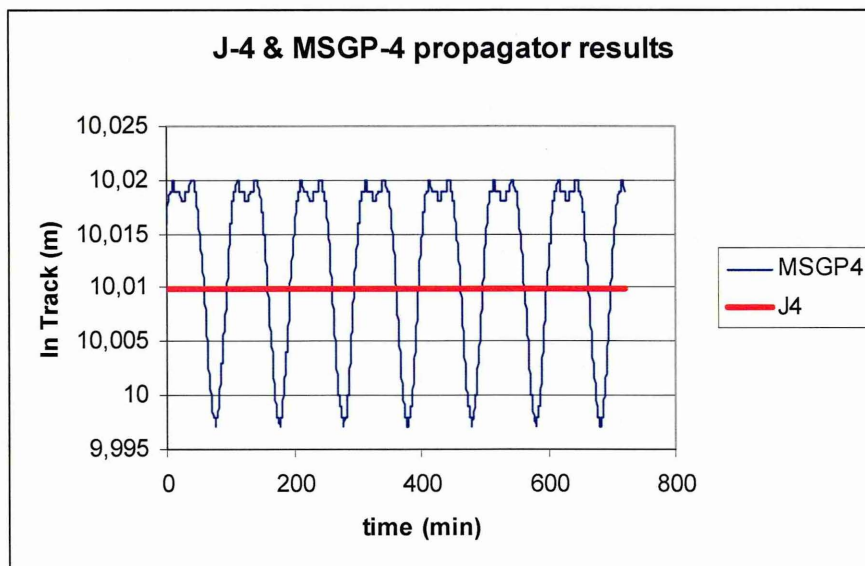


Figure 7-23 Interceptor to Target ‘InTrack’ (along the orbit) distance variation

The Earth isn't really spherical; it has more mass around the equator. This Earth 'oblateness' effect causes the right ascension of the ascending node, Ω , and the argument of perigee, ω , to change in a predictable way. Therefore oblate Earth has no effect on the relative orbital parameters between two S/C in a given orbit (especially for close proximity formation-flying). This is further explained with the propagator results shown in Figure 7-23, with up to J4 term considered (mathematical formula used to describe the oblate Earth). However according to MSGP4 propagator simulation analysis the perturbing effects, such as atmospheric drag, solar wind and third body, change the relative orbital parameters. It should be noted here that these changes are very small when the two satellites are identical. The situation for non-identical spacecrafts were analysed by varying the 'BSTAR' (B^*) parameter in the MSGP4 propagator. The B^* , an adjusted value of ballistic coefficient in SGP-4 type propagator, is defined according to Equation 7-12 as a function of satellite parameters and the media surrounding it. Here ρ_0 is the atmospheric density at the perigee point. For a 'Mustang' type satellite at 800 km circular orbit, B^* is in the order of 10^{-16} m^{-1} . However at the considered altitude even when doubling the B^* for 'Target' satellite with respect to 'Interceptor' do not affect the constellation adversely (Low the B^* less susceptible the object for drag). However if the selected orbit has a low perigee passage then the relative distance variations are considerable. The situation may further deteriorate for an unlikely formation-flying scenario, such as if the B^* ratio between the two spacecrafts is greater than a factor of 1000. The corresponding differential drag effect is shown in Figure 7-24, for a 'Mustang' type satellite following a 'Target' s/c that have a B^* , 1000 times that of 'Interceptor' and having a perigee altitude about 100 km (Perigee for air density determination in Equation 7-12).

$$B^* = \frac{1}{2} \frac{C_D A}{m} \rho_0$$

Equation 7-12

Where,

- B^* : Drag like parameter in MSGP-4 propagator
- C_D : Drag coefficient,
- A : Exposed frontal cross sectional area
- m : The total mass of the satellite.

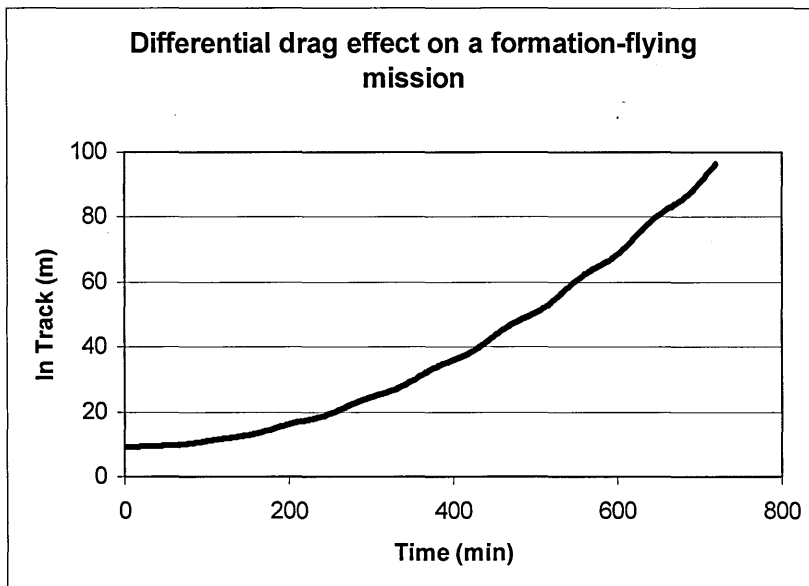


Figure 7-24 'Interceptor' to 'Target' in-track distance variation for two highly non-identical satellites (this situation is highly unrealistic for a nano-satellite formation flying mission)

According to Figure 7-24, as the relative 'Target' position changes continuously and in an exponential way, it will add more complexity to keep the constellation intact. However, this will not be a concern for almost all planned missions as the selected orbits are much higher than considered here. An important condition for a possible FEEP demonstrator mission will be, first to achieve its constellation positions, and then propagate while maintaining these positions. Such a situation arises when the two s/c are injected into an orbit in a stack configuration. This requires the two spacecrafts to first separate in orbit, prior to its formation-fly. As discussed earlier application of a ' ΔV ' for separation will automatically put the two s/c in different orbits. Therefore it is of interest to analyse relative motion between two close orbiting satellites. The relative motion analysis, the propagation of one satellite with respect to the other, is better explained with the 'Clohessy-Wiltshire' equations, also known as Hill's equations. These equations are derived with respect to a satellite-based coordinate system; also known as *RSW* coordinate system. A more detail analysis of this subject is given in (Sidi, 1997).

As shown in Figure 7-25 the '*RSW*' system moves with the satellite. The '*R*' axis points to the satellite from the Earth centre, the '*W*' axis is normal to the orbit plane, and the '*S*' axis is in orbit plane and normal to the position vector, *R*. Importantly the '*S*' axis will continuously align with the velocity vector only for circular orbits.

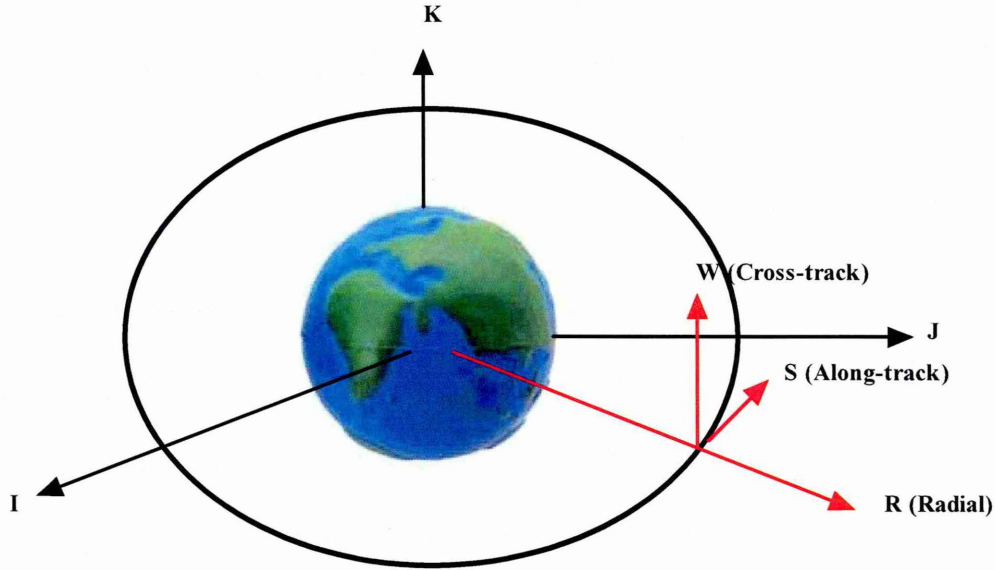


Figure 7-25 RSW satellite based coordinate system

The Hill's equations make some critical assumptions that limit its use: It assumes that the orbit of the target (Target) spacecraft is circular or nearly so, and that the distances between the target and interceptor (Interceptor) spacecraft are small, on the order of a km separation or less. This limits the analysis only to circular orbits. The equations also assume that the separation ΔV , applied as an impulsive force, affect only the Interceptor spacecraft. Therefore the relative position analysed using the Hill's equation in its original form basically gives relative position with respect to an unperturbed reference orbit. However in reality any separation force applied to the Interceptor will give an equal and opposite force to the Target as well. The original Hill's equation further limits practical application by assuming Earth as a spherically symmetric body. This will seriously limit the Earth bulge or the J_2 term effect on relative position. For similar ballistic coefficient satellites, J_2 effect on relative position will be substantial at least when taken into account short-term orbit variations. If the relative position of the interceptor in the *RSW* coordinate system is x , y and z respectively, then the Hill's equations for near circular orbits is expressed according to Equation 7-13 (Sidi, 1997) (page 347).

$$\ddot{x} - 2n\dot{y} - 3n^2x = f_x$$

$$\ddot{y} + 2n\dot{x} = f_y$$

$$\ddot{z} + n^2z = f_z$$

Equation 7-13

Here ' n ' is the mean motion of the target satellite for the selected orbit and ' f ' represents the external force field. If assumed, that no external forces are imparted by the interceptor satellite, then:

$$f_x = f_y = f_z = 0$$

For zero external force field, Equation 7-13 can be solved for relative position and velocity of the interceptor satellite with respect to its reference orbit. If the interceptor initial position and velocity with respect to target satellite is known such as:

$$x_0, y_0, z_0, \dot{x}_0, \dot{y}_0, \dot{z}_0$$

Then the Equation 7-13 can be solved for the interceptor position and velocity using 'Laplace transforms' theory. This results the interceptor relative position at a given time as shown in Equation 7-14. For details see page 351, (Sidi, 1997).

$$\begin{aligned} x(t) &= \frac{\dot{x}_0}{n} \sin(nt) - \left(3x_0 + \frac{2\dot{y}_0}{n} \right) \cos(nt) + \left(4x_0 + \frac{2\dot{y}_0}{n} \right) \\ y(t) &= \left(6x_0 + \frac{4\dot{y}_0}{n} \right) \sin(nt) + \frac{2\dot{x}_0}{n} \cos(nt) - \left(6nx_0 + 3\dot{y}_0 \right) t + \left(y_0 - \frac{2\dot{x}_0}{n} \right) \\ z(t) &= z_0 \cos(nt) + \frac{\dot{z}_0}{n} \sin(nt) \end{aligned}$$

Equation 7-14

Equation 7-14 provides some important information for relative motion analysis. These equations are valid only when external force field is zero, hence $F_{\text{thrust}} = 0$. The STK results shown in Figure 7-23 can be verified by inspecting the second equation in Equation 7-14. These results were derived for a situation where the formation has already established. Hence the following initial conditions exists:

Initial position: $x_0 = 0 \text{ m}; \quad y_0 = 10.1 \text{ m (along track)}$

Initial separation velocity: $\dot{x}_0 = \dot{y}_0 = 0 \quad \text{m/s}$

Substituting the above initial conditions in Equation 7-14 will result,

$$y(t) = y_0 = 10.1 \quad \text{m}$$

Hence once the formation has established, the propagation will continue maintaining the initial relative distance. This is in line with the Figure 7-23, J4 propagator results. Even the presence of an external force field will not adversely affect the relative distance, for identical satellites. This can be seen clearly as shown in Figure 7-23 with MSGP4 propagator results. Here the motion is somewhat oscillatory with the relative along track distance varying about 1 to 2 cm from the initial 10 m separation. However for non-identical satellites, differential drag effects must take into account.

After a separation $\Delta V (= 0.0002 \text{ m/s})$, the resultant propagation is shown with the STK 'full force' propagator model and Hill's equation as shown in Figure 7-26 and Figure 7-27 respectively. Both models predict quite well how the interceptor propagates initially in front and finally behind the target satellites. However the magnitude of the relative distance with the Hill's equation is at least a factor 3 smaller than what is predict using the STK full force model. This is because with the Hill's equation, the separation ΔV affect only for the Interceptor satellite. As with STK, the ΔV separation applies to both satellites (equal and opposite), and the resultant effect is twice as big as what Hill's equation predict. Beside that STK full force model takes into account all major perturbations affecting the two satellites, which contributes further increase in relative distance. However, Hill's equation can be used to understand and predict the relative motion, at lest during the first few minutes after separation.

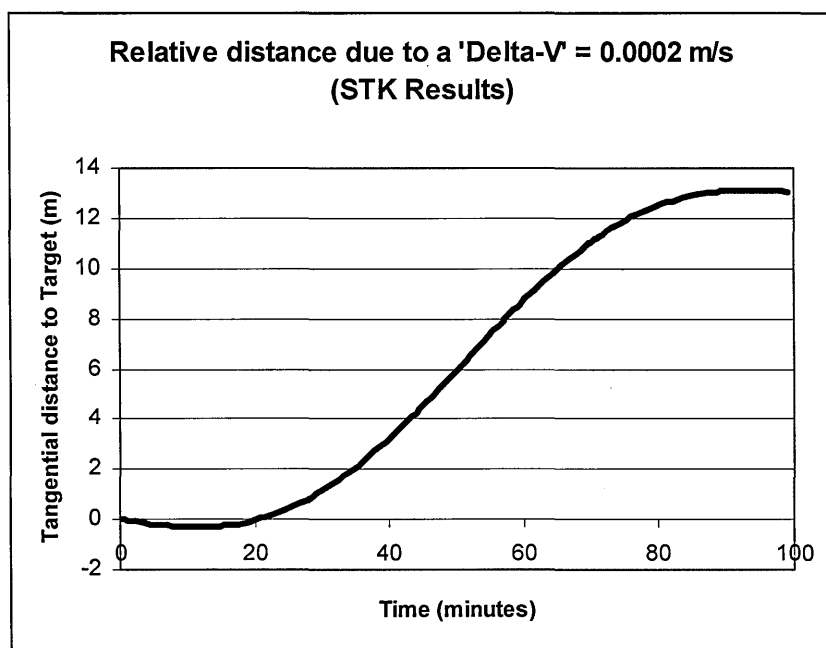


Figure 7-26 Relative tangential (along track) distance due to a separation $\Delta V = 0.0002 \text{ m/s}$ when Interceptor is stacked in front of the Target satellite (STK simulation results)

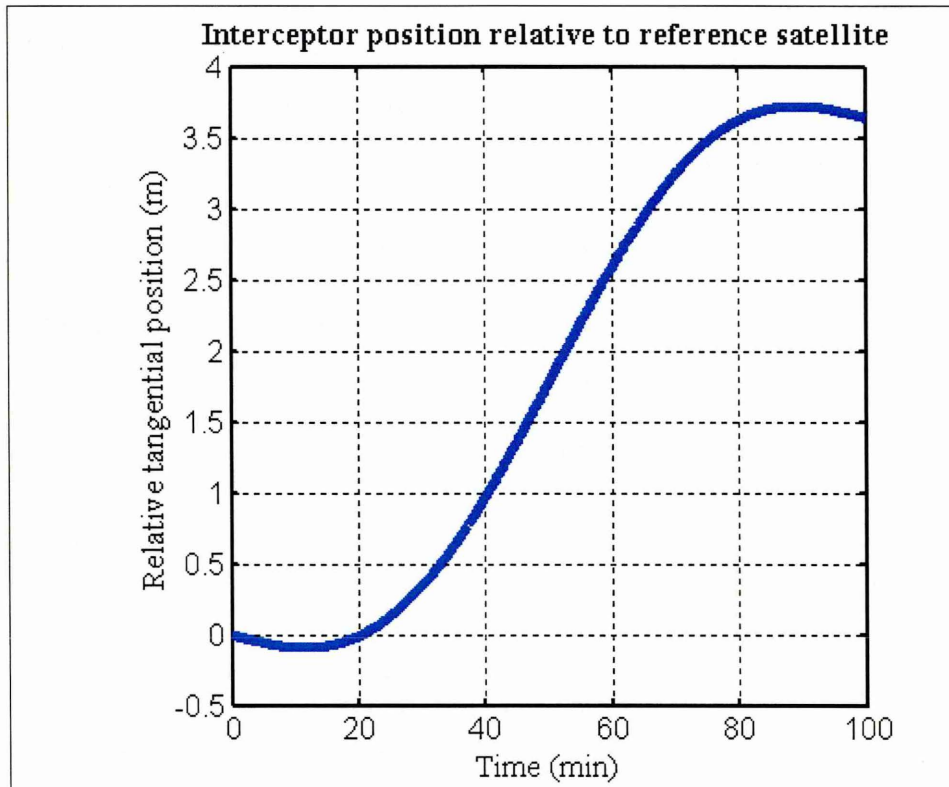


Figure 7-27 Relative tangential distance to a reference target satellite due to a separation $\Delta V = 0.002$ m/s (Hill's equation)

Although the interceptor doesn't actually orbit the reference target satellite, the instantaneous parameters often result in an elliptical-like motion. Importantly Equation 7-14 is helpful to determine, what happens during the separation, where the applied ' ΔV ' determines the interceptor initial velocity conditions. According to the second equation in Equation 7-14, for any non-zero velocity component in y direction, the interceptor will drift away from the target (Target orbit in this case is the Interceptor un-perturbed reference orbit). According to the Equation 7-14 first equation, any nonzero velocity component along the x direction (radial direction) will produce an osculating motion.

The drift term in along orbit track motion, $y(t)$, is only a function of initial ' ΔV ' and relative radial position, x_0 , of the 'Interceptor' (as y_0 is not time varying) . Therefore by carefully selecting these two initial conditions (x_0 and ΔV), the 'Interceptor' drift can be controlled. However, the amount of initial radial displacement is constrained according to the size and stacking configuration of the two satellites.

As shown in Figure 7-28 pure oscillatory motion of the interceptor arises when the ' ΔV ' is totally along the radial axis. A ' ΔV ' along the radial axis won't change the orbital period, but instead will change the eccentricity. The satellite will drift apart from its reference orbit for a half a period, then back and so on. Because the period is the same, the satellite will rendezvous with its reference orbit every complete period.

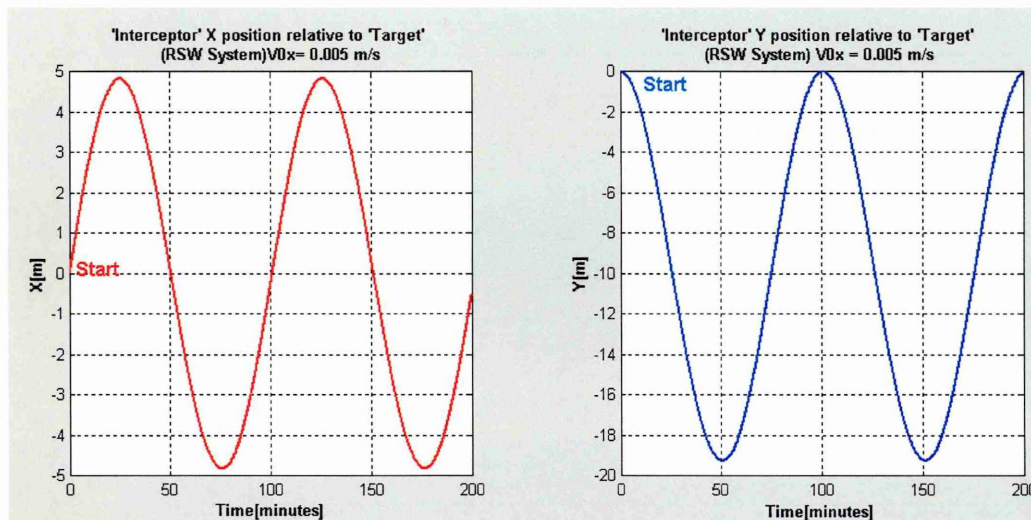


Figure 7-28 'Interceptor' position with respect to 'Target' for initial $\Delta V = 0.005$ m/s along radial axis

Any ' ΔV ' separation along the S axis (along track) as shown in Figure 7-29 will cause the y position of the 'Interceptor' to drift with time. For a negative ' ΔV ', the 'Interceptor' will propagate for a short period behind the 'Target' and then will move in front of it (this is true if the 'Target' is an un-distort reference satellite or that propagates with a positive ' ΔV '). This is self explanatory when examined the x position of the 'Interceptor', which oscillates around negative values in this case as shown in Figure 7-29.

A negative ' ΔV ' will move the 'Interceptor' into a lower orbit with respect to the 'Target' (having a positive ' ΔV '). Therefore for a negative ' ΔV ', 'Interceptor' semi-major axis will be reduced and hence the orbital period. Interestingly a positive ' ΔV ' separation for the 'Interceptor' will result the opposite: increase orbital period and hence move behind the 'Target'. A change in ' ΔV ' along the S axis will change both the period and the eccentricity. Although the 'Interceptor' will rendezvous with the un-distort 'Target' satellite altitude after one period, the along track distance between the two satellites will successively increase.

Perhaps the most important outcome from the above discussion is that by carefully selecting the relative position and velocity of the 'Interceptor', it is possible to control the drift of it with respect to the 'Target'. However for a formation-flying mission, which requires initial separation, the position of the Interceptor is confined to the size and configuration of the 'Target' satellite.

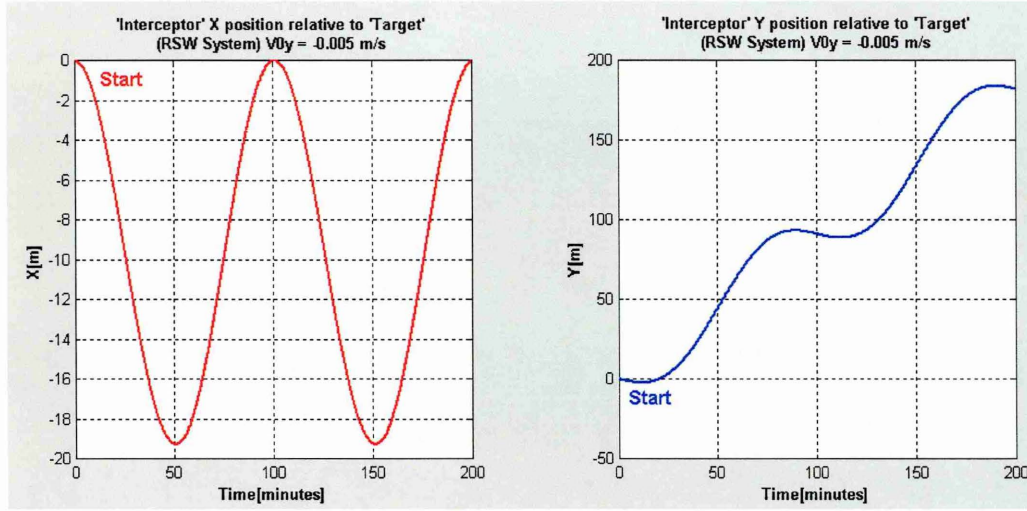


Figure 7-29 Interceptor position with respect to Target for initial $\Delta V = -0.005\text{m/s}$ along S axis

According to Hill's equations, the Interceptor propagates in an elliptical-like pattern around its initial reference orbit. The shape and location of this ellipse is of interest as it gives an idea how the interceptor propagates with respect to a possible Target satellite. This information can be used to optimise the control manoeuvres for formation flying in the *STK* simulation tool. Importantly because of the drift terms, the ellipse is instantaneous, and its position will change with time. Equation 7-15 represent an ellipse in the x - y plane.

$$\frac{x^2}{C^2} + \frac{y^2}{4C^2} = 1$$

Equation 7-15

Where,

C = magnitude of the variation in x (radial position) or semi-minor axis

$2C$ = the instantaneous magnitude of the y variation or the semi-major axis

By examining the Hill's equations C is defined as (Vallado, 1997):

$$C = \sqrt{\left[\left(3x_0 + \frac{2\dot{y}_0}{n} \right)^2 + \left(\frac{\dot{x}_0}{n} \right)^2 \right]}$$

Equation 7-16

The centre position of the ellipse is given by the initial displacement terms in the Equation 7-14. Hence the initial displacement is given by:

$$x_c = 4x_0 + \frac{2y_0}{n} \quad y_{c_0} = y_0 - \frac{2x_0}{n}$$

Equation 7-17

Solving Equation 7-17 with the drift term in the y direction, which is given by:

$\dot{y}_c = -6nx_0 - 3\dot{y}_0$, Gives:

$$x_c = -\frac{2}{3} \frac{\dot{y}_c}{n}$$

Equation 7-18

Let the interceptor centre of mass x and y coordinates (RSW co-ordinate) are 0.15 m and -0.3 m respectively. This will represent the FEEP demonstrator mission 'Interceptor' and 'Target' satellites, where the 'Interceptor' attached point is shifted relative to 'Target' centre of mass. (A more detail discussion of this special situation available later). Consider the initial separation ' ΔV ' has the velocity components: $V_{0x} = 0.01\text{m/s}$ and $V_{0y} = -0.0002\text{ m/s}$. As mentioned earlier and as shown in Figure 7-30, the interceptor relative motion has an elliptical-like pattern. Equation 7-17 gives the instantaneous centre of this ellipse as $X_c = 0.2\text{m}$ and $Y_{c0} = -20.3\text{ m}$. Therefore by carefully selecting the initial position and separation ' ΔV ' of the interceptor, it is possible to move the interceptor 'backward' to a safe distance prior to a formation fly.

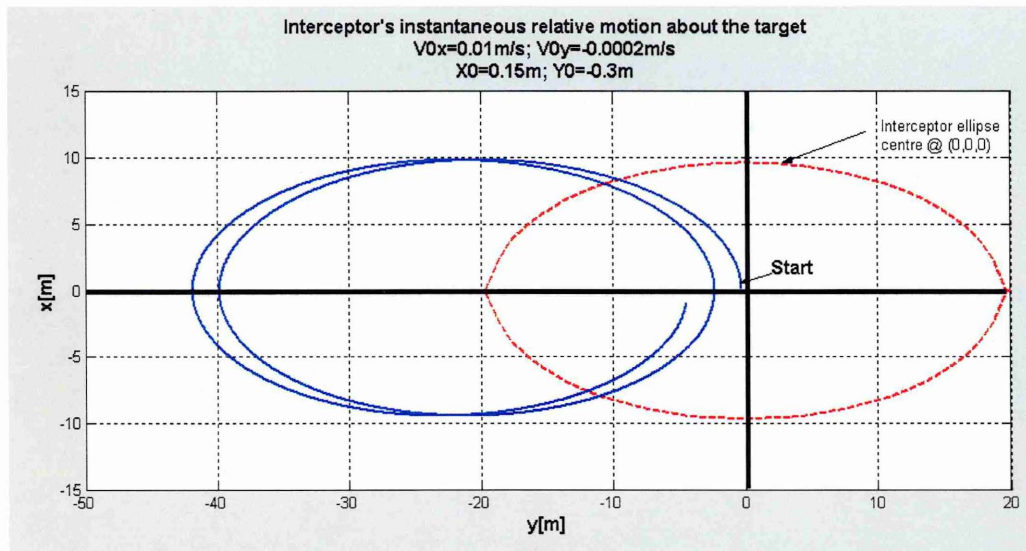


Figure 7-30 Interceptor's relative motion about the target satellite for non-zero initial conditions.

An important factor is the size of the interceptor ellipse is determined by Equation 7-16 ($2C$: semi-major axis). Therefore C may become zero for certain combination of initial conditions. According to Equation 7-16, C will be zero if:

$$\dot{x}_0 = \dot{y}_0 = x_0 = 0 \quad \text{or} \quad \dot{x}_0 = 0; \quad \dot{y}_0 = -\frac{3}{2}x_0 n \quad \text{or} \quad \dot{x}_0 = 0; \quad x_0 = -\frac{2\dot{y}_0}{3n}$$

Equation 7-19

If initial conditions are met as in Equation 7-19, the interceptor will follow a straight-line motion: without oscillations. However, the above results were derived neglecting the effects of external forces and equal and opposite ' ΔV ' achieved by the Target satellite. The effect of the external forces were studied with the help of *STK* 'Astrogator' simulation tool with the full force model propagator. Simulations were performed with and without external force field with '*STK*' and '*Matlab / Simulink*' simulation tools respectively. The same initial conditions as in Figure 7-30 were selected in both cases. The results of the two cases are summarised in Figure 7-31, which uses Matlab /Simulink, and Figure 7-32, which uses STK, for in track and radial distance relative to the target. It shows that due to the negative ' ΔV ' along the velocity vector, the interceptor propagates into a lower orbit, causing a change in both semi-major axis and period.

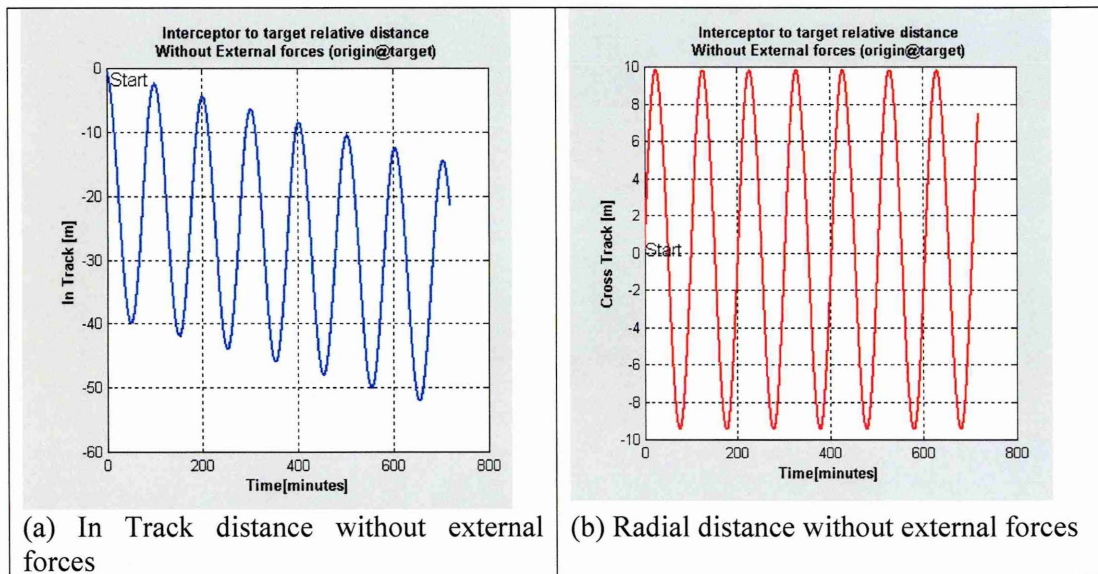


Figure 7-31 Relative distance variation due to separation ΔV , when the external force field is negligible (Matlab simulation results)

The Hill's equations predict, that by carefully selecting the initial position of the interceptor (above or below the target vehicle, along the radial vector), it is possible to 'back' (propagate behind the target) the interceptor with a negative ' ΔV '. This is in line with the Figure 7-31 relative distance variation. When there is an external force field present (orbital perturbations), as shown in Figure 7-32, the situation is somewhat different. During the first few minutes after separation, the relative distance variation is similar for both with and without external force field. As shown in Figure 7-32, due to the external forces (perturbations) the interceptor spacecraft propagates into a lower orbit, which in turn gives a higher orbital velocity. Due to the relative higher orbital velocity of the interceptor (shorter orbital period compared to the target vehicle), it tends to move in

front of the target (un-distort orbit) at some point of the propagation. However, the external force field has no significant effect along the radial axis for a close proximity formation-flying mission. This is in line with the Equation 7-14, which has no drift term along radial axis.

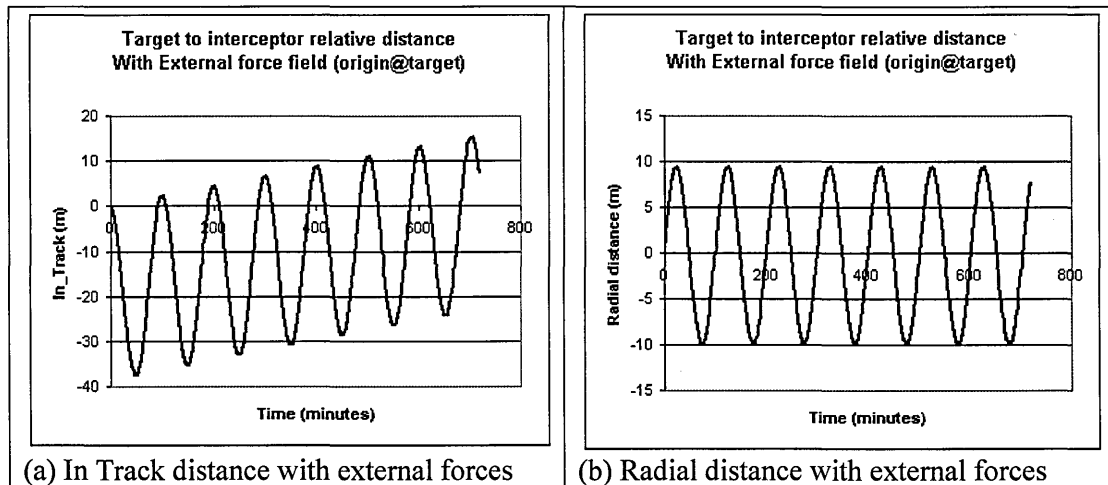


Figure 7-32 Relative distances between the Interceptor and its un-distort Target orbit with external forces (STK simulation results)

As for a FEEP demonstrator mission the interest is to propagate behind the 'Target' satellite, a negative ' ΔV ' (opposite to the orbital velocity vector) relative to the 'Target', will cause the 'Interceptor' to propagate in front of the 'Target' sooner or later. As discussed earlier a positive ' ΔV ' relative to the 'Target' and with carefully selected initial conditions would bring the 'Interceptor' behind the 'Target' as required. The conventional stacking procedure is to attach the two spacecrafts in series. Therefore in order to separate with a positive ' ΔV ' for the Interceptor, it should stack in front of the Target as shown in Figure 7-33. The required separation ' ΔV ' can only be achieved through impulsive separation mechanism (eg. spring load), as FEEP operation will contaminate the 'Target' spacecraft.

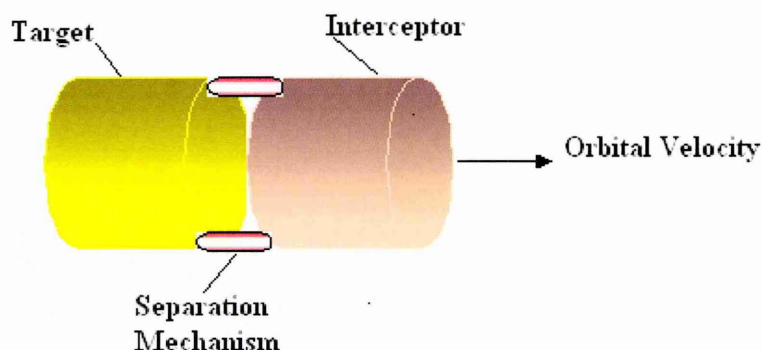


Figure 7-33 Interceptor attached in front of the Target for positive ' ΔV ' separation

As shown below the Interceptor will gradually move behind the Target for a positive ΔV separation. This is actually the opposite of what had happened during the negative ΔV separation as discussed earlier. Further the stacking architecture shown in Figure 7-33 allows the Interceptor frontal area free to mount tracking and communication devices. FEEP thrusters can be used to maintain the relative distance in this case. However in order to break the increase in relative distance and to maintain the Interceptor within close proximity of Target spacecraft requires a negative ΔV application. This situation will increase the risk of Target spacecraft contamination. The FEEP ion flux (mostly neutral) is moving about 40,000 m/sec (I_{sp} about 4000 seconds) and hits the Target spacecraft within a fraction of a second (Target few hundred meters away from the Interceptor).

The situation when the spacecrafts are stacked one above the other as shown in Figure 7-34 is rather an uncommon configuration. For example during launch spacecrafts are exposed to high shock loads and vibrations along the velocity vector. Therefore series stacking provides an advantage over lateral stacking. Further during separation the spacecrafts may get a tip off velocity due to displacements in the common separation interface.

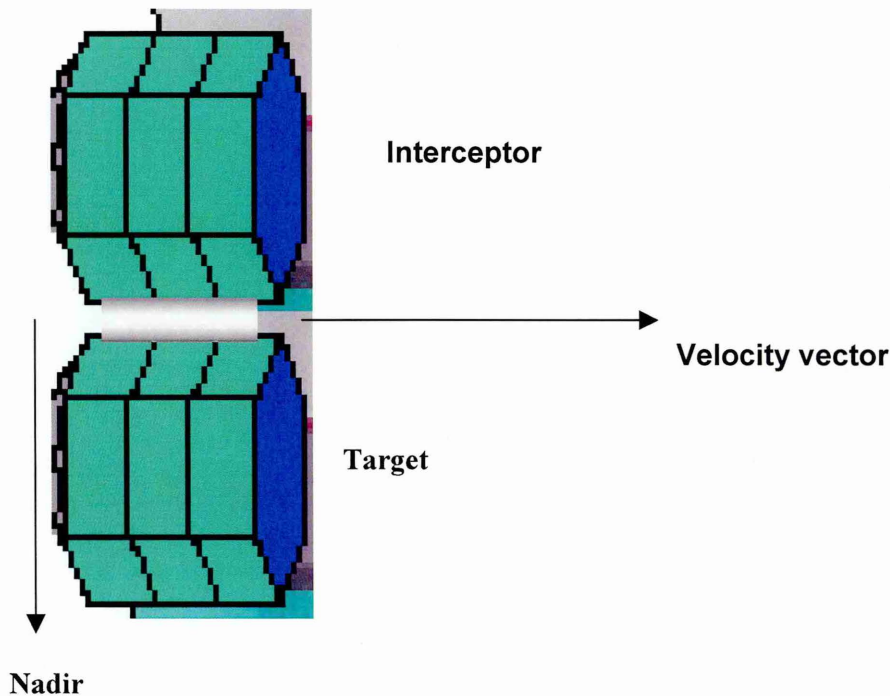


Figure 7-34 Two spacecrafts stacked one above the other

However if we consider a situation where the spacecrafts initial separation takes place perpendicular to the velocity vector, along the velocity nadir plane, and then the final separation ΔV is provided by FEEP thrusters, the situation is somewhat similar to the earlier discussed Hill's problem. Low shock separation devices such as clip bands can be used to achieve the lateral separation, as the objective here is to disintegrate the two spacecrafts from its common attachment interface. As the initial FEEP thrusting will take place along the velocity vector and the Target spacecraft propagates from start in front of the Interceptor, there will be no plume contamination problems. The simulated result with STK in a full force field is given below. It had assumed here a lateral impulsive separation force giving a ΔV of 0.0002 m/s (a relative velocity change of 0.0004 m/sec), prior to continuous FEEP operation. Finally allowing a thrust free propagation of 478 minutes brings the spacecraft to a relative distance of 60 meters. Here FEEP thrusters had operated about 23.6 minutes providing a total velocity change of 0.00005 m/sec. The simulations had performed with a thrust magnitude of 3.5 μN and an I_{sp} value of 3000 seconds. Compared to the previously discussed impulsive separation along the velocity vector (stack in series), this continuous low thrust separation very little oscillation in the orbit normal plane. This is advantageous from the tracking point of view.

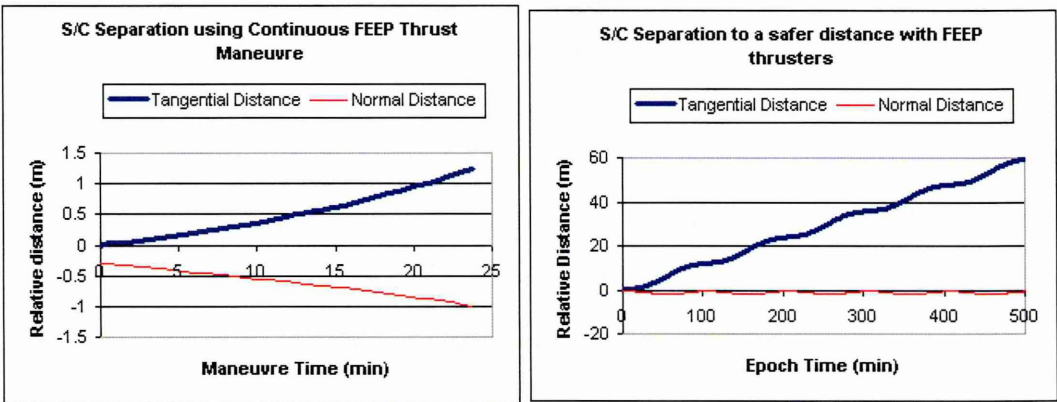


Figure 7-35 FEEP performance results in a full force field using STK simulator (Interceptor on top of the target. Distance is measured such that origin is at Interceptor)

Once the ‘Interceptor’ satellite is ‘backed’ to a safe distance, FEEP thrusters can be used to maintain the acquired relative distance. This requires a negative ΔV manoeuvre to bring the Interceptor spacecraft to the Target orbit (Target orbit has a higher orbital period compared to Interceptor orbit). Once the spacecraft had reached to its final safe distance, FEEP thrusters were operated along the anti velocity vector to maintain this relative distance. The final result of this simulation is shown in Figure 7-36. To achieve a ΔV of magnitude 0.000655 m/sec the FEEP thrusters were operated about 31 minutes along the anti-velocity vector. Like before a thrust magnitude of $3.5 \mu\text{N}$ with I_{sp} value of 3000 seconds were used in simulations. However during this FEEP operation along the anti velocity vector, it is important to shield the Target vehicle from the FEEP ion flow. This will be the main disadvantage and will cause severe limitation to FEEP thruster close control collaborative control applications.

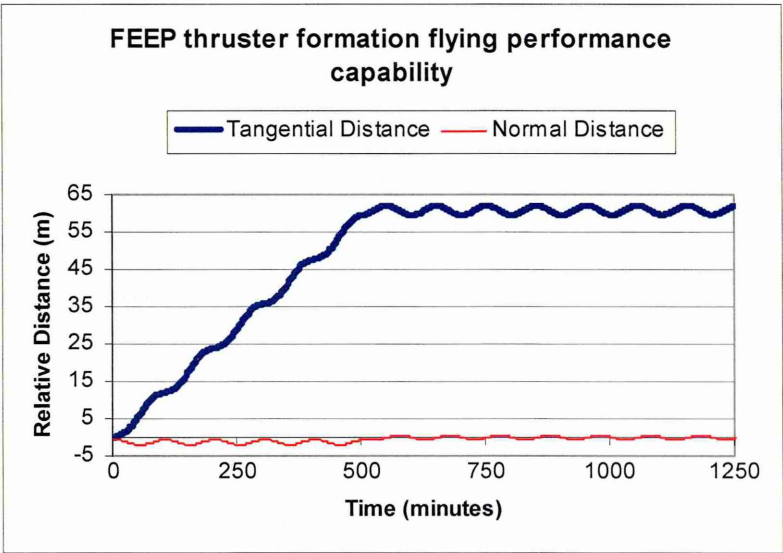


Figure 7-36 Spacecraft separation and final relative distance maintenance with FEEP thrusters. (Distance measurement origin at Interceptor)

As shown in Figure 7-36 with the selected ' ΔV ' separation, the 'Interceptor' takes about 10 hours to propagate to a distance of 60 meters. Once the 'Interceptor' has achieved the required distance from 'Target', in this case 60 meters, the FEEP thrusters could maintain this relative distance. The energy demand for the total manoeuvre is less than 0.2 watt-hours, which is only a fraction of the currently available battery capacity (taking FEEP specific power demand 60 W/mN). This shows that FEEP thrusters can be used effectively for nano-satellite collaborative control missions. However, contamination due plume impingement will be a major problem.

When low thrust propulsion systems are used, the thrusters must operate continuously for long periods of time to achieve a considerable ' ΔV '. For a constant thrust manoeuvre, assuming the mass loss is very little (which is true for FEEP thrusters), the thrust acceleration can be calculated using Equation 7-20.

$$a_{thrust} = \frac{dv}{dt} = \frac{T_{engine}}{m_{s/c}}$$

Equation 7-20

Thus for a given satellite with FEEP propulsion on board (mass of the satellite will not change considerably) with constant thrust force, achieves a constant acceleration. In a disturbance free field, the thrust acceleration is equal to the ratio of velocity change, ' ΔV ' to the thrusters operational time, ' Δt '. Figure 7-37 shows the time requirement for a constant thrust manoeuvre to perform a relative tangential distance correction in an external disturbance force field. As shown in Figure 7-38 there exists a linear relation between the applied separation ' ΔV ' and the required correction ' ΔV ' for the considered 800 km circular orbit formation flying analysis. This shows that the magnitude of the correction manoeuvre time increases linear to the applied separation ' ΔV '. During this analysis it has not taken into account the required correction in relative normal distance. However compared to the relative distance, normal distance variation is small.

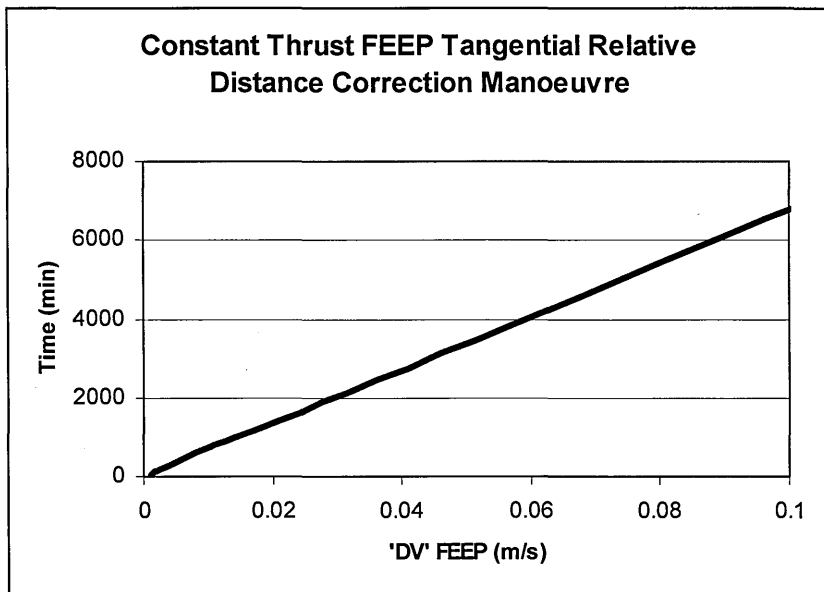


Figure 7-37 Correction time requirement for different ΔV ($T_{eng} = 3.53 \mu N$, $m = 10 kg$)

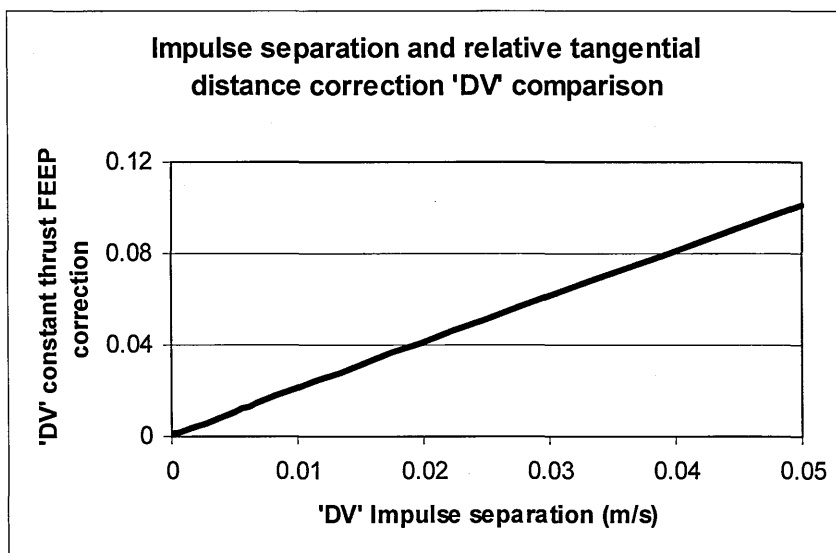


Figure 7-38 Required continuous thrust ' ΔV ' to correct the relative tangential distance after an impulse separation and free propagation to a distance of 50 meters in tangential.

As shown in Table 7-6 with higher separation ' ΔV ' (impulse), the initial separation distance will be considerable, although the correction manoeuvre had started when the relative tangential distance is 50 meters. Therefore for high separation ' ΔV ' values, low thrust FEEP emitters become less effective: correction manoeuvre times and the separation distances will become high for a short duration (couple of days), low cost FEEP demonstrator mission. Even for a standard FEEP propulsion on-board mission, with long FEEP correction times the payload operation times can be limited. The main

reason for this extended correction time is that with high separation ‘ ΔV ’, the interceptor reference orbit alters considerably and part of the propulsion energy is spent for orbital correction and the rest to overcome the perturbation forces acting on the satellite. Table 7-6 summarises this situation for few impulse separation velocities of interest.

Relative separation $\Delta V_{\text{Impulse}}$ (m/sec)	Average time to propagation to 50 meters in relative tangential (Minutes)	Maximum tangential separation between the two vehicles (m)	FEEP correction ΔV to bring the s/c back to 50 meters (m/sec)	FEEP continuous thrust time (Minutes)
0.0001	1078	51.37	0.001064450	65
0.0005	438	57.50	0.001866936	129
0.001	247	64.45	0.002506742	181
0.005	52	240.84	0.010334024	738
0.01	40	760.86	0.020918206	1440
0.05	27	15911.17	0.101388026	6885

Table 7-6 Required FEEP correction ΔV after an impulse separation at an 800 km reference circular orbit. ΔV and corresponding times takes into account perturbation forces. (10-kg reference satellite with 3.53 μ N constant thrust, STK simulations).

Note:
 Spacecraft are stacked on top of each other as of Figure 7-34
 $\Delta V_{\text{Interceptor}}$: Along velocity vector
 ΔV_{Target} : Along anti-velocity vector
 $\Delta V_{\text{Separation}} = \left| \Delta V_{\text{Interceptor}} \right| + \left| \Delta V_{\text{Target}} \right|$

According to Hill’s equations, the motion perpendicular to the orbit plane has harmonic oscillations, which are completely decoupled from the motion in the orbit plane (x-y plane). The motion along the orbit plane as illustrated above is more complex and comprises a linear drift in the along track direction (relative tangential) and a superimposed oscillation. This x-y motion results in an ellipse (actually half ellipses), which is twice as large in the along track direction (relative tangential) as in the radial direction (relative normal). Therefore it is important not to select high ‘ ΔV ’ separation, which need high ‘ ΔV ’ correction manoeuvres, and will result unacceptable oscillations. Oscillatory motion in relative tangential and normal directions after an impulsive separation (0.005 m/s) and continuous thrust propagation is shown in Figure 7-39.

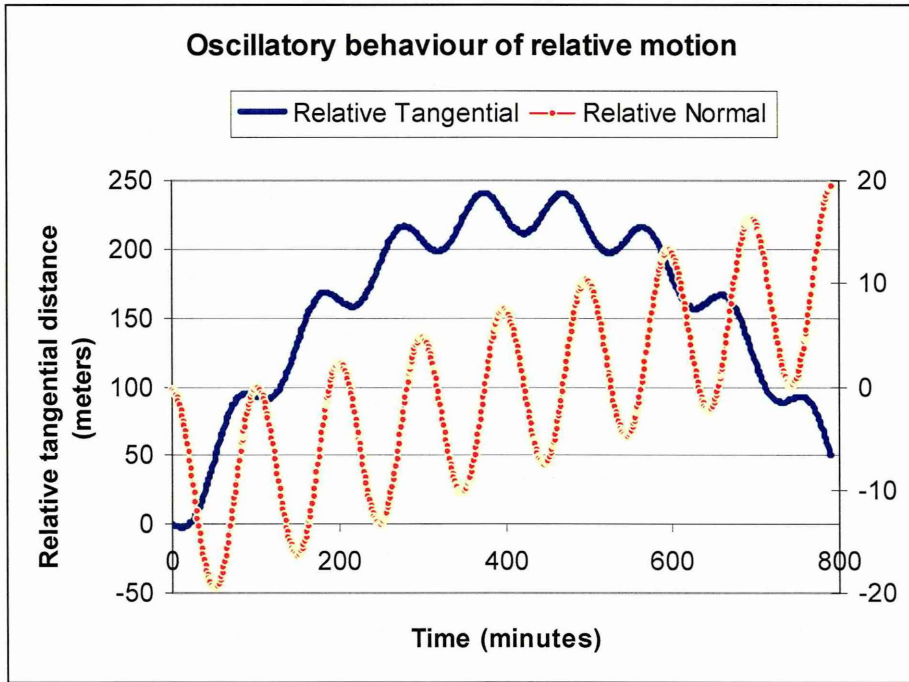


Figure 7-39 Relative distance variation after a 0.005 m/s relative impulse separation and a constant thrust duration of 738 minutes (52 minutes free propagation and ‘Interceptor’ on top of ‘Target’ prior to separation)

7.7 Summary

FEEP thrusters performance in attitude and orbit control modes were studied. The ‘Mustang’ satellite was used as the reference nano-satellite. FEEP-10 emitter was selected as it satisfied the set requirements for the propulsion system. Particular performance characteristics for this thruster were mentioned from the available vacuum chamber test data. It has showed that this thruster meets the nano-satellite limited power availability, if the thruster operates in its low thrust range. In order to understand the attitude and orbit control requirements affecting disturbances were estimated considering a LEO orbit. The analysis showed that the resulting disturbance torques and forces acting on the considered nano-satellite, is within the control capability of the selected FEEP thrusters.

FEEP thruster performances in various attitude control modes were studied using magnetic torque rods and reaction wheels as reference actuators. This analysis showed that FEEP thrusters performed well in disturbance rejection, slewing, wheel off loading and spin up/down modes. However it had serious limitations in s/c tip-off stabilisation mainly due to the selected high angular tip-off rates considered. The analysis further showed that the torque requirement is far below than the available maximum from the selected FEEP thrusters. Therefore it is possible to operate this thruster below the set power maximum in most of the considered modes of control. When the FEEP thruster

attitude control capabilities with respect to nano-satellites were compared with the selected reference actuators, this thruster showed superior performances. According to the previous studies (Marcuccio et al. 1997) FEEP's for attitude and orbit control in place of traditional systems became attractive for satellites exceeding 400 kg. However no work was carried out for satellites below 100 kg and thrust levels in μN range. The findings in this study showed that FEEP thrusters for attitude and orbit control are far attractive than traditional control methods. While the FEEP system showed limited mass and power savings compared to traditional systems for nano-satellites, it was far superior in its performances.

A unique study was performed within the field of FEEP thruster collaborative orbit control analysis: close proximity formation flying. The findings of this showed that if the initial conditions such as the separation ΔV , satellite-stacking configuration, size, etc, are met then it is possible to perform a successful formation flying using FEEP thrusters. The study also showed that the FEEP thrusters could be efficiently used to provide the separation ΔV , if the selected stacking configuration permits it. However with high ΔV separation ($> 1 \text{ m/s}$) the correction time ($> 1 \text{ day}$) and propagation distances ($> 100 \text{ m}$) exceeds the set limits in this study (relative distance between the two satellites were required to be less than 100 m). It has shown in with the current separation devices such as shape memory alloy release devices, small ΔV separation is possible to achieve. A more detail study of this release devices are given in Appendix-H.

One of the major limitations with this thruster appeared to be the possible contamination due to the selected Caesium propellant. Regardless of the propellant, any mission with such a close proximity between two satellites must take into account possible exhaust beam impingement on sensitive instruments onboard a satellite. This is especially true with electric thrusters where the beam consist of very high velocity ions (not all the ions neutralises in practise and for FEEP's ion exhaust speed can reach to about 100 km/s). If the target satellite is equipped with a proper shielding, such as shielding plates, this stream of beam coming from the 'Interceptor' can be used to propel the 'Target' satellite to meet fast relative orbit corrections demands. However this needs to be studied in more detail. It also showed that due to the present operational uncertainty, this thrusters area of application are limited. Importantly, most of the missions that FEEP thruster suits, are highly advanced and could not take any mission risks. Therefore it showed that the FEEP thrusters need a space flight demonstration to clear its image from the present hypothesis.

It was shown that with the current material and structural technology, meeting the nano-satellite mass limits with set mission goals as in this study would be a technological challenge.

8 Conclusions and Recommendations

8.1 Aims and Objectives

The aim of the work presented in this thesis is to understand the capabilities and the operational boundaries of the FEEP thrusters as nano-satellites propulsion candidate. In order to demonstrate this, the FEEP thruster performance was analysed in attitude and orbital control modes with respect to nano-satellites. Accurate attitude control and close proximity formation flying are set as the goals for the FEEP nano-satellite applications. Simulation analyses were performed mainly using STK and ESA/ADS software tools. The trend towards scientific payloads and low cost constellation designs in the near term applications of nano-satellites were identified. However, due to the complex operational characteristics of the FEEP slit type thrusters, it is recommended to test this technology in a space platform. In line with this, a nano-satellite platform is selected to perform this task.

8.2 Discussion

The application of FEEP as an attitude control thruster, gave more attractive results than its use as an orbital control propulsion system. The main reason for this was its fine pointing capability due to low thrust and the availability of a very low 'impulse bit' to perform this. When compared to other actuators such as micro reaction wheels and magnetic torque rods, the FEEP thrusters gave better performances and flexibility. The interest for FEEP thrusters were further increased due to the constant demand for disturbance free platforms and fine pointing requirements from the future scientific payloads. However, when providing general attitude control manoeuvres, while taking the system mass and power demands as decisive factors, the selection of FEEP thrusters gave no distinguish advantage over micro reaction wheels for a nano-satellite platform. This can be compared with the previous studies (Marcuccio et al. 1997) that had shown mass reduction in the range of 30 – 60 kg for satellites of mass 700 – 1000 kg for all thrusters no wheel system. Importantly the attractiveness of FEEP thrusters system is, that it can be used for both attitude and orbit control modes on demand, at the only expense of an additional power consumption for a short time. Unlike traditional reaction jet thrusters, FEEP thrusters can be operated both in proportional and pulse mode operation while still taking the advantage of its high specific impulse.

It was argued with the 10 mm long FEEP slit emitter (FEEP-10) that it is advantageous to operate the emitter and accelerator at a low voltage range (total voltage across emitter and accelerator less than 9 kV), than the possible high voltage levels. This enables the FEEP thrusters to operate at a low power and hence weight reduction in FEEP control electronics. In order to obtain the simplicity in the system, accelerator voltage was kept at a constant

value while varying the emitter voltage to obtain the required thrust profile. As the power and voltage converters operational frequency decreases with increasing voltages due to the increase in capacitor charge time, it is advantageous to have the emitter voltage as low as possible.

The applications of the FEEP-10 thruster as an orbit control propulsion system was limited due to its low thrust (mainly due to nano-satellite limited power availability) and long operational time. However when used as a drag compensator or as the propulsion system for a formation-flying mission, the initial conditions must be favourable for such a mission. When simulated for a mission scenario that it first separates from the parent vehicle and then follows the target satellite at a predefined distance, a micro Newton class thrusters such as FEEP thrusters, could meet its close proximity formation flying demands within a small range of initial conditions. The major disadvantage here was the limitation in the *S/C* stacking configuration and requirement for a very small separation ΔV to achieve favourable initial conditions. During the initial ΔV separation, as one can expect, the two satellites propagates into two different orbits. If required to propagate behind a target satellite without obstructing its orbit path, the interceptor must have a positive initial displacement along the radial vector according to Hill's theory of relative propagation. This can be achieved by attaching the Interceptor above the target satellite (i.e. along the radial vector), so that the two satellites get a small centre of mass displacement along the orbit radial. In orbital parameters this configuration gives for the interceptor centre of mass to be in a slightly higher orbit than the target centre of mass. According to the *STK* simulations, there is a small ΔV margin for successful close proximity formation flying with FEEP-10 thruster. It was also proved, that the required separation ΔV could be achieved by FEEP thrusters alone. This was shown to be the safest FEEP operation mode, as any thruster malfunction will not endanger the loss of the interceptor satellite.

Caesium was selected as the ionisation propellant mainly due to its low power requirement compared to other competitive alkali metals such as Rubidium and Indium. However the main disadvantage with Caesium is its chemically active nature, which can be the decisive factor for its selection. Therefore it is important to isolate this propellant from all types of contamination sources, not only while in orbit but also during the whole process of storing, integration and launching scenarios. It was explained that in order to overcome this contamination problem, it is important to keep the propellant in a sealed container, prior to its use in space. In line with this some engineering solutions were discussed, which can be implemented with current technology. Importantly the Indium propellant has been using successfully with pin type emitters without any contamination or operational problems. Although Indium propellant requires more power per unit thrust compared to Caesium, it is still advantageous to test this propellant in slit emitter configuration. The selection of Indium will eliminate many contamination problems that are associated with Caesium and Rubidium propellants. However for many applications discussed in this study, the required thrusters power demand is much smaller than the nano-satellites power generating capability. Therefore it is worth to study the FEEP slit emitter performance with Indium propellant.

Another major disadvantage with the FEEP thrusters, both the slit and pin type, were explained as the requirement to apply precise manufacturing methods for FEEP emitters. With the current manufacturing technology, the slit length was limited between 5 mm to 100 mm (selected FEEP-10 emitter slit length is 10 mm.). With the present manufacturing, testing, propellant loading and vacuum packing, the total cost will be far above the allocated budget, for possible nano-satellite projects.

It was shown that the FEEP thrusters give better performances at high altitudes than the low altitude applications discussed here. At high altitudes, as the drag decreases, the net control thrust force increases, providing more control authority towards attitude and orbit control manoeuvres. Further at high altitudes the presence of atomic oxygen decreases rapidly and the possible propellant contamination reduces. The prevailing extreme vacuum and solar energy can be used to 'vacuum bake' the emitters to get rid of possible trapped air atoms. After this the propellant can be introduced to the emitter.

Compared to pin emitters the slit emitters provided better performance with respect to achievable thrust. It is also showed that the number of ion emitting sites at the emitter slit depends on the applied potential field across the emitter and accelerator electrodes. As these ion emission sites are self adjusting, optimal conditions will always prevail for the slit emitter compared to the pin emitter. Any degradation at the slit will not endanger a total loss of thrust as it can be for a pin emitter. The slit emitters have a broader area of applications due to their large thrust spectrum and control capabilities. However, the Indium pin emitters have shown interesting performances compared to FEEP slit emitters with respect to operational anomalies and propellant contamination problems. The main operational problem with slit type emitter is the unpredictability of ion emission sites that give rise to inhomogeneous thrust distribution along the slit. Another area that FEEP slit emitters showed less successful in performance is the initial wetting process. The first wetting, the initial propellant flow to the emitter slit from its in-built tank, was carried out in the chamber by increasing the potential drop between the emitter and accelerator electrodes to very high values. However the main reason for this limitation is that the current FEEP emitter utilizes the surface tension forces passively. That is the surface tension forces are used to block any spillage of propellant without using any valves. Importantly it has been argued in this study that the same surface tension forces can also be used *actively* to control the movement of propellant flow. It can be further argued that the current electro-capillary effects, refers to a relationship between surface tension change of liquid metal due to applied voltage, can be successfully combined with thermo-capillary effects for first wetting. It is important to stress here that the proposed FEEP emitter combined with series of micro heaters does not pose any cost, as the current FEEP emitter must, in any case, carry a heater to melt the Caesium propellant. With the development of miniature electronics components, the FEEP thruster electronics can be developed much smaller than the proposed electronic box designed for the Shuttle get away canister mission. However for larger slit emitters with increase thrust demands and high frequency operations the FEEP electronics can be comparatively bulkier and power consuming.

8.3 Conclusions

The FEEP thrusters' nano-satellite applications gave successful results that met most of the objectives of this study. The thruster attitude control simulations gave attractive results due to its ultra fine pointing capability. The FEEP thruster orbit control capabilities, mainly nano-satellites close proximity formation flying, were met marginally as a result of limitation in thrust authority due to applied power constraints. It showed that the thruster operational effectiveness reduced greatly for low altitude and greatly dissimilar satellites due to the differential drag. An all-thrusters system gave almost equal to better attitude control performances compared to all wheel system. The FEEP thrusters can be used for nano-satellite formation-flying missions by carefully selecting its initial conditions. According to the current technology development it showed that it would continue to decrease the control electronics mass while the future satellites power generation and storing capability will continue to increase. This will allow nano-satellites with high power demanding propulsion systems such as FEEP feasible. However, Caesium propellant contamination was considered as the major problem for the present FEEP thruster successful operation. Therefore with increase power availability it is advantageous to test the FEEP slit emitter with alternative propellants to overcome the contamination problem. Most importantly this thruster operational capability needs to be validated on a real space platform. Only then it is possible to answer much of the uncertainties that surrounds this thruster technology. However with the suggestions made below the FEEP emitter technology looks promising as a future nano-satellite propulsion system.

8.4 Suggestions for Further Work

One of the major drawbacks in FEEP thrusters is the contamination due to the selected propellant (Caesium and Rubidium). However with the pin emitter studies done at the Austrian Space Research Institute Seibersdorf (ARCS), the use of Indium propellant has been successful. Though the power demand is much higher than the Caesium or Rubidium, it is worthwhile to study the use of Indium with slit emitters. The main advantage of using Indium is that it does not contaminate with residual O_2 like Caesium or Rubidium.

The electronics box for FEEP operations is currently designed for applications such as for the space shuttle get away canister mission. For this the mass penalty with respect to electronics is much lower than it will be for nano-satellite applications. It has been shown throughout this study, that there are many distributors who provide electronics that meet the demands of FEEP thrusters. Virtually most of these components are designed for terrestrial applications and therefore lack any space qualifications. As a first step towards their qualification, these components can be tested in a vacuum chamber for trouble free operation. Coatings such as epoxy can be used to seal the components to protect them from out gassing, which is the most common failure with not space qualified terrestrial electronics.

Prior to any space mission, the total FEEP system needs to be tested in vacuum chamber to understand the operational behaviour of the system. This should simulate the limited power availability, different propellant storage methods, adopted techniques for propellant introduction to the emitter slit etc. The FEEP thruster is required to be tested together with the suggested propellant containers to get an insight into the associated time delays for wetting. If successful, FEEP thrusters can be surely upgraded for a possible nano-satellite demonstrator mission.

As explained in this study FEEP thruster performance can be best measured in attitude control mode than the orbital control (collaborative control). In attitude control mode applications, it is advantageous to have a small reaction wheel as a reference control actuator to validate the FEEP test results.

It is assumed that the Caesium is stored in solid form prior to the thruster operation. However this needs to be liquefied and transport to the emitter slit for thrust generation using ion emission. This requires micro heaters for each individual thruster. Therefore by inducing a thermal gradient, the Caesium fluid can be transported to the emitter slit. This thermo-capillary 'micro pumping' can be used with FEEP emitters either to transport propellant from its in-built tank or from a central tank. If this technology is used with the central tank configuration, these thermo-capillary effects can be used as 'valves' to direct the fluid flow to the required thruster or thrusters. However as the current FEEP emitter does not have this technology associated to it, this requires developing a new emitter with in-built micro heaters. With proper current pulses to the heaters, the Caesium can be liquefied and the formed liquid can be transported towards the emitter slit. One can further develop this technology to protect the Caesium from contamination by introducing an inert gas between the emitter slit and the Caesium. Prior to the operation the inert gas can be discharged using thermo-capillary and electro-capillary forces, which is already built in to the system.

Reference List

1. Aitken, K.L., Jefferies, K.L. and Clampitt, R. (1975) Field Ionisation of Liquid Caesium on Sharp Wires. ESA/ ESTEC Internal Report, CLM/RR/E1/21, ESTEC Contract No: 2050/73, Abingdon, Oxon, OX14 3DB: AEA Research Group, Culham Laboratory, UK. ESTEC Internal Report.
2. Akimov, V.N., Gafarov, A.A., Gorshkov, O.A. and Ogloblina, I.S. (1997) Ion Thrusters for the Small Spacecraft: Elaboration of Requirements, Design Development, Prospects of Usage. AIAA-97-2780, Reston, VA: American Institute of Aeronautics and Astronautics.
3. Analytical Graphics Inc. (2003) Satellite Tool Kit Web Page. AGI.
<http://www.stk.com/>
40 General Warren Blvd.
Malvern, PA 19355; USA
Phone: 1-610-578-1000
1-800-220-4785
Fax: 1-610-578-1001 .
4. Anderson, E.H., Fumo, J.P. and Erwin, S.R. (2000) Satellite Ultraquiet Isolation Technology Experiment (SUITE). ISBN: 0-7803-5846-5, Big Sky, Montana, USA: IEEE.
5. Auweter-Kurtz, M. (1997) Resisto-Jets, Web Site. Institute fur Raumfahrtssysteme, Stuttgart, Electric Space Propulsion, Web Address: http://www.irs.uni-stuttgart.de/RESEARCH/EL_PROP/RES/e_resisto.html.
6. Bagnasco, G. and El Hamel, Z. (1995) Technology R&D Programme Support to Future ESA Science Missions. *ESA Bulletin No. 95, August 1995*
7. Bagnasco, G. and El Hamel, Z. (2001) Technology R&D Programme Support to Future ESA Science Missions. *ESA Bulletin 108, November 2001*
8. Bartoli, C., Rohden Von, H., Thompson, S.P. and Blommers, J. (1984) A Liquid Caesium Field Ion Source for Space Propulsion. No. 17, Vol ID No. 2473, UK: British Journal of Applied Physics. IOP electronic Journals, Institute of Physics Publishing, Inc., Bristol, UK.
9. Bartorelli, M. (1996-1997) Neutralizzatore a Bass Potenza per Propulsori

FEEP. Centrosazio Internal Report, ID T-069, Centrosazio, Pisa, Italy.: Internal Report.

10. Bauer, F., Bristow, J., Folta, D., Hartman, K. and Quinn, D. (1997) Satellite Formation Flying Using an Innovative Autonomous Control System (AUTOCON) Environment. AIAA-97-3821, American Institute of Aeronautics and Astronautics. Goddard Space Flight Center, Greenbelt, Maryland 20771.
11. Bender, P., Brillet, A., Ciufolini, I., Cutler, C., Danzmann, K., Fidecaro, F., Folkner, W.M., Hough, J., McNamara, P., Peterseim, M., Robertson, D. and Rodrigues, M. (1998) LISA Pre-Phase A Report. MPQ233, ESA:
12. Bertness, K.A., Kurtz, S.R., Friedman, D.J., Kibbler, A.E., Kramer, C. and Olson, J.M. (1994) 29.5% Efficient GaInP/GaAs tandem solar cells. Applied Physics Letters, Web Page
Web Address: <http://www-lib.final.gov/library/colloq/coloqfriedman.html>
Report ID 658, Issue ID: 0003-6951,
13. Bilitza, D. (1995) DGRF/IGRF Geomagnetic Field Model 1945-2005 and Related Parameters. *National Space Science Data Center, Mail Code 633, NASA/Goddard Space Flight Center, Greenbelt, MD 207*
Web Address: <http://nssdc.gsfc.nasa.gov/space/model/models/igrf.html>
14. Boland D. E. Jr. and Taesul Lee (1982) Langley Atmospheric Information Retrieval System (LAIRS). NASA/CR-3529, Washington, D.C: NASA Scientific and Technical Information Branch.
15. Brown, C.D. (2002) Elements of Spacecraft Design. 1st edn, Reston, Vagina, USA: American Institute of Aeronautics and Astronautics, ISBN: 1-56347-524-3.
16. Brown, C.D. (1998) Spacecraft Mission Design. 2nd edn, Washington, DC: American Institute of Aeronautics and Astronautics, ISBN: 1563472627. AIAA Education series.
17. Burnham, W.F., Morton, F.E., Sridharan, R., Viggh, H.E.M., Wiseman, A.J. and Zollinger, G.R. (2000) Mission Planning for Space-Based Surveillance with the Space-Based Visible Sensor. *Journal of Guidance, Control and Dynamics* **23**, No. 1,

18. Cassady R. J. (2001) Overview of Major U.S. Industrial Programs in Electric Propulsion. AIAA 2001-3226, Arlington, VA: AIAA.

19. Centropazio Web Site, C.P.I. (1998) FEEP Thrusters. Web Address:
<http://centropazio.cpr.it/Centropazio6FEEP.html>
centropazio@cpr.it.

20. Chang-Jin. and Kim C. J. (2001) Micropumping by electrowetting.
Proceedings of 2001 ASME International Mechanical Engineering Congress and Exposition

21. Curiel R.A. da Silva (2003) Small Satellites Home Page. Web Address:
<http://www.SmallSatellites.org>, Surrey Space Centre, University of Surrey, Guildford, Surrey GU2 5XH, UK: Surrey Space Centre, University of Surrey, Guildford, Surrey GU2 5XH, UK.

22. Danzmann, K. , Bender, P., Ciufolini, I. and etc. (1995) LISA: Laser Interferometer Space Antenna. ESA: ESA Pre-Phase A Report ID MPQ-208.

23. David, F.C. (1998) Development of ROCSAT-1 Program, TAO, Supplementary Issue, 1-6 March, 1999. Taiwan: National Science Council, Taiwan R.O.C. TAO, Supplementary Issue, 1-6 March, 1999.

24. De Aragon and A. Martinez (1996) Future Applications of Micro/Nano-Technologies in Space Systems. European Space Agency, ESA bulletin no.85, February 1996, The Netherlands: European Space Agency.

25. Detwiler, R. , Surampudi, S., Stella, P., Clark, K. and Bankston, P. (1995) Designs and Technologies for Future Planetary Power Systems. AIAA-95-0029, Washington, DC: American Institute of Aeronautics and Astronautics, AIAA-95-0029.

26. Eagle Picher Technologies Web Site (2000) Power Subsystems. USA: Web Address: <http://www.tech.epcorp.com/pwrsubsys/index.htm>
contact: Phone 417.623.8000// Fax: 417.781.1910, P.O. Box 47, Joplin, MO 64802. Eagle-Picher Technologies, LLC. / wmccracken@epi-tech.com.

27. EMCO High Voltage Corporation Web Site (2000) Ultra-Miniature DC to HV DC Converters. 11126, Ridge Road, Sutter Creek CA 95685: EMCO High Voltage Corporation. Web Address :

www.emcohighvoltage.com. sales@emcohighvoltage.com/
Telephone: (209)223-3626/ Fax: (209)223-2779.

28. ESA Contractor Report (1999) Free-Flying Micro Operator Study, Executive Summary; ID: FFMO/MMT/TN/10, ESA report reference. ESA CR (P) 4258. Alenia, Italy: Matra Marconi Space. ESA report reference. ESA CR (P) 4258.
29. ESA Launchers Web Site (2001) ESA, About Launchers. Web Address: http://www.esa.int/export/esaLA/ASE3DU0TCNC_launchers_0.html,
30. Fabrizio, R. (1996) Sistema Di Alimentazione Del Propellante Emodulo FEEP Per La Prove in Volo, Internal Report, Centropazio, Italy, ID: T070. Centropazio, Pisa, Italy.
31. Fearn, D.G. (1999) The Compatibility of Hollow Cathode Characteristics with a variety of Ion Thruster Designs, IEPC-99-133. Conference Proceedings IEPC-99-133
Space Department, Defence Evaluation and Research Agency, Farnborough, Hants, GU14 0LX, UK, Japanese Society for Aeronautical and Space Sciences.
32. Fearn, D.G. (1987) Factors influencing the integration of the UK-10 Ion Thruster System with a spacecraft. AIAA-87-1004, 1633 Broadway, New York, NY 10019: American Institute of Aeronautics and Astronautics.
33. Fehringer, M., Rudenauer, F. and Steiger, W. (1999) Micronewton Indium Ion Thrusters. Web Address: <http://www.arcs.ac.at/space/pdf/Micronewton.PDF>
Austrian Research Centre Seibersdorf, A-2444 Seibersdorf., michael.fehringer@arcs.ac.at.
34. Filliben, J.D. (1999) Electric Propulsion for Spacecraft. Web Address: <http://www.jhu.edu/~cpia/electric.html>
CPIABulletin 9/96, filli_j@jhunix.hcf.jhu.edu, CPIABulletin 9/96.
35. Franklin F. G., Powell J. D. and Emami-Naeini A. (1994) Feedback Control of Dynamic Systems. 3rd edn, USA: Addison Wesley Publishing Company, ISBN: 0-201-53487-8. ISBN: 0-201-53487-8.
36. Fridlund, C.V.M. (2000) Darwin - The Infrared Space Interferometry Mission. *ESA Bulletin* 103 pp 20-25. Space Science Department, ESA

Directorate of Scientific Programmes, ESTEC, Noordwijk, The Netherlands.

37. Fulvio Drigani (2000) Earth Observation Earthnet Online. Web Address:
<http://www.esa.int/export/esaCP/index.html>
contactesa@esa.int, ESA: contactesa@esa.int.
38. Gaposchkin, E.M., Braun, C.v. and Sharma, J. (2000) Space-Based Space Surveillance with the Space-Based Visible. Journal of Guidance, Control and Dynamics, Article 23, No. 1, Lincoln Laboratory, Massachusetts Institute of Technology, Lexington, Massachusetts 02420.
39. Genovese, A. , Marcuccio, S., Pozzo, D.D. and Andrenucci, M. (1997) FEEP Thruster Performance at High Background Pressure. Proceedings of the 25th Electric Propulsion Conference, IEPC-97-186
Available: <http://www.centropazio.cpr.it/HighPres.pdf>, Cleveland, OH: IEPC.
40. Genovese, A. (1995) Drag-Free Satellite Control Microthrust FEEP Tests. ESTEC Contract 10016/WO 12; Centropazio Internal report, DASA Sub-Contract Z7290 50225/CS, Centropazio Internal report.
41. Ghislanzoni, L., Petagna, C. and Trippi, A. (1990) The Application of Arcjet Propulsion Systems for Geostationary Satellites, AIAA-90-2548. Conference Proceedings: AIAA-90-2548, Washington, D.C.: AIAA. ESA/ESTEC Technical Directorate.
42. Golden S. & Associates (1997) ITHACO Space Systems, Inc. Web Address:
<http://www.ithaco.com/index.html>
ITHACO Space Systems, Inc. , 950 Danby Rd., Suite 100 ,Ithaca, NY 14850-5720
marketing@ithaco.com//Tel: 607-275-9827 or 607-275-9822 //Fax: 607-272-0804, ITHACO Space Systems, Inc. , 950 Danby Rd., Suite 100 ,Ithaca, NY 14850-5720 .
43. Gomer, R. (1979) The Mechanism of Liquid Metal Electron and Ion Sources. Journal Article Applied Physics 19 (1979),
44. Gorshkov, O.A. (1998) Low-Power Hall Type and Ion Electric Propulsion for the Small Sized Spacecraft. AIAA-98-3929, Reston, VA 20191 : AIAA.

45. Gorshkov, O.A., Koroteev, A.S., Maslennikov, N.A., Murashko, V.M., Lukyashenko, V.I., Utkin, V.V., Kim, V. and Popov, G.A. (1998) Activity of Russian Space Agency in The Field of Electric Propulsion, AIAA-98-3176. Conference Proceedings: AIAA-98-3176, Reston, VA: American Institute of Aeronautics and Astronautics.
46. Gorshkov, O.A. and Muravlev, V.A. (1999) Low Power Ion Thrusters with Slit type Grid system. Conference Proceedings, AIAA-99-2825, AIAA.
47. Grahn S. (2003) Swedish Space Corporation Satellites . Web Address: <http://www.ssc.se/ssd/index.html>
Sven.Grahn@ssc.se , Sven.Grahn@ssc.se .
48. Günther B., Göhl A., Müller G., Givargizov E., Zadorozhnaya L., Stepanova A., Spitsyn B., Blaut-Bachev A.N., Seleznev B. and Suetin N. (2001) Microscopic field emission investigation of diamond and AlN coated Si tips. *Journal of Vacuum Science Technology* **B19**, 3,
49. Hechler, F. and Eckstein, M.C. (1970) Drag and Lift Perturbations of the Orbit of the TD-1 Satellite. ESRO-SR-14, 92 Neuilly-sur-Seine (France): European Space Research Organisation .
50. Helvajian, H. (1995) Microengineering Technology for Space Systems. AIAA 1997. Conference Proceedings: AIAA 1997 ID1884989055. The aerospace corporation, Segundo, California.
51. Hideyuki, H. and Itsuro K. (1999) Characterization of a Very Low Power Arcjet. Reston, VA, 20191-4344: American Institute of Aeronautics and Astronautics. Conference Proceedings: AIAA-99-2435.
52. Iannotta, B. (Apr 10, 1999) Pocket Rocket. *New Scientist* **Vol. 162** (Issue 2181):38 pages New Scientist. Web Address: www.newscientist.com.
53. Janson, S.W. (1994) Chemical and Electrical Micropropulsion Concepts for Nanosatellites. AIAA-94-2998, American Institute of Aeronautics and Astronautics.
54. Janson, S.W. (1992) The Impact of Advanced Diagnostic Techniques on Electrothermal Thruster Design. AIAA-92-3242, Washington, D.C.: American Institute of Aeronautics and Astronautics, AIAA-92-3242.
55. Janson, S.W. (1993) The On-Orbit Role of Electric Propulsion. AIAA-93-

2220, Washington, DC: American Institute of Aeronautics and Astronautics, AIAA-93-2220.

56. Jordan, I.J.E. (2000) Electric Propulsion, Which one for my Spacecraft. 744 Spacesystems-1 course at JHU, Whitling school of engineering.
57. Jung J.H. , Ju B.K. , Lee Y.H. , Jang J. and Oh M.H. (1999) Emission Stability of a Diamond-like Carbon Coated Metal-tip Field Emitter Array. *Journal of Vacuum Science Technology* Vol. B17 - 2, Pages 486-488.
58. Koelle, H.H. and Koelle, H.K. (1961) 1st edn, New York: McGraw-Hill Book Company. ISBN: 38006101189291.
59. Kuninaka, H. (2002) Activities on Electric Propulsion in Japan- Space Flight from Basic Research. AIAA 2002-3563, American Institute of Aeronautics & Astronautics.
60. Larson, W.J. and Wertz, J.R. (1995) Space Mission Analysis and Design. 2nd edn, USA: Microcom, Inc., Kluwer Academic Publishers. 1-881883-01-9.
61. Lau, K., Colavita, M., Blackwood, G., Linfield, R., Shao, M. and Gallager, D. (1997) The New Millennium Formation Flying Optical Interferometer. AIAA 97-3820, AIAA GN&C Conference: American Institute of Aeronautics and Astronautics.
62. Leach, R.Dr. and Kerry L. N. (2003) Discussion of Micro-Newton thruster requirements for a drag-free control system, 16th Annual USU conference on small satellites. Web Page: <http://www.aria.seas.wustl.edu/SSC02/papers/viii-1.pdf>. 16th Annual USU conference on small satellites.
63. Lithium Technology Corporation Web Site (2000) Lithium Ion Polymer Technology. Address: <http://www.lithiumtech.com/index.html>. Lithium Technology Corporation™, 5115 Campus Drive, Plymouth Meeting, Pennsylvania 19462, Phone: (610) 940-6090 Fax: (610) 940-6091.
64. Lockheed Martin Communications and Power Center (1998) Ultra High-Density DC/DC Power Converters. Web Page: <http://www.payloads.com/> Phone : 215-497-1559/ Fax: 215-497-1564. Lockheed Martin in Communicatios and Power Center, 100 Campus

Drive, Newtown PA 18940.

65. Marconi Applied Technologies Web Site (1997) EEV Solar Power for New UK Spacecraft. Web Address: <http://www.eev.com/solarpr2.htm>
webmaster@eev.com.
66. Marcuccio, S. (1998a) FEEP Thrusters. Web Address:
<http://centrospazio.cpr.it/>.
67. Marcuccio, S. (1998b) Flight Demonstration of FEEP on Get Away Special. Paita, L. and Saviozzi, M., (Eds.) AIAA-98-3332. Proceedings of the 34th Joint Propulsion Conference, Cleveland, OH.: AIAA. Conference Proceedings: AIAA-98-3332. American Institute of Aeronautics and Astronautics (AIAA).
68. Marcuccio, S. and Genovese, A.A.M. (1997) FEEP Microthruster Technology Status and Prospects. 48th International Astronautics Federation Congress, Turin, Italy: IAFC. Centrospazio, Via Gherardesca 5, 56014 Ospedaletto, Pisa, Italy.
69. Marcuccio, S., Giannelli, S. and Andrenucci, M. (1997) Attitude and Orbit Control of Small satellites and Constellations with FEEP Thrusters. IEPC-97-188. Centrospazio, Pisa, Italy: Proceedings of the 25th Electric Propulsion Conference. Conference Proceedings: IEPC-97-188.
70. Marston, R.A. Marston, R.A., (Ed.) (2000) Introduction. Astrium Ltd. Gunnels Wood Road, Stevenage, Hertfordshire, SG1 2AS, England: E-Mail: bob.marston@astrium-space.com //Telephone: +44-(0) 1438-773331//Telefax: +44 (0) 1438-773555.
71. Massey B. S. (1968) Mechanics of Fluids. 6th edn, 11, New Fetter Lane, London: Chapman & Hall. ISBN: 0-412-34280-4.
72. McCarron, T. (1997) Tanner's General Chemistry Web Site. Web Address: <http://www.tannerm.com/first.htm>.
73. Meier, D.L., Szedon, J.R., Bartko, J. and Chung, M.A. (1988) Responce of Single Junction GaAs/GaAs and GaAs/Ge Cells to Multiple Doses of 1 MeV Electrons. NASA CP-3030. page 178 NASA, Washington, DC: NASA Lewis Research Center, Cleveland, Ohio. Conference Publication.

74. Micci, M.M. and Ketsdever, A.D. (2000) Micropropulsion for Small Spacecraft. 187. Virginia, USA: American Institute of Aeronautics and Astronautics, Inc. 1-56347-448-4.

75. Mitterauer, J. (1989) FEED: Mass and Optical Spectroscopy of Beam Components. ESA/ESTEC Internal Report: European Space Agency Contract Report, ID 6545/85/NL/PH. ESA/ESTEC.

76. Morrison, S.R. (1977) The Chemical Physics of Surfaces. Stanley Roy, (Ed.) 2nd edn, 233, Spring Street, New York, N.Y. 10013: Plenum Press. ISBN: 0-306-43549-7.

77. Moss, P.J.B. (1993) Space propulsion systems and performance. Cranfield MSc Rocket Propulsion Lecture (Unpublished), Cranfield, UK: Cranfield University, Cranfield, Bedfordshire, MK43 0AL; Tel +44-(0)1234-750111.

78. Motorola Products Web Site (1999) Global Positioning Systems (GPS) Products. Web Address:
<http://www.motorola.com/ies/GPS/products/gpsprod.html>
 Motorola, 4000 Commercial Avenue, Northbrook, IL 60062 USA
 Call: 888.298.5217 ; fax : 847.714.7325
 Europe Call: +44-1462.831.111 ; fax: +44-1462.831.390.

79. Myers, R.M. (1993) Electromagnetic Propulsion for Spacecraft. AIAA-93-1086, Washington, D.C.:

80. Nerem R.S., Lerch, F.J., Marshall, J.A., Pavlis, E.C., Putney, B.H., Tapley, B.D., Eanes, R.J., Ries, J.C. and Schutz, B.E. (1994) Gravity Model Development for TOPEX/POSEIDON: Joint Gravity Models 1 and 2. *Journal of Geophysical Research* **99**,

81. Ovchinnikov, M.Yu. (1994) Small Satellite and in the World: Current State. Paris, France: International Astronautical Federation. Report ID: IAA-94-IAA.11.1.761.

82. Panasonic Web Site (1998) Lithium Ion Batteries. Address:
http://www.panasonic.com/industrial_oem/battery/battery_oem/chem/lithion/lithion.htm.

83. Patterson, M.J. and Oleson, S.R. (1997a) Low Power Ion Propulsion for small Spacecraft. AIAA. Conference Proceedings, AIAA-97-3060.

84. Patterson, M.J. and Oleson, S.R. (1997b) Low-Power Ion Propulsion for Small Spacecraft. Reston, VA: American Institute of Aeronautics & Astronautics. Conference Proceedings: AIAA-97-3060.

85. Peffer, A. (2000) Development and Transition of Low-Shock Spacecraft Release Devices for Small Satellites. USA: Conference Proceedings: SSC00-XI-5. Kakson and Tull, 1900 Randolph Road, Suite H, Albuquerque, NM 87106, 505-853-6322/andrew.peffer@kirtland.af.mil.

86. Pisacane, V.L., Moore, R.C. and Sullivan, R.M. (1994) Fundamentals of Space Systems. Oxford : Oxford University Press. Inc. ISBN: 0-19-507497-1.

87. Polk, J.E., Kakuda, R.Y., Anderson, J.R., Brophy, J.R., Rawlin, V.K., Patterson, M.J., Sovey, J. and Hamley, J. (2001) Performance of the NSTAR Ion Propulsion System on the Deep Space One Mission. Reston AIAA 2001: AIAA. Conference Proceedings: AIAA-2001-0965.

88. Pollard, J.E., Jackson, D.E., Marvin, D.C. and Jenkin, A.B. (1993) Electric Propulsion Flight Experience and Technology Readiness. Presented at the 29th joint propulsion conference and exhibit, Monterey, CA : American Institute of Aeronautics and Astronautics. Conference Proceedings: AIAA-93-2221.

89. Praxis Inc Clementine Web Site (1994) Clementine. Web Address: <http://www.pxi.com/clementine/sensors/startracker.html>, Lawrence Livermore National Laboratory (LLNL) Praxis Inc. webmaster@pxi.com.

90. Rayman, M.D. (1998) Advanced Technologies, Solar Electric (ION) Propulsion. Web Address: <http://nmp.jpl.nasa.gov/ds1/tech/sep.html> Jet Propulsion Laboratory, NASA DS1 Education & Public Outreach, MailStop 301-235, 4800 OakGrove Dr. Pasadena, CA 91109,

91. Roberts, P.C.E., Bowling, T.S. and Hobbs, S.E. (2002) MUSTANG: A Technology Demonstrator for Formation Flying and Distributed Systems Technologies in Space. Bowling, T.S., (Ed.) Page 327-335. Cambridge, UK: Cranfield University Press. 1 871315 79 4. Cranfield University, Cranfield College of Aeronautics.

92. Roberts, P.Dr. (Sep, 2001) Project Mustang Formation Flying at 17,000 mph.

Cranfield University Aerogram 4, . Cranfield, UK: Cranfield University. Table-2 Mass budget.

93. Rycroft, M. (2001) "Smaller Satellites: Bigger Business? Concepts, Applications and Markets for Micro/Nanosatellites in a New Information World" . ISBN: 1-4020-0199-1, 3300 AA Dordrecht, The Netherlands : KLUWER Academic Publishers.

94. Sackheim, R.L., Fritz, D.E. and Macklis, H. (1979) The Next Generation of Spacecraft Orbit Control Propulsion Systems. USA: AIAA. Conference Proceedings . AIAA 79-1301.

95. Sarafin, T.P. (1995) Spacecraft Structures and Mechanisms. 1st edn, USA: Microcosm, Inc. and Kluwer Academic Publishers. ISBN: 0-7923-3476-0.

96. Sharma, J. (2000) Space-Based Visible Space Surveillance Performance. *Journal of Guidance, Control, and Dynamics* **23**, No. 1 Lincoln Laboratory, Massachusetts Institute of Technology, Lexington, Massachusetts 02420.

97. Sidi, M.J. (1997) Spacecraft Dynamics and Control. 1st edn, Cambridge: Cambridge University Press. ISBN: 0-521-55072-6.

98. Society of Automotive Engineers (1984) Space Systems Technology . Volume ID: 38006100924706. Warrendale, PA: Society of Automotive Engineers . ISBN: 0898838142 .

99. Sovey, J.S., Hamley, J.A., Patterson, M.J. and Rawling, V.K. (1992) The evolutionary development of high specific impulse electric thruster technology. Myers, R.M., (Ed.) Washington, DC: American Institute of Aeronautics and Astronautics. Conference Proceedings: AIAA-92-1556.

100. Stamerjohanns, G. (1998) EUROCKOT User's Guide. 2 edn, Bremen, Germany: EUROCKOT Launch Services GmbH. EUROCKOT Launch Services GmbH, P.O. Box 286146, D-28361 Bremen, Germany.

101. Stewart, D. , Wilson, P.D. and Nicholas, M.G. (1980) Study on additional propellants for field emission thrusters. ESA/ESTEC Internal Report No. 3806/78/NL/JS(SC); R.664/June/1980.

102. Strömdal, T. (2001) The Munin Satellite Web Page . Web Address:
<http://munin.irf.se>. IRF, P.O. Box 812, 981 28 Kiruna, Sweden.

103. Stuhlinger, E. (1964) Ion Propulsion for Space Flight. New York: McGraw-Hill.

104. Sutton, G.P. and Biblarz, O. (2001) Rocket Propulsion Elements. 7th edn,
 New York, USA: John Wiley & Sons. ISBN: 0-471-32642-9.

105. Tajmar, M. (2002) Survey on FEEP Neutralizer Options . Reston VA, USA:
 AIAA. Conference Proceedings: AIAA-2002-4243.

106. Valentian, D., Bugeat, J., Petit, R., Bartoli, C. and Rohden, M.V. (1989)
 Application of Field Emission Thruster to Spacecraft Control and Orbit
 Positioning. 25th Joint Propulsion Conference, Monterey, CA: AIAA.
 Conference Proceedings: AIAA-89-2277.

107. Vallado, D.A. (1997) Fundamentals of Astrodynamics and Applications.
 USA: The McGraw-Hill Companies, Inc. ISBN: 0-07-066834-5.

108. Ward, A.K. and Gardner, S.J. (1994) Available Technologies for Small
 Satellites. IAF-94-U.3.474, Jerusalem, Israel: International
 Astronautics Federation.

109. Watson, S.D., Martin, A.R., Latham, P.M., Bond, A. and Preest, A.A. (1990)
 An Electrically Propelled Upper Stage for the Launch of Small
 Satellites. AIAA-90-2622, Washington, D.C.: American Institute of
 Aeronautics and Astronautics.

110. Weizer, V.G. (1971) Field Evaporation Ion Source With Possible Application
 to Electrostatic propulsion . E-6540. Lewis Research Center, NASA,
 Cleveland, Ohio 44135: National Aeronautics and Space
 Administration (NASA), Washington, D.C. Conference Proceedings:
 NASA TN D-6593.

111. Welker, D. and Dudzinski, L.A. (2001) Magneto-Plasma-Dynamic (MPD)
 Thrusters. Web Page:
http://trajectory.lerc.nasa.gov/aig7820/projects/electric/ep_devices.htm
 1. Electric Propulsion Homepage; NASA Lewis Research Center.

112. Wertz, J.R. (1980) Spacecraft Attitude Determination and Control. Kluwer
 Academic Publishers. ISBN: 9027712042. 629.78.

113. Wertz, J.R. and Larson W. J. (1999) Space Mission Analysis and Design. 3rd edn, USA: Microcosm Press, Kluwer Academic Publishers. ISBN: 1-881883-10-8.
114. Wiesel, W.E. (1989) Spaceflight Dynamics. 1st edn, 323 United States of America: McGraw-Hill Inc. ISBN: 0-07-070106-7.
115. William, M.F., Fehringer, M., Ruedenauer, F. and Steiger, W. (1998) Laser Interferometer Space Antenna. ISBN: 0-12-379050-6, Pasadena, California: The American Institute of Physics. Austrian Research Centre Seibersdorf/ A-2444 Seibersdorf.
116. Willoughby, J.K. (1994) An Implementation Strategy to Reduce Mission Operations Costs for Small Satellites Programs . Paris, France: International Astronautics Federation. Report ID: IAA-94-IAA.11.3.772.
117. Wilson, A. (1996) Jane's Space Directory 1996-97. 12th edn, Sentinel House, 163 Brighton Road, Coulsdon, Surrey, UK: Jane's Information Group Limited. ASIN: 0-7106-1457-1.
118. Wilson, P.D., Hatt, B.A. and Rogers, T. (1989) FEED Emitter Module Industrialisation (Phase-2). R1152/2/September 1989, Internal report, ESA/ESTEC.
119. Winter, M. (1993) Chemistry WebElements: Periodic Table. Web Page: <http://www.webelements.com>. The University of Sheffield and WebElements Ltd, UK.
120. Wolfgang, S. (2003) The Seibersdorf Indium Liquid Metal Ion Source (LMIS). Web Address: <http://www.arcs.ac.at/space>
Email: wolfgang.steiger@arcs.ac.at
Fax: +43 (0)2254-780 2813, Internal Report, Austrian Research Centers, A-2444 Seibersdorf, Austria.
121. Zakrzewski, C., Benson, S. and Sanneman, P. (2002) On-orbit testing of the EO-1 pulsed plasma thruster. Indianapolis Indiana: American Institute of Aeronautics and Astronautics. Conference Proceedings: AIAA-2002-3973.
122. Zandbergen, B.T.C. (1986) Field Emission Electric Propulsion (FEED). Report ID: TSP/BZ/996/ar, ESA/ESTEC Internal Report, The Delft, The Netherlands: ESA, ESTEC; an investigative report.

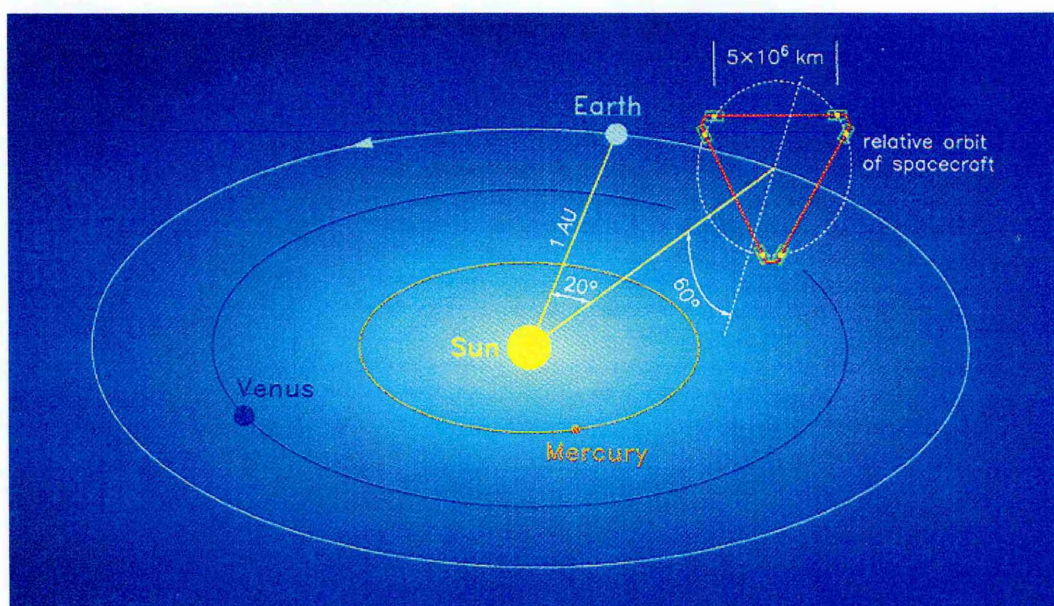
123. Zhirnov, V.V., Wojak, G.J., Choi, W.B., Cuomo, J.J. and Hren, J.J. (1997) Wide Band Gap Materials for Field Emission Devices. *Vacuum Science Technology Journal* **Journal Volume ID: 1733-1738** (Issue ID: 15-3):
124. Zurlev, D.N. and Forbes, R.G. (2003) Field Ion Emission: the effect of electrostatic field energy on the prediction of evaporation field and charge state. L74-L78 Institute of Physics Publishing Ltd. Journal ID: PII:S0022-3727(03)63916-1.

Appendices

Appendix A

A.1 LISA mission summary

The LISA space experiment consists of 3 pairs of identical satellites arranged in an equilateral triangular configuration in space. Separated by 5-million kilometres in space these 3-satellites forms a giant ‘Michelson’ interferometer. Synchronised laser beams are sent around the triangle, and flight times are accurately measured by interferometric techniques. The arrival of a gravity wave will periodically change flight times between satellites. A difference of flight times along the three legs of the interferometer gives information on the direction of arrival.



LISA - Laser Interferometer Space Antenna (Danzmann et al. 1995)

Ultra-high precision attitude and orbit control is necessary for aiming the laser beams and keeping the distance between the LISA satellites constant. The required accuracy can only be achieved by using highly controllable FEEP thrusters. LISA drag compensation propulsion system consists of six clusters. Each cluster contains four thrusters, mounted on the outside of the main structural cylinder of the spacecraft. This arrangement will provide full control in case of a loss of one or one cluster of thrusters. Each thruster will be able to provide thrust in the range 5 to 100 μN , with a noise level of less than 0.1 μN . About 50 grams of propellant for the small ion engines will be needed during the five years of science operations.

Spacecraft	3-axis stabilised drag-free
Mass	Each S/C in orbit 203-kg Launch mass about 400-kg
Propulsion module	132-kg for each spacecraft with 27-kg of propellant per S/C.
Power	151 W for each s/c in orbit
Drag-free performance	10^{-15} m/s ² (rms) in the band 10^{-4} to 10^{-1} Hz achieved with 6x4 Caesium FEEP thrusters.
Mission Lifetime	2 years (nominal); 10 years (extended)
Launcher	Delta337925H, (9.5ft fairing)
Jettisonable propulsion module	Provide ΔV of 1300m/s per S/C using solar-electric propulsion (Ion drive)

LISA mission summary (Bender et al. 1998)

A.2 'Mustang' Project Summary

'Mustang' which stands for 'Multi University Space Technology Advanced Nanosatellite Group', is university led collaboration with the UK space industry to demonstrate formation flying in space, micro-systems and relevant associated technologies. Cranfield University's space team and the University of Southampton jointly lead the project, which is sponsored by the British National Space Centre (BNSC). Also involved are Astrium, Cranfield's School of Industrial and Manufacturing Science (SIMS), Cambridge University, Oxford University and Imperial College.

The primary aim of the project is to demonstrate formation flying in space and the technologies associated with this. The current design for the formation-flying payload uses three reaction wheels and magnetic torque rods for attitude control and a set of twelve cold gas thrusters to control each spacecraft's position. Sensing of the relative position of the two spacecraft is carried out using a combination of differential GPS and laser ranging. The designs of the hardware and control algorithms to enable the formation flying are currently being developed by the University of Southampton. Circular, true polar orbit (600 km altitude and 90 deg inclination) has been selected as the operational orbit. Two 'Mustang' type satellites will be used in stack configuration during the launch.

Appendix B

B.1 Survey of small satellites from 1980 - 1991

Year	Month	Company/ Sponsor	Satellite	Mission	Mass (kg)	Launch vehicle
1986	Aug	JARL/Japan	JAS-1	Communication	50	H-1A
1990	Feb	Japan	DEBUT (ORIZURU)	Research	50	H-1-6F
	Feb	JARL/Japan	JAS-1b (FUJI 2)	Communication	50	H-1-6F
	Jan	AMSAT	WEBERSAT	Communication	10	Ariane 4
	Jan	Univ. of Surrey	UoSAT 3 (D)	Communication / Research	45	Ariane 4
	Jan	Univ. of Surrey	UoSAT E	Communication / Research	45	Ariane 4
	May	DSI	MACSAT	Military	23	Scout
1991	July	DSI	ASTP/Lightsat	Communication (7 micro-sats)	23 each	Pegasus 2
	July	U. of Surrey	UoSAT 5	Communication	49	Ariane 4
	July	Tech U Berlin	TUBSAT-A	Communication	25	Ariane
		ESIEESPACE	SARA	Radio Astronomy	27	Ariane
		OSC/CIT	ORBCOMM- X (VaSTAR)	Communication	17	Pegasus 2

Small satellites with mass BOL less than 50 kg (Larson and Wertz, 1995).

Basic launcher/satellite facts (Wilson, 1996):

- 147 primary satellites were launched commercially over the period of 1988-1995
- 85% of them had a telecommunication payload.
- 'Arianespace' accounts for 60% of launchers during 1988-1995.
- Projected 'Arianespace' long-term launch rate was estimated to be around 15 annually.
- China has offered at least 30% less launch cost than its US counterpart during this period.
- Projected mid term payload cost must bring down to around \$2,200/kg instead of current \$22,000/kg (valid 1995, to GEO, US standards).

B.2 Alternative propulsion system characteristics:

Propulsion system parameters	Biprop	Resistojet N ₂ H ₄	Arcjet N ₂ H ₄	Hall Xe	Ion Xe	PPT
Typical input power to PPU (kW)	N/A		0.4 to 0.8 low power 1.2 to 1.8 SOA 30 high power	0.5 to 1.5	0.44 to 2.5	N/A
Typical PPU Mass (kg)	N/A		2 to 4		12 to 18	
Typical PPU efficiency			0.9		0.82 to 0.92	
Typical thruster mass (kg)			1 to 2		~1.5 to 7	
I _{sp} (sec)	315 sec	300 - 310 sec	340 - 600 sec	~1600 sec	2550 - 3500 sec	~1588 sec
Typical thruster efficiency	N/A	.9	.30 - .35	0.58	.5 - .62	0.45
Propellant tankage fraction	~5.3%	~5.3%	~5.3%	10-15%	10-15%	N/A
Number of ON/OFF Cycles		>1000 for GEO SK	> 1000 for LEO to GEO	>1000 for LEO to GEO	>1000 for LEO to GEO	1000/sec of operation
Rated life at power level (hrs)	N/A	N/A	~1500	~7,000 hrs@1.4 kW (SPT-100)	~15,000 hrs@0.6 kW (Artemis mission)	~4,000 hrs@1.5 kW

Characteristics of electric propulsion devices (Welker and Dudzinski , 2001).

PPU: Power processing unit
 N/A: Not available
 Biprop: Bi-Propellant
 Pwr.: Power
 SOA: State-of-the-art
 PPT: Pulsed Plasma Thrusters
 SPT: Stationary Plasma Thruster
 SK: Station keeping
 LEO: Low Earth Orbit
 GEO: Geosynchronous Earth Orbit

B.3 Representative propulsion requirements

Manoeuvre	Thruster Requirements			Duty Cycle/Usage Characteristics
	Total Impulse (N.s)	Frequency	Burn Duration (s)	
<i>Orbit Control</i> E-W Stationkeeping	>5000	~Weekly or monthly	Up to 3600	Steady state firing
North-South Station Keeping (SK) (Inclination SK)	>100000	~Twice per day	Up to 7200	High total impulse
Reposition	>6000	2 steady thrust periods for each manoeuvre	Up to 3600	Low pulses Off-pulse modulation
Station Acquisition (Attitude control)	>2000	Few burns	0.5 to 60	Pulse mode firings Low total impulse High pulses Limit cycle minimum bit, low rate mode

Typical thrust requirement for a 3-axis control geo-synchronous satellite with a 7-year mission, (Sackheim et al. 1979).

Manoeuvre	Thruster Requirements			Duty Cycle/Usage Characteristics
	Total Impulse (N.s)	No. Of Pulses	Burn Duration (s)	
Velocity Control Station-keeping E-W	>10000	> 40000	Up to 36000	Pulse mode (trains)
N-S	>200000	>100	Up to 7200	High total impulse, all steady state
Reposition	>25000	>5000	Up to 3600	Pulse mode (trains)
Precession Control	>10000	>10000	0.02 to 0.15	Pulse mode firings
Spin control	>8000	>10	Up to 1000	Short steady-state burns

Typical spin stabilised satellite thruster requirements for a 10-year mission, (Sackheim et al. 1979).

Propulsion mode	Typical requirement
Initial spin-up	1 to 60 rpm
LEO to higher orbit raising	60 to 1500m/s
➤ Drag makeup ΔV	60 to 500 m/s
➤ Controlled re-entry ΔV	120 to 150 m/s
On-orbit operation	
➤ De-spin	60 to 0 rpm
➤ Spin control	± 1 to ± 5 rpm
➤ Orbit correction	15 to 75 m/s per year
➤ East-west station keeping	3 to 6 m/s per year
➤ North-south station keeping	45 to 55 m/s per year
Attitude control	3 –10% of total propellant mass
Acquisition of Sun, Earth, Star	Typically < 5000 N.s, 1K to 10K pulses, 0.01 to 5.0 s pulse width.
On-orbit normal mode control with 3-axis stabilisation, limit cycle	100K to 200K pulses, minimum impulse bit of 0.01 N.s to 0.25 s pulse width.
Precession control (spinners)	Typically < 7000 N.s, 1K to 10K pulses, 0.02 s to 0.2 s pulse width.
Momentum management (wheel unloading)	5 to 10 pulse trains every few days, 0.02 to 0.10 s pulse width.
3-axis control during ΔV	On/off pulsing, 10K to 100K pulses, 0.05 to 0.20s pulse width.

Typical functions and requirement figures for space propulsion (Larson and Wertz, 1995).

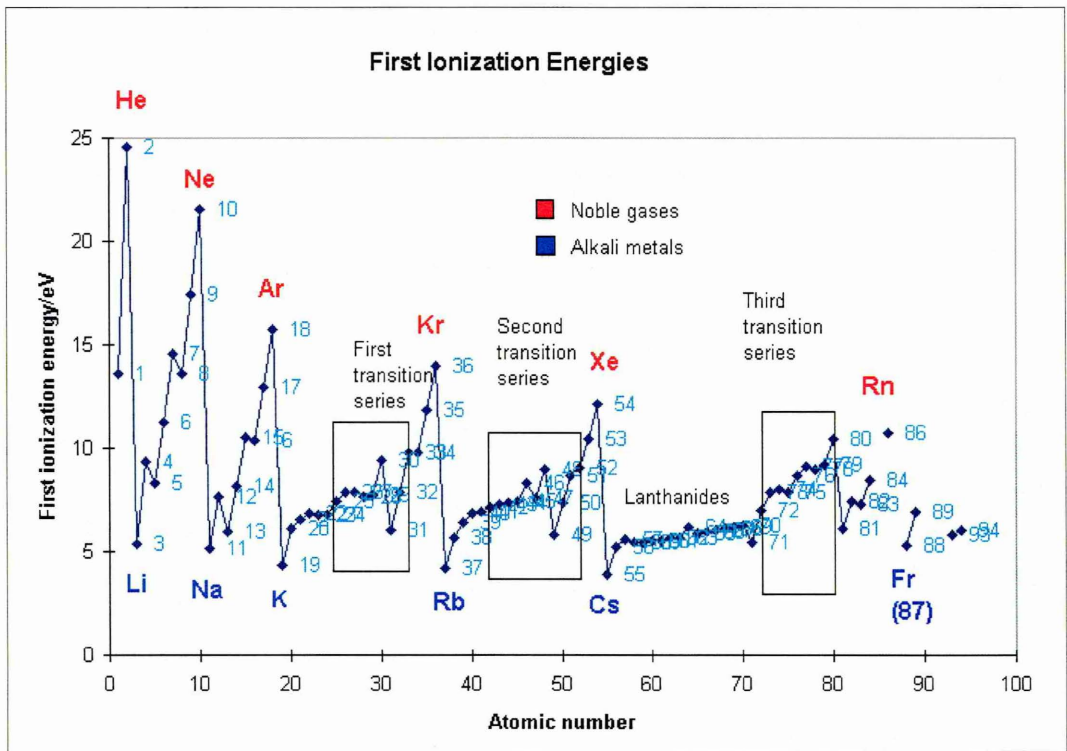
B.4 Overview of electric propulsion applications:

Propulsion System	Spacecraft/ Launch date	Thrust (mN) / Power (W) / No. Of thrusters	Propellant	Application
Ablative Pulsed Plasma	LES-6, USA (1968)	0.018 / 3 / 4	Teflon	EWSK
	TIP II, USA (1976)	0.40 / 30 / 2	Teflon	Drag makeup
	TIP III, USA (1981)	0.40 / 30 / 2	Teflon	Drag makeup
	ETS-IV, Japan (1981)	0.09 / 20 / 4	Teflon	Experiment
	MDT-2A, China (1981)	-	Teflon	Experiment
	NOVA 1,2,3, USA (1984-1988)	0.29 / 30 / 2	Teflon	Drag makeup
Stat Plasma Thruster	Meteo series, USSR, (1972 -)	20 / 390 / 2	Xenon	Orbit correction
Pulsed MPD thruster	MS-T4, Japan (1980)	0.24 / 15 / -	Ammonia	Experiment
	Spacelab-1, USA/Japan (1983)	- / 500 / 1	Argon	Experiment
	Space Flyer Unit, Japan, (1994)	23 N pk / 430 / 1	Hydrazine	Experiment
Ion thruster	SERT I, USA (1964)	20 / 1600 / 1	Mercury	Experiment
	Snap 10A, USA (1965)	9 / - / 1	Caesium	Experiment
	ATS 4, USA (1968)	20 – 90 / - / -	Caesium	Experiment
	ATS 5, USA (1969)	-	Caesium	Experiment
	SERT II, USA (1970)	28 / 1000 / 2	Mercury	Experiment
	ATS 6, USA (1974)	4 / 150 / 2	Caesium	Experiment
	ETS III, Japan (1982)	2 / 110 / 2	Mercury	Experiment
	EURECA 1, ESA (1992)	10 / 440 / 1	Xenon	Experiment
	ETS VI, Japan (28/08/ 1994)	23 / 780 / 4	Xenon	NSSK
	ARTEMIS, ESA-ESTEC (1999)	15 / 580 / 4	Xenon	NSSK+ Orbit raising
Ion source	SCATHA P78-2, USA (1979)	0.052 / 60 / 1	Xenon	Experiment
Resistojet	20 flight tests, USA (1965-1971)	0.018– 0.36 / - / -	Nitrogen, Ammonia, Hydrazine	Experiments, Attitude control, Orbit adjust
	11 flights including G-STAR, SPACENET, ASC, SATCOM (1980-1988)	250 / 450 / 2	Hydrazine	NSSK
	INTELSAT V ETS VI, Japan (1993)	320 / 370 / 2 - / 500 / 2	Hydrazine Hydrazine	NSSK NSSK
Arcjet	Telstar 4, USA (Sept. 1994, Ref. 18)	200 / 1400 / 4	Hydrazine	NSSK
	ESSEX, USA (1995)	~2500 / 26000 / 1	Ammonia	Experiment

Different electric propulsion systems used in past missions (Sovey et al. 1992).

Appendix C

C.1 Propellant Characteristics



Alkali metals first ionisation energy (McCarron, 1997).

Element	Atomic mass	Mass (10 ⁻²⁵ kg)	q/m (singly charged) (10 ⁵ C/kg)	Atomic number	1 st ionisation potential (eV) ^a	2 nd ionisation potential (eV) ^a
Caesium (Cs)	132.9	2.21	7.25	55	3.89	25.1
Mercury (Hg)	200.59	3.33	4.80	80	10.44	18.75
Xenon (Xe)	131.293	2.18	7.34	54	12.13	21.21
Krypton (Kr)	83.8	1.39	11.5	36	13.999	24.359
Argon (Ar)	39.948	0.66	24.13	18	15.799	22.629
Rubidium (Rb)	85.4678	1.42	11.28	37	4.16	
Indium (In)	114.818	1.909	8.3915	49	5.79	18.9

Properties for frequently use 'electrostatic' propellants ($q = 1.6 \times 10^{-19}$ C) (Winter, 1993)

Atomic number	37
Atomic weight	85.4678
Melting point	38.89 °C
Boiling point	686 °C
Surface Tension (99.8% purity by wt) - dyne/cm @ 39 °C @ 52 °C @ 477 °C @ 632 °C	77±5 84 55 46.8
Density (grams/cm ³) @ 45.5 °C @ 47.4 °C @ 49.3 °C	1.4505 1.4498 1.4469

Some general properties for the Rubidium propellant.

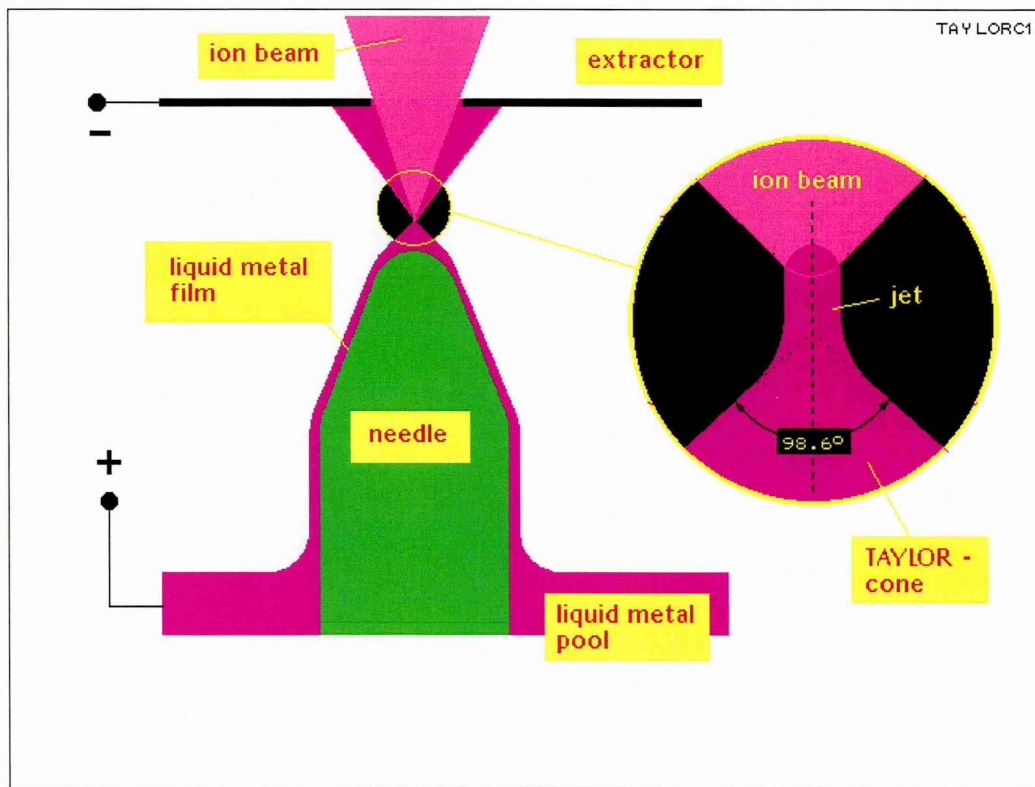
C.2 Caesium Properties

One of the world's richest sources of Cs is located at Bernic Lake, Menitoba. Caesium is silvery-white, soft and ductile alkali metal. It is the most electropositive and most alkaline element. It reacts explosively with cold water, and reacts with ice at temperatures above -116 °C. The 1982 price of Caesium was about US \$ 25 per gram.

State	Crust (solid)
Atomic number	55
Atomic weight	132.9054
Melting point	28.5C
Boiling point	678.4C
Shells	2,8,18,18,8,1
Filling Orbital:	6s1
Electron Configuration:	2-8-18-18-8-1
Surface tension (99.95% purity by wt)-Dyne/cm @ 28.4 °C @ 39 °C @ 494 °C @ 642 °C	73.74 69.5 42.8 34.6
Density (g/cm ³) @ 28.4	1.843
Isobaric molar heat capacity, C_p (J.K ⁻¹ . mol ⁻¹) @ 298.15 K	32.2
Standard molar entropy @298.15 K (J.K ⁻¹ .mol ⁻¹)	85.2

Caesium general properties

C.3 Taylor Cone Model



Taylor cone ideal angle and its 'rounded off' phenomena

C.4 Emitter Material Properties

Metal	Work function ϕ (photoelectric) eV	Melting temperature T_m °C
Ag	4.7	960
C	4.81	3550
Cr	4.37	1890
Ir	4.9	2454
Mo	4.3	2620
Os	4.5	2700
Pt	5.2	1773
Re	5.1	3167
Ta	4.1	3027
W	4.5	3370

Work function and melting temperature of few metals.

Appendix D

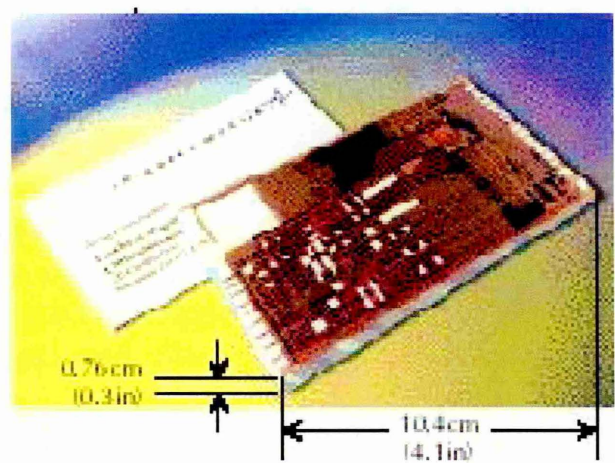
D.1 Small Satellite Power Subsystem

Satellite	Launch	Orbit	Battery	Solar Panel
OSCAR-1 (4.5-kg)	1961	(Lifetime:312 revs)	-	None
UoSAT-1 (60-kg)	1981	554km polar, Sun-synchronous	2 of 5 NiCd cells	4 Body Mounted
UoSAT-2 (65-kg)	1984	679x697 km, 98.5°	2 of 5 NiCd cells	4 Body Mounted
UoSAT-3/4 (45-kg)	1990	790x805 km, 98.7°	2 of 5 NiCd cells	4 Body Mounted
UoSAT-5 (45-kg)	1991		2 of 5 NiCd cells (6Ah/cell)	4 Body Mounted GaAs (18W)
ISES/REX	1991	778x871 km, 89.6°	-	4 Deployed
S80/T	1992	1302x1329km, 66.1°	Total 11 NiCd cells	4 Body Mounted GaAs
KitSat-1 (50-kg)	1992	1301x1402 km, 66.1°	2 of 5 NiCd cells	4 Body Mounted GaAs
KitSat-2 (47.5-kg)	1993	795x805 km, 98.68°	2 of 5 NiCd cells	4 Body Mounted GaAs
PoSAT-1 (47.5-kg)	1993	794x806 km, 98.68°	2 of 5 NiCd cells (6Ah/cell)	4 Body Mounted GaAs
HealthSat-2 (47.5-kg)	1993	793x805 km, 98.68°	2 of 5 NiCd cells (6Ah/cell)	4 Body Mounted GaAs (20W)
Cerise (50-kg)	1995	667x675 km, 98.07°	2 of 5 NiCd cell	4 Body Mounted, Si 4 Deployed, Si
UPM-Sat 1 (47-kg)	1995	665x676 km, 98.07°	2 of 14 NiCd cells (6Ah/cell)	4 Body Mounted (20W) 1xGaAs + 3xSi
Astrid-2 (27-kg)	1995	1000-km circular, 83°	22 NiCd cells (2.3Ah/cell)	4 Deployed, Si (42 W)

Typical Power sources for some small and mini satellites (Wilson, 1996).

D.2 Power Converters

Product specifications from Lockheed Martin Communications and Power Center



PHDI ultra thin power converter

Bus Specifications		
Voltage range	22V to 36V, 50V, 70V, 100V, 120V	
Efficiency	85% to 90% @ full load	
Standby current	0.5 mA maximum	
Input over voltage and survival	120% of V _{nom} steady state over voltage	
Inrush current	Peak < 2A, < 100mA/μs	
Load Specifications		
Output voltages	Output1: 6V to 8V	0mA to 8mA
	Output2: 10V to 15V	10mA to 50mA
	Output3: -5V to -15V	10mA to 50mA
	Output4: 3.3V	10mA to 120mA
Converter output power	40W to 100W	
Ripple voltage	50mV _{p-p}	
General Specifications		
Size	10.5cm x 5.9cm x 0.77cm	
Weight	170grams	
Temperature	-24°C to 61°C acceptance	
Mission life	> 15 years	

PHDI dc/dc power converter specifications (Lockheed Martin Communications and Power Center, 1998)

D.3 EMCO High voltage converters (dc to dc)



Ultra-miniature DC to DC voltage converters

Model	Input voltage	Output voltage	Output current	Ripple	Input current	
					No load	Full load
Q20-5	0V to 5V	0V to 2kV	0.25 mA	< 0.25%	<100mA	<200mA
Q20N-5	0V to 5V	0V to -2kV	0.25 mA	< 0.25%	<100mA	<200mA
Q30-5	0V to 5V	0V to 3kV	0.16 mA	< 0.50%	<100mA	<200mA
Q30N-5	0V to 5V	0V to -3kV	0.16 mA	< 0.50%	<100mA	<200mA
Q40-5	0V to 5V	0V to 4kV	0.125 mA	< 0.50%	<175mA	<300mA
Q40N-5	0V to 5V	0V to -4kV	0.125 mA	< 0.50%	<175mA	<300mA
Q50-5	0V to 5V	0V to 5kV	0.100 mA	< 0.50%	<250mA	<400mA
Q50N-5	0V to 5V	0V to -5kV	0.100 mA	< 0.50%	<250mA	<400mA

Q20 to 50 series characteristics (EMCO High Voltage Corporation Web Site, 2000).

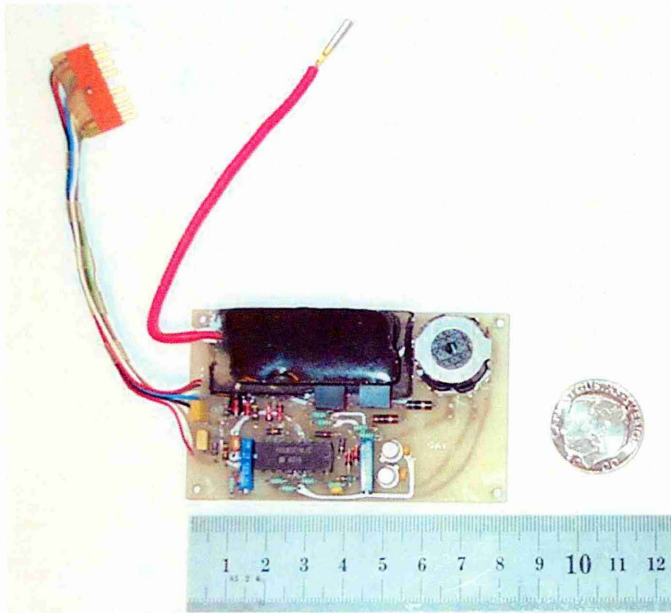
Q series Specifications:

Input voltage	0 to 5, 12 or 24 V
Output voltage (directly proportional to the input voltage, 0.7V to max.)	0 to ± 5 kV
Turn-on voltage	0.7V
Output voltage tolerance 2 full load	+2%, -% typical
Output power	0.5W
Output ripple	<0.5%
Operating temperature	-25°C to +70°C
Size	12.7mm x 12.7mm x 12.7 mm
Weight:	4.25 grams
Case material	Glass filled epoxy
Price of Q50 & Q50N	\$287.60 per unit as at 31 st May 2000

Q50 & Q50N Specifications

D.4 IRF, Sweden voltage converter

Input voltage:	28 V DC
Output voltage	3 kV (Proportionately controllable)
Maximum switching frequency	30 kHz
Mass	50 grams
Size	8cm x 5cm x 2cm
Qualification	Space qualified



Voltage converter from IRF Sweden

D.5 Batteries

D.5.1 Nickel Hydrogen Batteries

Type	Diameter (in.)	Ah Range
IPV	2.5	10-20
IPV	3.5	10-100
IPV	4.5	100-300
IPV	5.5	200-500
CPV	2.5	6-20
CPV	3.5	20-60
CPV	4.5	50-150
CPV	5.5	100-225
SPV	5.0	10-25
SPV	10.0	30-60
SPV	13.0	80-120

Nickel Hydrogen battery specifications from Eagle Picher (Eagle Picher Technologies Web Site, 2000)

Systems Highlights

IPV, CPV, and SPV battery cell and battery design

Independent Pressure Vessel (IPV) is a single cell in a pressure vessel (1.25 nominal voltage).

Common Pressure Vessel (CPV) is, two cells in series in one pressure vessel (2.5 nominal volts).

The Single Pressure Vessel (SPV) is numerous cells (battery) in one pressure vessel (27.5 nominal volts with 22 cells).

Long life, high reliability systems.

Strain gage, state-of-charge monitoring assemblies available.

Excellent overcharge capability.

Environment-friendly product.

Wide ranges of capacities are available.

Cells and batteries are configured to customer specifications or from past proven designs.

ISO-9001 certified.

D.5.2 Lithium Ion Batteries

Part Number	Nameplate Capacity (Ah)	Actual BOL Capacity (Ah) @ C/2, 23°C		
SLC-16002	20	24		
SLC-16016	35	42		
SLC-16020	66	79		
SLC-16022	100	120		
SLC-16024	275	300		
SLB-16002	20	25		
Part Number	Specific Energy (Wh/Kg)	Energy Density (Wh/L)		
SLC-16002	120	259		
SLC-16016	124	283		
SLC-16020	129	303		
SLC-16022	139	308		
SLC-16024	139	308		
SLB-16002	90	175		
Dimensions (inches & grams)				
Part Number	Width	Length	Height	Weight
SLC-16002	0.94	3.00	7.13	720
SLC-16016	1.00	4.63	7.13	1218
SLC-16020	1.06	6.69	8.63	2212
SLC-16022	4.00	3.00	7.13	3120
SLC-16024	3.63	6.69	8.63	7450
SLB-16002	3.5	7.375	8.5	7000

Lithium-Ion Cell family from Eagle Picher.

System Highlights

Flexible prismatic stack and spiral wound designs

Over 50% savings in weight and volume

High current density and stability

Proven cycle life

Nominal Voltage: 3.6 V

Charge Voltage Limit: 4.1

Discharge Voltage Limit: 3.0

Wide operational temperature range -5°C to +30°C

Pulse Capability: 10C @ 100% SOC & 5C @ 20% SOC

D.5.3 Nickel Cadmium batteries

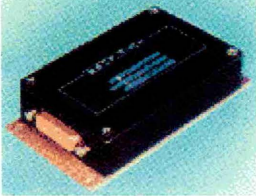
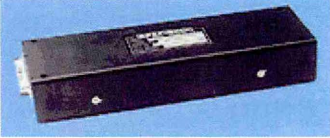
Technical Data								
Model No.	Type	Nominal Voltage (V)	Discharge Capacity ¹		Dimensions (with tube)		Approximate Weight oz (g)	Size
			Average ² Capacity (mAh)	Minimum Capacity (mAh)	Diameter inch (mm)	Height inch (mm)		
P-11AA/FT	N	1.2	120	110	0.57+0/-0.03 (14.5+0/-0.7)	0.69+0/-0.04 (17.5+0/-1.0)	0.23 (6.5)	1/3AA
P-11AAH/FT	H	1.2	120	110	0.57+0/-0.03 (14.5+0/-0.7)	0.69+0/-0.04 (17.5+0/-1.0)	0.23 (6.5)	1/3AA
P-18N/FT	N	1.2	190	180	0.47+0/-0.03 (12.0+0/-0.7)	1.18+0/-0.04 (30.0+0/-1.0)	0.28 (8)	N
P-22AAA	N	1.2	250	220	0.41+0/-0.03 (10.5+0/-0.7)	1.75+0/-0.04 (44.5+0/-1.0)	0.35 (10)	AAA
P-22AAAR/FT	R	1.2	250	220	0.41+0/-0.03 (10.5+0/-0.7)	1.75+0/-0.04 (44.5+0/-1.0)	0.35 (10)	AAA
P-25AAA	N	1.2	280	250	0.41+0/-0.03 (10.5+0/-0.7)	1.75+0/-0.04 (44.5+0/-1.0)	0.35 (10)	AAA
P-25AAAR/FT	R	1.2	280	250	0.41+0/-0.03 (10.5+0/-0.7)	1.75+0/-0.04 (44.5+0/-1.0)	0.35 (10)	AAA
P-30AAR/FT	R	1.2	330	300	0.57+0/-0.03 (14.5+0/-0.7)	1.11+0/-0.04 (28.2+0/-1.0)	0.42 (12)	2/3AA
P-50AAH/FT	H	1.2	580	500	0.57+0/-0.03 (14.5+0/-0.7)	1.90+0/-0.04 (48.3+0/-1.0)	0.74 (21)	AA
P-60AA	N	1.2	640	600	0.57+0/-0.03 (14.5+0/-0.7)	1.97+0/-0.04 (50.0+0/-1.0)	0.74 (21)	AA
P-60AAR/FT	R	1.2	640	600	0.57+0/-0.03 (14.5+0/-0.7)	1.90+0/-0.04 (48.3+0/-1.0)	0.74 (21)	AA
P-60AS	S	1.2	660	600	0.67+0/-0.03 (17.0+0/-0.7)	1.12+0/-0.06 (28.5+0/-1.0)	0.63 (18)	2/3A
P-70AA	N	1.2	740	700	0.57+0/-0.03 (14.5+0/-0.7)	1.97+0/-0.04 (50.0+0/-1.0)	0.78 (22)	AA
P-70AARC/FT	R	1.2	740	700	0.57+0/-0.03 (14.5+0/-0.7)	1.90+0/-0.04 (48.3+0/-1.0)	0.78 (22)	AA
P-80AAS/FT	S	1.2	880	800	0.57+0/-0.03 (14.5+0/-0.7)	1.90+0/-0.04 (48.3+0/-1.0)	0.71 (20)	AA
P-100AASJ	S	1.2	1080	1000	0.57+0/-0.03 (14.5+0/-0.7)	1.97+0/-0.04 (50.0+0/-1.0)	0.81 (23)	AA

Panasonic Nickel Cadmium battery features (Panasonic Web Site, 1998)

Appendix E

E.1 Actuators

E.1.1 MAGNETOMETERS

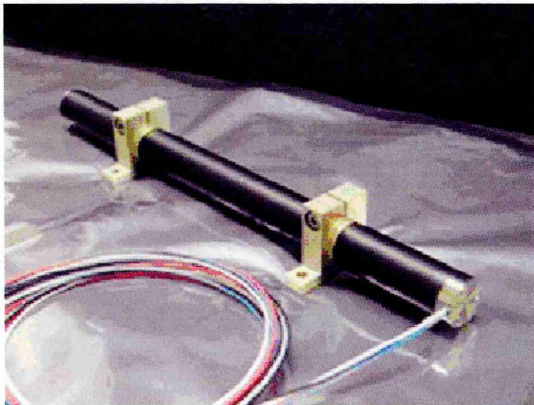
		
IM-102 Two-Axis Magnetometer	IM-103 Three-Axis Magnetometer	IM-203 Three-Axis Magnetometer

IM-Series Magnetometers (Golden S. & Associates , 1997).

	IM-102	IM-103	IM-203
Number of Axes	Two orthogonal	Three orthogonal	Three orthogonal
Orthogonality (* After calibration.)	0.5° *	0.5° *	0.5° *
Field Measurement Range	±600 mG	±600 mG	±1000 mG
Scale Factor	1.5 V/G ±5%	8.3 V/G	8.3 V/G
Zero Field Bias	<20 mG	<10 mG	<10 mG
RMS Noise (0.1 to 80 Hz)	<50 µG	<12 µG	<12 µG
Frequency Response	-3 dB @ >200 Hz	-3 dB @ >100 Hz	-3 dB @ >80 Hz
Supply Voltage	+5 ± 0.25 V	±15 V ± 5%	28 ± 6 V
Power Consumption	<50 mW (0 mG) <100 mW (600 mG)	<0.8 mW (0 mG) <1.0 mW (600 mG)	<1.4 mW (0 mG) <1.7 mW (600 mG)
Operating Temp.	-40°C to +70°C	-40°C to +80°C	-25°C to +60°C
Non-Operating Temp	-40°C to +80°C	-50°C to +90°C	-25°C to +65°C
Dimensions	11.4 x 5.8 x 2.5 cm (4.5" x 2.3" x 1.0")	5.5 x 4.2 x 3.6 cm (6.1" x 1.7" x 1.4")	20.3 x 7.9 x 3.6 cm (8.0" x 3.1" x 1.4")
Mass	<220 g	<227 g	<635 g
Temperature Output	N/A	40 mV/K	N/A

IM-Series Magnetometer performance summary

E.1.2 Magnetic Torque Rods



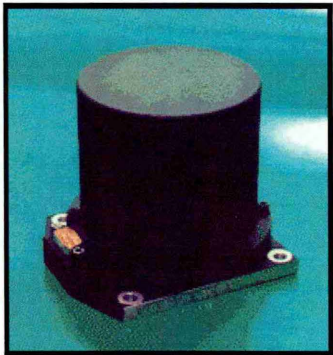
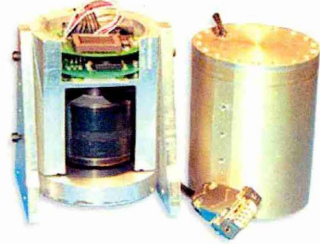
E.1.3 Off-the-shelf torque rods

Model	Linear Dipole Moment [mA ²]	Saturation Moment [mA ²]	Length [mm]	Diameter [mm]	Mass [kg]	Power [W]	Voltage [V]	Status
MT-5-2	5	6.5	240	18	0.3	0.85	5	Delivered
MT-6-1	6	8.5	325	15	0.23	0.25	5	Delivered
MT-6-2	6	8.5	322	15	0.312	0.36	5	Delivered
MT-10-2	10	13	350	18	0.5	0.8	5	Design Study
MT-15-1	15	20	330	17	0.43	1.11	14	Delivered
TR1UPN	1.1	1.13	127	12.7	0.10	0.18	5.6	Delivered (Without Case)

Off-the-shelf torque rods for possible nano-satellite applications

E.1.4 Momentum Wheels

Reaction wheels are essentially torque motors with high-inertia rotors. They can spin in either direction, and provide one axis of control for each wheel. Momentum wheels are reaction wheels with a nominal spin rate above zero to provide a nearly constant angular momentum. This momentum provides gyroscopic stiffness to two axes, while the motor torque may be controlled to precisely point around the third axis.

Model	Characteristics
<p>Dynacon MicroWheel 200</p> 	<p>Size: 102 x 94 x 89 mm Mass: 0.77 to 0.94 kg Speed range: ± 10000 RPM Angular momentum capacity: 0.05 to 0.18 Nms Torque capacity: 30mN-m Power consumption (@ 20C): 0 RPM: 0.6W 5000 RPM: <2.1W 9000 RPM: <3.2W +1W for optional rate sensor Manufacturer: Dynacon Enterprises Limited, Canada Contact: Kieran A. Carroll, Phone: 905-672-8828 x 232 E-mail: kac@dynacon.ca URL: http://www.dynacon.ca</p>
<p>Bantam reaction wheel</p> 	<p>Size: 107 x 76 mm Mass: 1.4 kg Speed range: $\pm 30,000$ RPM Angular momentum capacity: 0.5693 Nms Torque capacity: 5.6 μN-m Power consumption: 13$\pm$2Vdc 20mA to 2A 250 mA typical Manufacturer: One Stop Satellite Solutions 1805 University Circle, Ogden, UT 84408-1805 Phone: (801) 626-7272 Fax: (801) 626-7951 E-Mail: info@osss.com URL: http://www.osss.com/</p>

Appendix F

F.1 Attitude Control

Sensor	Accuracy (deg)	Mass (kg)	Power (W)
Sun Sensors	0.005 - 3	0.05 - 2	0 - 3
Earth (Horizon) Sensors:	- 0.5	0.05 - 1	1
Pulse Generators	0.5 - 3	- 10	0.5 - 14
Passive Scanners	0.05 - 0.25	3 - 8	7 - 11
Active Scanners			
Star Sensors	0.0003 - 0.1	1.5 - 10	1.5 - 20
Magnetic Field Sensors	0.5 - 5	0.6 - 2	0.5 - 2
GPS	~ 0.1	2 - 10	15

Attitude Sensors for Spacecraft (Page 193, Sidi, 1997)

F.2 Case Studies

Satellite	Weight & Size	Payload (Instruments)	Power	Launch Date
ASUSAT-1	5 kg	GPS, Camera, FM repeater	6-8 W continuous	1998
Bitsy	1 kg		Unregulated 8V power bus 4Whr rechargeable lithium-ion battery	
Munin	5.5 kg: Cubic in shape, 200mm x200mm x250mm	A Spectrometer	15V-25V range	October 1999
PICOSAT (P97-1)	67 kg	PBex, IOX, CERTO, OPPEX	22 W	30/09/2001
SNAP-1	7 kg	GPS, Cold gas thrusters		28/06/2000
PixelSat	25mm diameter discs	Discs with only 12 transistors		

Current and future missions with nano and pico satellites.

Pbex: Polymer Battery Experiment
 IOX: Ionospheric Occultation Experiment
 CERTO: Coherent Electromagnetic Radio Tomography
 OPPEX: Optical Precision Platform Experiment

F.3 Sensors

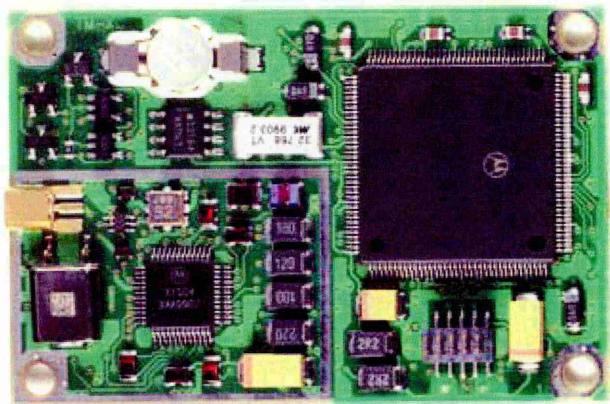
Model	FOV per sensor	Max. No. of sensors	Least significant bit size	Accuracy	Transfer function	Output	Size (mm) Electronics / Sensor	Weight (g) Electronics / Sensor	Input power	Design status	Remarks
Digital systems for spinning vehicles											
15564	128°	1	1.0°	0.50°	A	7-bit parallel Gray TTL	43 x 33 x 66	145	+5.13vdc 2.0ma/ 6.12 vdc 2.5 ma	Flown ESRO IV	Sensor and electronics in one unit
17083	180°	1	0.5°	0.25°	A	9-bit parallel Gray TTL	74 x 66 x 97	368	+12±0.36 vdc 6 ma / -12±0.36 vdc 1ma	Flown NRL	Sensor and electronics in one unit
17212	128°	1	1.0° coarse Analog fine	0.25°	B	7-bit parallel Gray TTL Two analog	64x58 x37 / 66x33x25	172 / 104	+15vdc 4ma -15vdc 4ma	USAF	
Two axis digital systems											
15486	128°x128°	1	1.0°	0.5°	D	7 bit/ axis parallel Gray 0-10V	76x76x51 / 84x41x1	358/ 82	+15vdc 19.5 ma	Flown ATS-6	

							8				
Two axis analogue systems											
18560	30° cone	1	N.A.	1' at null	±1° linear	±5V	56x36x3 1	191	+15vdc -15vdc 100mw	Designed NASA Lewis	
18394	180 solid angle	1	NA	2° at null	±30° linear	±0.1 ma peak	NA / 48x48x3 3	None / 82	None	Flown OAO-B, C ATS-6	Cluster of five 11866 cosine-law sensors
Single axis digital systems											
17061	100x100	1	0.006°	0.05°	E	14-bit serial Natural binary; open collector	102x86x51 / 109x64x28	517/ 322	+5vdc 65 mw / +28vdc 700mw	USAF Block V	
Single axis analogue sensors											
17470	40° x 60°	1	NA	6' at null	±1° linear	0-5 V 2.5 V at null	69x51x2 8	118	±15 ±0.6 vdc 100mw	Qualified CTS	Used on CTS for solar array pointing

Catalogue of sun-angle sensor systems (Table B.3.1;(Sidi, 1997)).

F.4 GPS

In the commercial market there are many micro-GPS receivers available. Though they are limited mainly to the ‘terrestrial’ market, it is possible to upgrade them for the required space application. However some degradation in function is a possibility. Here the ‘M-12 Oncore’ GPS Module is highlighted in order to understand the current technology.



‘M12 Oncore’ GPS Module

Receiver Architecture	12 parallel channel L1 1575.42 MHz C/A code (1.023 MHz chip rate) Code plus carrier tracking (carrier aided tracking)
Tracking Capability	12 simultaneous satellites
Dynamics	Velocity: 515 m/s (1000 knots);>515 m/s at altitudes< 18km Acceleration: 4g Jerk: 5 m/s ³ Vibration: 7.7G per Military Standard 810E
Acquisition Time (Time To First Fix, TTFF) (Tested at –30 to +85°C)	<15 sec typical TTFF-Hot <40 sec typical TTFF-warm <60 sec typical TTFF-cold (No stored information) <1.0 sec internal reacquisition (typical)
Position Accuracy	100 meters 2dRMS with SA as per DoD specification Less than 25 meters, SEP without SA
Timing Accuracy (1PPS) Antenna	<500ns with SA on Active micro strip patch Antenna Module Powered by Receiver Module (15mA@3Vdc) (optional 5Vdc available)
Datum	WGS-84 One user definable datum
Output Messages	Latitude, longitude, height, velocity, heading, time Motorola binary protocol at 9600baud NMEA 0183 at 4800 baud (GGA, GLL, GSA, GSV, RMC, VTG, ZDA)

	Software selectable output rate (continuous or poll) TTL Interface (0 to 3V) Second COM port for RTCM input
Power Requirements	2.75 to 3.2 Vdc; 50 mV p-p ripple (max.)
“Keep-Alive” BATT Power	External 2.75 Vdc to 3.2 Vdc, 5 μ A (typical @ 2.7Vdc)
Power Consumption	<0.225W@3V without antenna
Dimensions	40 x 60 x 10 mm
Weight	Receiver 25 g Active Antenna module < 40g
Connectors	Power/Data:10pin (2x5) unshrouded header on 0.05in centers. RF: Right Angle MMCX (Subminiature Snap-on)
Antenna to Receiver Interconnection	Single coaxial cable with 6db maximum loss at L1 (active antenna) Antenna Sense Circuit
Operating Temperature	-40° to +85°C
Humidity	95%noncondensing +30°C to +60°C
Altitude	18km maximum >18km for velocities <515 m/s
Standard features	Motorola DGPS corrections at 9600 baud on COM port-1 RTCM SC-104 input Type 1 and type 9 messages for DGPS at 2400, 4800 or 9600 baud on COM port-2 NMEA 0138 output Inverse DGPS support
Optional features	Lithium battery backup

M12 Oncore Characteristics (Motorola Products Web Site, 1999)

Appendix G

G.1 Trade Studies

	FEEP	Ion (Not suitable for tiny thrust)	SPT (Not suitable for tiny thrust)	Resistojet
Specific Impulse (sec)	8000	5000	3000	300
Thruster lifetime (hrs)	> 2000*	> 15,000	> 7,000	> 390
Thruster efficiency (%)	98	75	48	90
Overall efficiency (%)	80	61	39	74
Thruster specific power (kW/N)	49	40	38	2
Power per 10^{-4} N (W)	4.9	4.0	3.8	0.2
Solar array area (to generate above power) with 15% eff. (cm ²)	241 (Ga-As)	197 (Ga-As)	187 (Ga-As)	10 (Ga-As)
Solar array mass: (40 W/kg) (kg)	0.1225	0.1	0.095	0.005
Battery capacity (W-hrs)	2.94	2.4	2.28	0.12
Battery mass (kg)	0.05	0.04	0.04	0.002
Power control mass (kg)	0.098	0.08	0.076	0.004
Regulators/ converters (kg)	0.1225	0.1	0.095	0.005
Wiring (kg)	0.2	0.2	0.2	0.2
Nozzle (kg)	0	0		0.2
Propellant for 1 year mission (kg)	0.013	0.021	0.034	0.335
Propellant phase Tank Vol. (cm ³) Storage tank (kg)	Liquid / Ions 9.2 0.006	Liquid / gas / Ions 14.6 0.008	Liquid / Plasma 24.4 0.011	Liquid / gas 240.1 0.05
Propellant feeding (kg)	0.01	0.01	0.01	0.01
Neutralizer (kg)	0.1	N/A	-	-

Mass breakdown structure for 4 thrusters (* Not space qualified at the time of writing)

G.2 Ion Engine

Input Power (W)	Specific Impulse (s)	Thrust (mN)	Efficiency (%)	Beam Current (mA)
85	1810	3,6	37,2	88,2
99	2030	4	40,1	88,2
113	2230	4,4	42,2	88,2
128	2410	4,8	43,9	88,2
138	2180	5,6	43,2	123
158	2390	6,1	45,5	123
177	2590	6,6	47,2	123
187	2680	6,8	48	123
207	2510	8,1	47,9	162
233	2720	8,7	49,7	162
246	2820	9	50,5	162
286	2860	10,9	53,6	204
303	2960	11,3	54,3	204

8cm ion engine performance (Patterson and Oleson, 1997a), (table-1)

Input Power (W)	Specific impulse (s)	Thrust (mN)	Efficiency (%)	Beam Current (mA)
50	2300	2,2	49	44
80	2700	3,4	57	68
116	3100	4,7	61	81

5cm ion engine performance (Gorshkov and Muravlev, 1999)

G.3 FEEP thrusters

Propulsion system type	Slit emitter FEEP	Pin emitter Indium
Propellant	Caesium	Indium
Thruster operation	Continuous	Continuous
Thrust range [μN]	0.1 – 1200	1 – 100
Thrust noise [$\mu\text{N}/\text{Hz}^{-0.5}$]	0.1	< 0.1
Specific Impulse [s]	7000 - 11000	10000
Minimum impulse bit [N.s]	1×10^{-9}	$< 10 \times 10^{-9}$
Thrust to power ratio [$\mu\text{N}/\text{W}$]	16 - 20	15
Mass	~2.2	~2.5

FEEP Caesium and Indium propulsion system properties (Leach and Kerry L. N., 2003)

Appendix H

H.1 Spacecraft Separation Systems

Satellites require separation systems to accomplish release from the launcher, separation from each other and the separation adapter to hold them in place during launch. The traditional pyrotechnic release devices produce high shock, contamination and have a costly handling requirement due to their hazardous nature. Nano-satellites are particularly susceptible to shock and vibration related malfunction because of the proximity of sensors and instruments to the shock source. In addition piggyback and multi-satellite launch missions will experience multiple release shocks from adjoining satellites prior to their own separation. With the renewed interest in nano- and pico- satellites, low shock separation devices are of interest (Peffer, 2000). The major development in the low shock separation system research is the study of Shape Memory Alloys to use as a separation actuator. There are two SMA based released devices, known as the Low Force Nut (LFN) and Two Stage Nut (TSN), which have been built and tested, both in laboratory and on-orbit (Peffer, 2000).

H.1.1 Low shock separation devices goals:

- Develop separation devices which reduce shock levels at least one order of magnitude (< 500 g)
 - Pyro device shock is typically 5000 – 7000 g range
- Develop non-pyrotechnic release device
 - Mitigates safety and contamination concerns
- Comparable to pyrotechnic
 - Size, mass and required launch vehicle firing pulse
- Develop resettable device
 - Allows tests of separation systems
- The device that is tested is the device that is flown

H.1.2 Shape Memory Alloy (SMA) programs with Lockheed Martin and Starsys Research

- Lockheed Martin to develop SMA release devices
 - Low Force Nut (LFN) and Two Stage Nut (TSN)
 - LFN and TSN are not pyrotechnic firing pulse compatible
 - Devices meet all other requirements
- Starsys Research Corp. to develop release devices (Phase-I SBIR)
 - Redesign LFN to be pro-compatible
 - Improve QWKNUT design
 - Flight level testing of QWKNUT
 - Develop nano-satellite separation concepts
- Starsys Phase-II SBIR
 - Develop SMA clamp band (Marmon band) technology

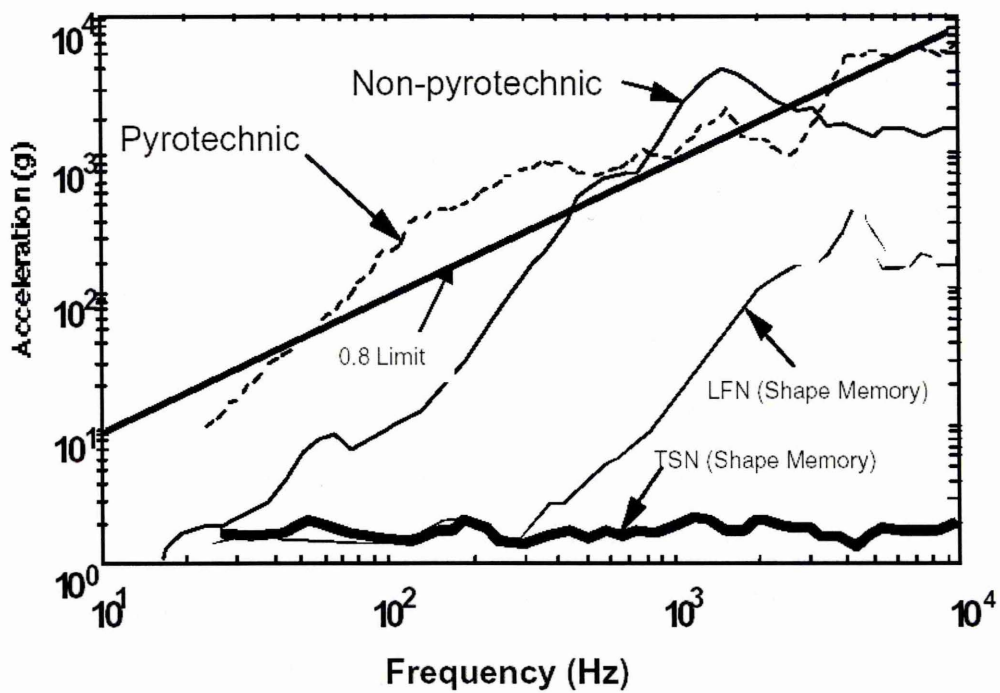


Table H-1 Shock levels with different separation systems



QWKNUT
(STARSYS Research)



LFN
(Lockheed Martin)



TSN
(Lockheed Martin)

	Pyrotechnic	LFN	TSN	QWKNUT
Preload, (kN)	39.1	12.7	24.5	13.3
Shock, (g)	7,200	400	< 150	< 100
Mass, (grams)	120	250	300	200
Release Time, (ms)	15	62	22	35
Reset (YES/NO)	N	Y	Y	Y
Pyro Pulse	-	N	N	Y

Table H-2 Characteristics of 0.635 cm bolt spacecraft separation devices (Peffer, 2000)

Notes:

SMA: Shape Memory Alloy

LFN: Low Force Nut

TSN: Two-Stage Nut

QWKNUT: This is a low shock discrete pint device that like the LFN releases a 1/4 inch bolt.

'Y' entry indicates compatibility with typical launch vehicle 'pyro-pulse'

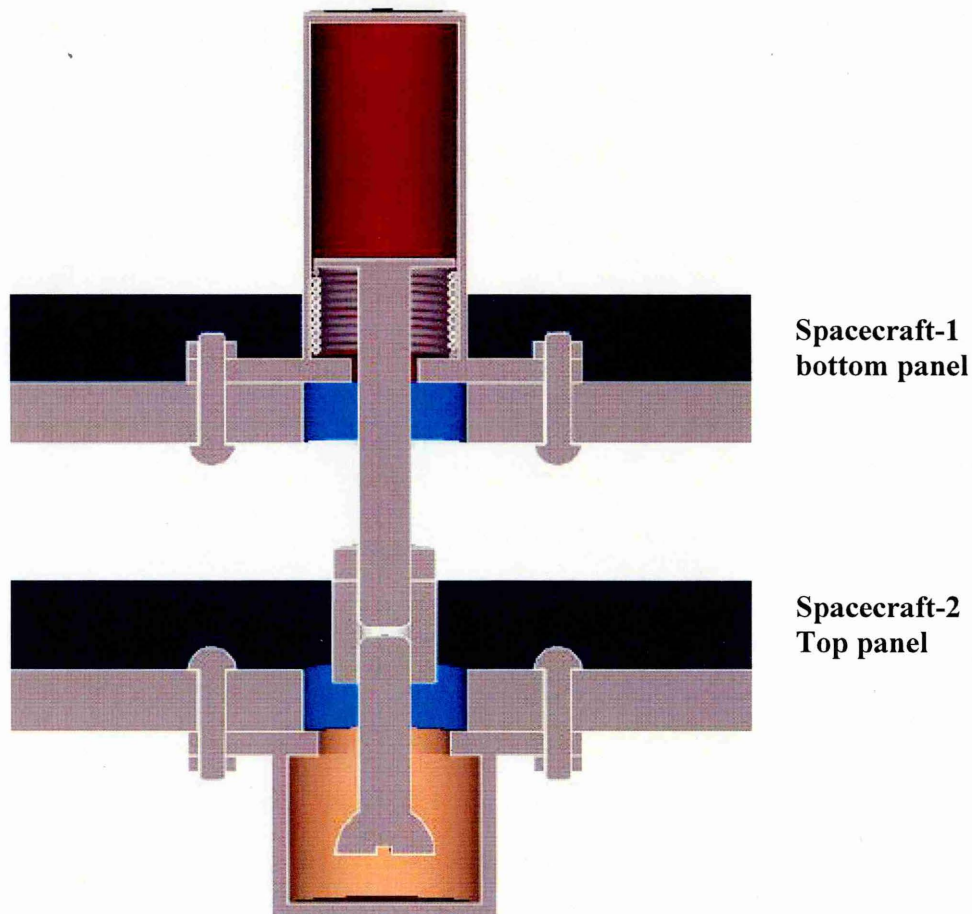


Figure H-1 Inter-satellite separation actuator (Peffer, 2000)

H.1.3 Fuse link low shock devices

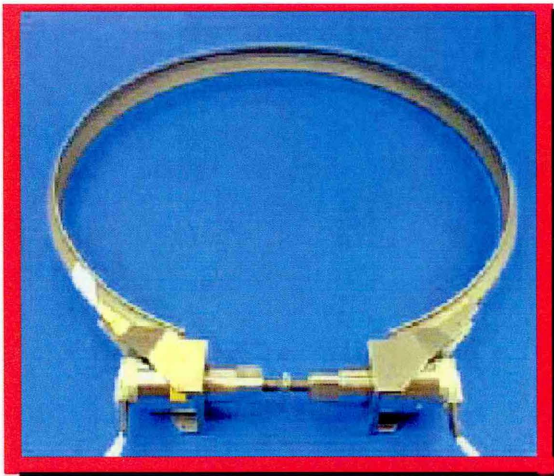


Figure H-2 NEA V-band release device

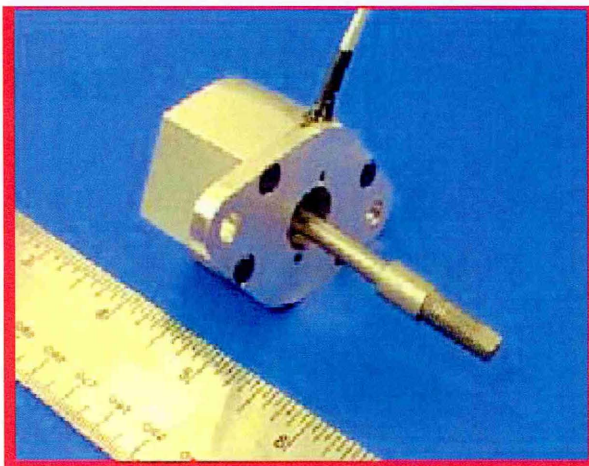


Figure H-3 NEA model 9101B release device (Peffer, 2000)

H.1.4 Clamp band system overview for spacecraft stacking

