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Miniature Wire Boom System for Cubsat Application

Keith R. Bradford
Utah State University

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MINIATURE WIRE BOOM SYSTEM FOR CUBESAT APPLICATION

by

Keith R. Bradford

A report submitted in partial fulfillment of the requirements for the degree of

MASTER OF SCIENCE

in

Mechanical Engineering

Approved:

Dr. R. Rees Fullmer
Major Professor

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Committee Member

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Committee Member

UTAH STATE UNIVERSITY
Logan, Utah

2013
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Abstract

Miniature Wire Boom Deployment System for CubeSat Application

by

Keith R. Bradford, Master of Science
Utah State University, 2013

Major Professor: Dr. R. Rees Fullmer
Department: Mechanical and Aerospace Engineering

Small satellites and especially CubeSats are becoming more widely used to study the space environment. The Ionosphere is one region of particular interest, more specifically the altitude region of 85 km to 600 km. Small satellites are particularly useful for studying this region of the Earth’s atmosphere since the effects of aerodynamic drag on a CubeSat are much less than those on a larger more traditional satellite, thus the lifespan of a CubeSat in this region is much longer. In order to observe the electric field in space, the electric potential between various points needs to be measured. These measurements are most effectively taken when the sensors are located several meters from one another. A deployment mechanism is needed in order to position the sensors at these distances. A miniature wire boom deployment system was developed by Utah State University and the Space Dynamics Laboratory in Logan, Utah which accomplishes this task. The deployment system is capable of deploying sensors up to 5 meters in 4 directions using a piezoelectric motor controlled mechanism. This system conforms to all CubeSat specifications and is modular so it can be integrated into any CubeSat application. Recently this miniature wire boom deployment system was integrated into the two satellites of the DICE program.
Acknowledgements

Anyone who knows me knows that writing, any kind of writing, is one of the most painful tasks that anyone could ask of me. That being said it has been a pleasure and a once in a lifetime experience to work on this project and on the DICE project. It has been one of immense learning, hard work, great friendship, long hours, and at times delicious take-out food. I am beyond words grateful for the time and opportunity to work on this project and with the good people on the DICE team and at the Space Dynamics Laboratory. Specifically I would like to thank Dr. Swenson for involving me on this project from the very beginning and for being such a great teacher and mentor. Chad Fish for his leadership and knowledge and his patience with a bunch of nerdy engineering students but most of all for his ability to teach those students how to be real engineers. I would also like to thank the friends who worked so closely on this project with me namely, Josh Martineau, Steven Grover, Mark Anderson, Steven Burr, Erik Stromberg, Tim Neilson and others on the DICE team both past and present. Thanks to Robert Lowe and the other SDL machine shop crew for their fine work and patience for working on such small and complex parts. Thanks to the other incredibly helpful and resourceful SDL staff that contributed to this project.

I would also like to thank my family members who have been a source of encouragement and motivation. A special thanks to my loving wife Heather and to our children Carsen and Hailey who have been incredibly patient, loving, and supportive throughout this project. Without the time they sacrificed I never could have complete this project…this is for you!

Finally I would like to dedicate this report to my Mother, Marsha Lynn Hebertson Bradford, who lost her brave and inspiring fight with breast cancer on March 5, 2013. Your encouragement and optimism was infectious, thanks for believing in me.

-Keith Richard Bradford
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<th>Acronym</th>
<th>Description</th>
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<tr>
<td>ADCS</td>
<td>Attitude Determination &amp; Control System</td>
</tr>
<tr>
<td>ASSP</td>
<td>Auroral Spatial Structures Probe</td>
</tr>
<tr>
<td>AWG</td>
<td>American Wire Gage</td>
</tr>
<tr>
<td>CMM</td>
<td>Coordinate Measurement Machine</td>
</tr>
<tr>
<td>DCP</td>
<td>Direct Current/Langmuir Probe</td>
</tr>
<tr>
<td>DICE</td>
<td>Dynamic Ionosphere CubeSat Experiment</td>
</tr>
<tr>
<td>EDM</td>
<td>Electrical Discharge Machining</td>
</tr>
<tr>
<td>EFP</td>
<td>Electric Field Probe</td>
</tr>
<tr>
<td>EMI</td>
<td>Electromagnetic Interference</td>
</tr>
<tr>
<td>FPGA</td>
<td>Field Programmable Gate Array</td>
</tr>
<tr>
<td>MOI</td>
<td>Moment Of Inertia</td>
</tr>
<tr>
<td>MWBS</td>
<td>Miniature Wire Boom System</td>
</tr>
<tr>
<td>NASA</td>
<td>National Aeronautics &amp; Space Administration</td>
</tr>
<tr>
<td>NOVA</td>
<td>NanoSat Operation Verification &amp; Assessment</td>
</tr>
<tr>
<td>NPP</td>
<td>NPOESS Preparatory Project</td>
</tr>
<tr>
<td>PCB</td>
<td>Printed Circuit Board</td>
</tr>
<tr>
<td>PET</td>
<td>Polyethylene Terephthalate</td>
</tr>
<tr>
<td>SED</td>
<td>Storm Enhanced Density</td>
</tr>
<tr>
<td>TAM</td>
<td>Three Axis Magnetometer</td>
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<tr>
<td>USU</td>
<td>Utah State University</td>
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Chapter 1

Introduction

For more than a decade the CubeSat has been internationally used as a teaching tool for students pursuing careers in space engineering and research. Student built CubeSats have been flown by countries such as the United States, Canada, Denmark, Switzerland, Japan, Israel, and Romania. Government organizations such as the United States National Science Foundation, NASA and the Department of Defense have invested to promote CubeSat technology by directly funding projects and by providing launch opportunities. Companies have also invested in CubeSat technology because their value for getting small experiments and payloads into space quickly has been demonstrated. Low power electronics technologies, driven by the consumer portable electronics market, have made it possible to construct highly capable CubeSats by making use of components created for smart cell phones and handheld computers. These technologies enable the development of sophisticated satellites despite the restrictions of small mass, power and volume of CubeSats.

The first CubeSats carried simple payloads such as beacons or transceivers, for communication and tracking, more recent missions have research payloads to increase our understanding in areas of research like pharmacology, space weather, micro-gravity, and space engineering. New CubeSats are being used to test advanced satellite technologies such as state-of-the-art solar energy cells and attitude control and guidance systems. The primary reason these experiments are being performed on CubeSats because the total cost of getting into space is considerably lower for a secondary payload such as a CubeSat than for a traditional satellite even though the cost per kg may be similar. Smaller in this case is cheaper.

1.1 CubeSats in Space Weather Research

CubeSat were initiated in 1999 by Stanford University and The California Polytechnic State University (Cal Poly). The essence of the CubeSat is to develop a standard size and volume container for carrying nano-satellite into space that can be attached to any launch vehicle without impacting the primary mission. The secondary payload’s ride into space is low cost and is therefore expected to be available more often. The original motivation for the CubeSat was to promote research and development in space engineering at
the education level. The resulting CubeSat standard is becoming more and more popular because of reduced time and costs of satellite development. CubeSats enable more access to space, which is an encouraging sign for the community dedicated to space research missions.

CubeSats can be further classified based on their form factor. A CubeSat with 1U form factor (meaning one unit) refers to structural dimension of 10x10x10 cm. A 1U CubeSat generally weighs around 1 kg. Other commonly used form factors for CubeSats are 1.5U (10x10x15 cm), 2U (10x10x20 cm) and 3U (10x10x30 cm). Since CubeSats are all 10x10 cm in cross-section (regardless of height) they can all be launched and deployed using common deployment systems with minimal alteration between missions.

The United States’ National Science Foundation has the CubeSat-based Science Missions for Space Weather and Atmospheric Research program which aims to support the development, construction, launch, operation, and data analysis of small satellite science missions to advance space weather and atmospheric research [1]. Started in 2008 within the Division of Atmospheric and Geospace Sciences, this program has funded eight CubeSat based projects for space weather research. Table 1.1 gives a brief summary of these NSF funded projects.

One of the compelling applications of the CubeSat is the deployment of large numbers of CubeSats in a LEO constellation missions to address fundamental questions on space weather or to provide multipoint monitoring capability of earth space.

1.1.1 Space Weather and Electric Fields

Understanding Space weather requires understanding the motion or winds of the thin gasses in the upper atmosphere of the Earth. Electric field measurements in the Earth’s ionosphere are one way of observing this motion because any bulk motion of the ionospheric plasma is accompanied by an electric field. An instrument called the Electric Field double Probe (EFP) is commonly used to carry out these electric field measurements in the space environment. The electric field at a point in space is equal to the negative gradient of the electric potential. The EFP estimates this through a pair of conducting spheres immersed in the ionospheric plasma and separated by several meters distance. The potential difference between the conducting spheres is measured and then divided by the separation distance to give the component of electric field vector along the direction of the boom [2]. Within the earth’s ionosphere the conductivity along the magnetic field is very high which essentially shorts the electric field in that
direction. Crossed booms are typically all that is needed to observe the electric field perpendicular to the Earth magnetic field. The measurements are best made from spinning spacecraft such that errors and offsets in the technique can be identified and removed from the data. Previous mission have either used ridged booms or have attached sensors at the ends of the wire booms that are held in position by the centrifugal force on spinning spacecraft. Use of wire boom offers an ultra-lightweight design for long separation distances between EFP sensors.

Table 1.1 List of NSF Funded Projects

<table>
<thead>
<tr>
<th>Mission</th>
<th>Objective</th>
<th>University/Institution(s)</th>
<th>CubeSat</th>
</tr>
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<tr>
<td>RAX</td>
<td>Measure small scale plasma density irregularities in the ionosphere.</td>
<td>University of Michigan and SRI International</td>
<td>3U CubeSat</td>
</tr>
<tr>
<td>Firefly</td>
<td>Explore Causal links between ground lighting and terrestrial γ-ray flashes.</td>
<td>Siena College and NASA Goddard Space Flight Center</td>
<td>3U CubeSat</td>
</tr>
<tr>
<td>FIREBIRD</td>
<td>Investigate size, persistence, and energy dependence of relativistic electron bursts from inner radiation belts.</td>
<td>University of New Hampshire and Montana State University</td>
<td>Two 1.5 U CubeSats</td>
</tr>
<tr>
<td>DICE</td>
<td>Measure ionosphere density and electric field variability with the formation of geomagnetic storm.</td>
<td>Utah State University and Astra</td>
<td>Two 1.5 U CubeSats</td>
</tr>
<tr>
<td>CINEMA</td>
<td>Map strong geomagnetic currents and energetic neutral atoms associate with storm time precipitation</td>
<td>UC Berkeley(lead), Imperial College London, NASA Ames, Kyung Hee University</td>
<td>3U CubeSat</td>
</tr>
<tr>
<td>CSSWE</td>
<td>Measure energetics of solar produced relativistic electrons and protons during periods of intense solar flare activity.</td>
<td>University of Colorado at Boulder</td>
<td>3U CubeSat</td>
</tr>
<tr>
<td>CADRE</td>
<td>Measure density and composition of perturbed thermosphere using a novel sensor.</td>
<td>University of Michigan</td>
<td>1~1.5U CubeSat</td>
</tr>
<tr>
<td>EXOCUBE</td>
<td>Measure density of select species of neutral and ionized atoms in uppermost levels of the Earth’s atmosphere.</td>
<td>University of Wisconsin, Cal Poly, and Scientific Solutions Inc.</td>
<td>3U CubeSat</td>
</tr>
</tbody>
</table>
1.1.2 **Wire Boom Systems**

The study of wire booms and other flexible bodies for use on a spacecraft began in the early 1970’s by NASA scientists. Studies were conducted, in part, to explain the tumbling motion of the Explorer 1 satellite which was equipped with flexible antennas. The Explorer 1 spacecraft was intended to spin along its long axis while in orbit but settled into a precessing rotation instead due to energy dissipation in the flexible antennas [3].

Even as early as the 1970’s the United States Air Force (USAF) was investigating the use of the double probe measurement technique to measure the electric field in the ionosphere [4].

Wire boom deployment systems have been used on several spacecraft missions in the recent past and they all have some fundamental similarities. The systems provide a mechanism to deploy weighted sensors at the ends of flexible wires from spinning spacecraft. They basically differ in the length of the wire booms used and the mechanism adopted to deploy them. For example the electric field Instrument for THEMIS [5] and the FAST Satellite [6] made use of motors within the wire boom mechanism to actively control the deployment of the sensors. Similar deployment is being adopted for BepiColombo mission to the magnetosphere of Mercury to deploy its MEFISTO-S and WPT sensors to a length of 15 meters [7] [8].

Sounding Rocket missions have also made use of wire boom systems. Cornell University has developed the SIERRA wire boom system [9]. This system is similar to the yo-yo de-spin system that is used on sounding rockets that consists of a wire wrapped around the body of the rocket with a weight that when released slows the spin rate of the vehicle. The SIERRA system uses a rotary damper to avoid wire re-wrap around the spacecraft after the booms have been deployed yo-yo style from the sounding rocket.

1.2 **DICE Mission**

The Dynamic Ionosphere CubeSat Experiment (DICE) mission was selected and funded by the National Science Foundation in October 2009 in response to a cooperative proposal from Utah State University’s Space Dynamics Laboratory (USU/SDL), ASTRA Inc., and Embry Riddle University. DICE is one of several missions developed under NSF’s CubeSat-based Science Mission for Space Weather and Atmospheric Research program. Variations in the Ionosphere’s plasma density affects radio frequency based systems like communication systems, surveillance and navigation systems on earth and in space. Therefore, it becomes highly important to conduct studies on geomagnetic storm enhanced density (SED)
features that occur in the Earth’s Ionosphere. The DICE mission is headed to investigate the relationship between penetration electric fields and the formation and evolution of SED. DICE consists of the two CubeSats which are identical in design and function.

Students working with professionals at the Utah State University Space Dynamics Laboratory spearheaded the design, fabrication and testing of the CubeSats. The launch of DICE occurred on October 28th, 2011 from Vandenberg Air Force Base in California at 2:48 a.m. local time. Both DICE spacecraft were inserted jointly into orbit from a P-POD carried on Delta II rocket for the Suomi National Polar-orbiting Partnership spacecraft (NPP). The DICE CubeSats will be aligned with the Earth’s geodetic axis and spun up for stabilization. After sufficient time has elapsed to ensure safe inter-spacecraft separation, the wire booms of the EFP will be deployed using a novel miniature wire booms deployment system that involves controlled rate of wire boom deployment.

1.2.1 Overview of DICE CubeSat’s Systems

Each DICE CubeSat conforms to a 1.5U form factor (10x10x15 cm) and can be roughly divided into payload, electronics, communication, and attitude control sub-systems. Electrical power is provided by solar panels attached to the outer faces of the CubeSat. When the CubeSat is in eclipse, power stored in a high energy density lithium polymer battery will be used which afterwards get recharged through the solar panels. A Pumpkin FM430 flight control module containing a Texas Instruments MPS430 microcontroller provides computing. Communication is provided by a Cadet-U modem developed by L3 Communications providing a 2.6 Mbit/s downlink in the 460 to 470 MHz band and 19.2 kbit/s uplink at 450 MHz [10]. Torque coils provide attitude control while the attitude determination system consists of the sun sensors, a magnetometer and a miniature GPS receiver. The DICE CubeSat’s systems are shown in Figure 1.1.
Each CubeSat has three science instruments, a Langmuir Probe (DCP) to measure in-situ ionospheric plasma densities, an Electric Field Probe (EFP) to measure DC and AC electric fields and a Three Axis Magnetometer (TAM) to measure field-aligned currents. The four EFP booms each extend 5 m from the spacecraft with spheres on the ends of the booms.

1.2.2 Wire Boom Deployment System for DICE

The miniature wire boom deployment system used in DICE consists of the four electric field probes, the corner probe mounts, the spool, the motorized braking mechanism. The whole system is shown in Figure 1.2. The four electric field sensors are located on each corner of the deck plate (shown in red) and seated in a mount. The spool is located in the center of the deck plate and contains the wire for the booms. The brake assembly is located on the bottom of the deck plate and is used to control the deployment of the wire booms. The deck plate also serves as an interface between the wire boom deployment system and the electronics of the EFP.
The wire booms are deployed from the spacecraft by spinning up the CubeSats until the centrifugal force on the EFP’s sensors becomes sufficient to overcome the static friction present with the brake in the off position. To ensure that the deployment of the wire booms is stable, the rate at which the wire booms extend from the spool is controlled by the brake. On DICE a small, non-magnetic, piezoelectric actuator called a Squiggle motor is used to actuate the brake as illustrated in Figure 1.3. The piezoelectric actuator contains several small plates that flex when electrically excited. This excitation causes the nut inside to vibrate in an orbit or “hula hoop” motion which drives the lead screw back and forth. The Squiggle motor itself is only 1.8 x 1.8 x 6mm in width, height, and length respectively.
1.2.3 Challenges of Implementing Wire Boom Systems

In a torque free environment, angular momentum is always conserved and a free fall or microgravity environment is a good example of such an environment particularly when drag coefficients are small. This is an important consideration when deploying wire booms from a spacecraft because its attitude controllability can be difficult or in some cases impossible. This is because a spinning body will always tend to its minimum-energy spin state unless a sufficient outside torque can maintain a different spin orientation. The minimum-energy spin axis will always be the largest principal moments of inertia for the spinning body. The principal moments of inertia, represented by a triad called the principal axes, do not necessarily line up with the geometric axes of a body particularly when the geometry of the body is more complex and mass is unevenly distributed. While many satellites are simple in shape their mass is usually unevenly distributed due to the need to place components in various locations on the spacecraft. As wire booms are deployed from a spinning satellite the inertia about the spin axis is increased. This greater inertia about the spin axis theoretically stabilizes the spacecraft about that axis but could in reality lead to an undesired orientation due to energy losses or mode excitations in the wire booms.

The DICE spacecraft was no exception to these challenges which were exaggerated due to the size of the satellite and the limited space and power available for sophisticated control mechanisms. When analyzed as a solid body with uniformly distributed mass the DICE spacecraft is naturally a minor axis spinner. As a minor axis spinner, meaning the spin axis has the smallest moment of inertia (MOI), the spacecraft would theoretically be stable but with the energy dissipated from the oscillations in the wire booms during deployment the spacecraft could begin to tumble. For this reason it was necessary to increase the MOI about the spin axis by adding mass at a distance from spin axis. This was accomplished by extending the length of the antenna booms to four times longer than originally planned and packing the tips with tungsten rods as shown in Figure 1.4. This solution put the greatest MOI about the desired spin axis. The mass moments of inertia can be seen in Table 1.2.
Table 1.2 DICE Spacecraft Mass Moments of Inertia Showing the Spacecraft as a Minor Axis Spinner and as a Major Axis Spinner. Spin Axis is the Z axis.

<table>
<thead>
<tr>
<th></th>
<th>Before Antenna Addition</th>
<th>After Antenna Addition</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Ixx</td>
<td>Iyy</td>
</tr>
<tr>
<td>Ixx</td>
<td>0.007891</td>
<td>0.007875</td>
</tr>
<tr>
<td>Iyy</td>
<td>0.004461</td>
<td>0.004461</td>
</tr>
<tr>
<td>Izz</td>
<td>0.004461</td>
<td>0.004461</td>
</tr>
</tbody>
</table>

Even with the moments of inertia favorable to the spin of the spacecraft, the wire booms would still need to be deployed at a controlled rate. If the booms were released too quickly the amount of energy freed from the system along with oscillations of the booms could cause the satellite to tumble uncontrollably. Several analyses were done on the DICE spacecraft and the dynamics of the wire boom deployment and the conclusion of those analyses with respect to the rate of deployment was that the boom would need to be deployed at a rate of <1 cm/s. The results of these analyses can be found in PowerPoint presentations located on the CD-ROM paired with this report. A thesis titled “Wire Boom Deployment Dynamics and Control System Model for Small Satellites” has also been written by a USU student which discusses the dynamics and control of wire booms being deployed from the DICE and ASSP spacecraft.

Figure 1.4 DICE Tungsten Weighted Antennas with Detail Showing the Dual Tungsten Rods at Antenna Tip

1 ASSP is a sounding rocket payload that measures the high-latitude electric fields using the MWBS
1.3 Report Overview

This report describes the design and outlines the development of the wire boom systems developed for the CubeSats of the DICE mission. This report will focus on the mechanical design and build of the EFP instrument, hereafter referred to as the Miniature Wire Boom Deployment System (MWBS). This report researches and describes the miniature wire boom system and its direct application to measuring the electric field density in the Ionosphere as well as its implementation on a CubeSat. Because wire booms have never been implemented on a CubeSat it is necessary to address and document design requirements and constraints as well as address challenges encountered during the build and test phases of the MWBS project. The purpose of this report is to create a viable design for a wire boom deployment system that can be used on a standard CubeSat platform.

In Chapter 2 we present the design goals and requirements for the MWBS project. The primary sources for the design requirements are the DICE science mission objectives. Additionally, a number of constraints were put in place by the “CubeSat Design Specifications” document published by Cal-Poly. The key design requirements are listed and the major challenges associated with these requirements are addressed.

Chapter 3 presents the design and build of the MWBS. The chapter is broken into section according to the subsystems within the MWBS. In each section follows the same basic format which answers three questions. The first question is, “what is this component/system and why is it necessary?” The second question is, “how did it get to its final state?” The third question is, “How did the changes improve upon the original design and how did the component/system perform generally?”

Chapter 4 then presents the performance results for each of the MWBS subsystems/components and is organized similarly to Chapter 3. The basic outline for each section within Chapter 4 is first, discuss why the test was performed and its objective. Second, a description of the test procedure and setup is given. Finally, a description of the test results are given.

Chapter 5 is the conclusion of this report and summarizes the results of the project and the overall performance of the MWBS and then provides recommendations for further development of the MWBS addressing specific problems and proposed solution(s).
Chapter 2

Design Goals and Requirements

There is currently no solution for deploying electric field sensors on a CubeSat platform although some concepts have been explored. This is a challenging engineering issue because the stowed system must be compact (~1 x10 x 10 cm) and non-magnetic and include the necessary conductors and insulators to electrically connect the sensors to the instrumentation. Above all such a system must be compatible with the attitude control and stability of the host spacecraft throughout deployment and operations. The miniature wire boom deployer was designed to meet all these requirements. In order for the miniature wire boom deployer to properly integrate with a CubeSat bus it had to conform to and established set of design standards.

The CubeSat design standards can be found in the “CubeSat Design Specifications” document which can be found at the website CubeSat.org which is maintained by Cal Poly. This document basically defines the overall allowable dimensions of the CubeSat, the mass allowance, and the materials that can be used for manufacture. The specifications for a 1U CubeSat are shown in Figure 2.1.

Figure 2.1 Cal Poly 1U CubeSat Dimensional Specifications
As discussed in Chapter 1 the design intent for the miniature wire boom deployment system was integration and use on a CubeSat mission called DICE. While the modular design and CubeSat compliance of the miniature wire boom system allows its use for variety of CubeSat or small satellite applications the deployer for DICE was specifically designed for gathering electric field potential measurements. Thus, the requirements for CubeSat design and the DICE mission and science objectives inherently defined the design requirements for the miniature wire boom deployer. These requirements are listed in Table 2.1.

Table 2.1 Miniature Wire Boom Deployer Science-Derived Requirements

<table>
<thead>
<tr>
<th>Number</th>
<th>Requirement</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>The wire boom deployer shall fit within a 9x9cm area.</td>
</tr>
<tr>
<td>2</td>
<td>Magnetic materials shall not be used for the components of the wire boom deployer.</td>
</tr>
<tr>
<td>3</td>
<td>The spin rate of the wire boom deployer shall be controlled independently of the spacecraft/spinning body.</td>
</tr>
<tr>
<td>4</td>
<td>The electrical connection from the wire booms to the onboard science electronics shall be a point-to-point connection. The use of brushes and slip rings etc. shall not be allowed.</td>
</tr>
<tr>
<td>5</td>
<td>Each electric field sensors shall be spherical, ≥ 0.75 cm diameter and deployed ≥ 2 meters from the spacecraft with a goal of 10 meters tip to tip distance.</td>
</tr>
<tr>
<td>6</td>
<td>The location of each of the electric field sensors in the spacecraft body coordinate shall be known to within 1/1000 of the boom length when the sensors are fully deployed</td>
</tr>
</tbody>
</table>

Each of these requirements had a significant impact on the final design for the wire boom deployer. There are, however, two of these requirements which played the largest part in the final product. The first of these was requirement 2 which states “Magnetic materials shall not be used for the components of the wire boom deployer”. The second was requirement 4, “The electrical connection from the wire booms to the onboard science electronics shall be a point-to-point connection. The use of brushes etc. shall not be allowed”.
The requirement to avoid magnetic materials was derived from the science requirement M1\(^2\) which outlines the accuracy of the measurements made by the science magnetometer. This requirement created a difficult problem for a few reasons. Since many common hardware items such as fasteners, springs, pins, washers, etc. inherently magnetic even if they are stainless steel. Where possible, 304 stainless steel was used for any hardware items since it is virtually non-magnetic. Springs, which were necessary in the design, are always somewhat magnetic if they are made with any kind of steel because they are cold worked during manufacture. Although the springs and fasteners would only be a small source of magnetic noise the use of anything such as a traditional magnetic-core motor could be a large enough source of noise as to saturate the measurements taken by the science magnetometer. It was impossible to meet this requirement exactly for the reasons mentioned above but it was still important to avoid using materials that would cause significant magnetic noise.

The major exception to requirement 2 was the use of a magnetic tracking device for an actuator. Early in the conceptual design phase it was decided that a small piezoelectric motor called a Squiggle Motor would be used as an actuator for the braking system. The Squiggle Motor became one of the key elements in the design for the wire boom deployer. As the system developed and as extensive testing was performed it became clear that the use of the small position tracker would be necessary for proper operation of the Squiggle Motor. This position tracking module, shown in Figure 2.2, uses a magnetic strip and Hall sensors to detect the spatially varying magnetic field created when the sensor moves parallel to the Hall sensors. Even though the use of the magnetic tracker was a violation of the requirement the simplification and reliability it provided to the mechanical and software/control design made it a justifiable exception.

The other requirement that had a heavy influence on the deployer design was the requirement to have a point-to-point electrical connection in the wire booms. Aside from the wires used for the booms being structural they are also the electrical connection from the sensors to the science electronics and thus carry the measurement signal all the way down to the science PCB from the spheres. Because this measurement is so small it is sensitive to noise and the use of a brush or slip ring to transfer the signal would introduce to much unpredictability into the measurement. Therefore the point-to-point connection requirement was necessary to produce the cleanest possible signal.

\(^2\) See Appendix A
The point-to-point connection requirement was not derived from any one science requirement directly but was established to ensure the cleanest possible measurement signal. In addition to the previously mentioned requirement to avoid magnetic materials, this requirement too, added a great deal of complexity to the deployment spool assembly design. The details of this design are contained in section 3.1 of this report. Simply stated this requirement forced the use of a very intricate inner spool design with a void in the center which would accommodate each wire boom to deploy with no intermediary connection by the use of a take-up ribbon cable.

While these two requirements dictated a majority of the design each design or performance requirement became very relevant at some stage in the design process.
Chapter 3
Design and Build

The objective of this chapter is to give an overview of the various components of the miniature wire boom deployer and outline the design considerations and evolution of the system. The miniature wire boom deployment system is a complex mechanism that is sensitive to dynamic factors such as balance and friction. These factors affect the sensors and actuators in the deployment system and without characterizing and controlling them the wire booms could not be deployed. The following sections discuss the major subsystems/components that make up the miniature wire boom deployer and the how the aforementioned factors affected each subsystem’s/component’s design and evolution to create a functional wire boom deployer.

3.1 Spool Design

To fulfill requirement to be able to deploy the wire booms to a length of 2m with a goal of 5m the spool needed to be large enough in diameter to accept a significant amount of wire. To be able to deploy 5m of wire in 4 directions the spool mechanism needed to be able to spin independently of the spacecraft and still maintain a point-to-point electrical connection. Thus the spool is made up of two sub-assemblies: the inner spool assembly and the outer spool assembly which are shown in Figure 3.1.

The entire spool assembly went through numerous iterations as additional requirements were introduced and as the assembly went through testing. A few of the major changes were the complete redesign of the inner spool, the material change of the discs for the outer spool assembly, and the redesign of the top plate of the outer spool assembly.

3.1.1 Inner Spool

The inner spool assembly essentially ties everything to the spacecraft both electrically and mechanically. Mechanically it is fastened to the PCB at the inner spool base by four fasteners that run through the entire thickness of the inner spool and consequently secure all inner spool components together. Electrically it acts as the conduit for the five wires for the EFP instrument. These wires are combined into one 5-lead ribbon cable which is tightly wound around the inner spool spacer and which allows the outer spool to spin.
Figure 3.1 Deployment Spool Assembly
The inner spool assembly evolved from one solid piece that was not machinable to a several-layer design to match the outer spool and finally to the final 4 piece design. The 4 piece design allowed for the five wires to be combined into one 5-lead ribbon cable. The ribbon cable greatly simplified the problem of how to get a point-to-point electrical connection. The inner spool spacer allowed the ribbon cable to be wound up tightly inside the inner spool and also provided the room necessary for the ribbon cable to unwind as the outer spool assembly spins to release the wire booms.

3.1.2 Outer Spool

The outer spool assembly contains up to five meters of 29 AWG wire which become the booms when deployed. Because the outer spool assembly is free to rotate relative to the inner spool, as the spacecraft spins up to stabilize centrifugal force acts on the EFP’s and the wire booms are deployed. The outer spool assembly is made up of thin disks machined out of aluminum and PET polyester. The top of the outer spool assembly is attached to the inner (non-rotating) portion of the spool and a bushing/bearing is used to allow rotational motion.

As a way to reduce mass the main (larger diameter) discs were made from Delrin. During the testing phase it was discovered that the outer spool assembly was very sensitive to the flatness of its discs. The basic idea is the flatter the discs the more uniform planar rotation is observed. There was an estimated .015”-.030” waviness in the Delrin discs which caused the brake ring attached to the bottom of the outer spool assembly to rub on the PCB. The rubbing not only induced unnecessary friction into the system but also eroded the solder mask off the PCB and destroyed critical electrical traces. The solution was to make all layers of the outer spool assembly, except the pliable polyester, out of aluminum. When machined, the aluminum would remain flatter than the Delrin. This improved the spool assembly’s performance noticeably.

In connection with the previously mentioned design change the outer spool top disc went through an additional design iteration. There were a couple of issues that drove the change, however, the first and foremost was the flatness problem. While the material change in all the discs helped the spool assembly performance the top disc which makes direct contact with the bearing was still only .030” thick and the residual waviness in that part still caused the entire outer spool assembly wobble. The final product, shown in Figure 3.2, was a .110” thick piece of aluminum which retained specified flatness as well as
accommodated wire harnessing for the point-to-point electrical connection via shallow channels cut into the top surface of the disc.

3.1.3 Spool Bearing

Even though it is referred to as a bearing this piece does not bear much weight but instead acts as more of a bushing to create a spin axis for the outer spool assembly. The bearing design basically remained the same through the design process. However, the changes that did take place to the bearing came about primarily to address three problems, friction, bearing slop, and deployment spool stability.

The first bearing design was small in diameter compared to the diameter of the outer spool assembly and was originally made from aluminum. The aluminum bearing piece was changed to Delrin during the study on the spool system friction to help reduce friction as well as the risk of thermal expansion issued posed by the difference in thermal expansion coefficients between aluminum and Delrin. At the same time this change was being made the diameter of the bearing piece was enlarged slightly to eliminate the large
amount of free-play at the bearing-outerspool interface. Figure 3.3 Shows the old bearing design and the amount of slop in the spool. This elimination of lateral play in the outer spool assembly around the bearing was key in the successful performance of both the deployment spool and the braking system.

Figure 3.3 Diagram of Old Bearing and Loose-Fitting Bearing

During the system testing the encoder was not operating as expected. The test data and a close inspection of the deployment spool operation made it clear that the stability of the spool while spinning was significant in the performance of the encoder. With little time left for a drastic overhaul of the deployment spool the bearing piece and outer spool top disc were redesigned instead. The change was a larger diameter interface to increase stability by decreasing the allowable tip angle and increasing the contact area. However, with the increased contact area and weight came increased friction. This was overcome by using a lubricant called Black Magic, a moly-based lubricant with Teflon. The results of the redesign were increased stability, better encoder performance, and a negligible friction increase. The final design is shown in Figure 3.4.
3.2 Encoder Design

The purpose of the encoder is to track the position and velocity of the spool as the wire booms are deployed from the spacecraft. The two components that make up the encoder system are the Avago optical surface mounted encoder chip and the encoder ring. These components are shown in Figure 3.5.

Through much of the development of the wire boom deployer the purpose for tracking the position and velocity of the spool assembly was to feed that information into the controller for the braking system. The need for position and velocity information changed when the braking system reached its final design and
velocity information was no longer required. Position information, however, remained critical information that was needed to track the deployed length of the booms. The final encoder ring design was driven in part by the problem of finding materials with adequate reflectivity properties for optimal performance, which requires a reflective surface contrasted by an absorptive surface.

Figure 3.6 shows that the encoder ring is really made up of two components, the ring itself and the backing. The encoder ring, which provides the highly reflective surface, is made of .005” brass coated with hard gold to achieve the specified reflectivity. During initial testing with the first batch of encoder rings it was discovered that the optical encoder is extremely sensitive to scratches and blemishes in the reflective surface of the encoder ring. A closer inspection of the encoder rings revealed light striations across the surfaces that were introduced during the manufacturing process. In addition to the striations on the surface the window-to-bar ratio on the encoder was not 1:1. One of the specifications for the optical encoder chip is that the window-to-bar ratio be in the range of 0.9-1.1. The specification given to the manufacturer of the encoder ring was to have a 1:1 ratio, this was not met. These imperfections are clearly shown in Figure 3.7. As a result the encoder system performed poorly, registering less than 30% of the total counts. The encoder rings were manufactured a second time with ultra-smooth surfaces and performed well.

See file “AEDR-8300.pdf” Avago Data Sheet on CD-ROM

3 See file “AEDR-8300.pdf” Avago Data Sheet on CD-ROM
The encoder ring backing only needed to be any material that had a reflectivity of less than 10%. Initially it was thought that black anodize would perform well, however, during testing it was found that the encoder was only registering about 25% of the 1680 counts around the encoder ring. A couple of materials were substituted in place of the black anodize and the best performance came from black construction paper and a graphite spray lubricant called Aerodag-G both of which enabled the encoder to register 98% of the total counts\(^4\). Aerodag-G was chosen as the final encoder backing material since it was easy to apply to aluminum and would also be stable in a high vacuum environment.

Despite the relatively good performance with the aforementioned changes to the encoder system a small number of counts were being missed for each revolution of the spool. It was calculated that if this many counts were missed for each revolution of the spool that over the full deployment of wire booms the final length of the booms could be off by as much as 12 inches. The mean error for two test sets is shown in Figure 3.8. This data shows how the encoder counts missed can affect the known length of the wire booms.

\(^4\) See file “AEDR-8300.pdf” Avago data sheet on CD-ROM for performance specifications
The final parameter in the encoder system design was the gap between the optical encoder chip and the encoder ring surface. Figure 3.9 shows the gap being defined as the distance between the surface of the optical encoder chip and the surface of the encoder ring. The data sheet for the Avago AEDR-8300 series encoder states that the gap must be between 1mm and 2.5mm. Although the nominal gap in the design fell within this range it was at the high end and resulted in poor performance with all of the encoder backing materials that were tested.

Because the science measurements required knowledge of the wire boom sensor positions to an accuracy of 1/1000 of the deployed length that meant that only 4 missed counts would be acceptable. Through testing it was found that at the gap distance of 1.5mm the encoder system performed flawlessly counting all 1680 counts per revolution.

Figure 3.8 Known-Length Error in Wire Booms Due to Missed Encoder Counts

Figure 3.9 Encoder Gap
3.3 Probe Design

As mentioned in the introduction to this report the electric field probes (EFP's) are the sensors that make the measurements of the electric field possible and are located at the tips of the wire booms. The design of the EFP's for DICE are a derivative of similar instruments used on larger space missions. One recent mission was the Floating Potential Measurement Unit (FPMU), which flew a couple of spherical probes, served as a large scale model for the much smaller EFP's used on the wire boom deployer. The spherical design provides a symmetric surface geometry for simpler measurement processing while the hard gold finish allows for a very clean uniform surface for more accurate measurements. These design parameters were well established and never changed through the design process, however, the sphere material itself and electrical interface to the spherical sensor became challenging design issues.

During the early stages of design and development for the wire boom deployer there was an interest in modeling the dynamics of the system to study spin rates and resulting stress in the wire booms. To ensure that the wire booms would not break during deployment aluminum was chosen as the sphere material and 36 gauge stranded copper wire with a Teflon coating was selected. Initial testing of the deployment system showed both the aluminum spheres and the Teflon coated wire performed well with the system spinning at approximately 3.5Hz

As the design for the deployment spool matured the rotational friction in the system became better understood and turned out to be much greater than was initially estimated in the dynamic model. The model assumed negligible friction compared with the applied torque but the measured friction of the rotating system was measured to be about 20 gram-force approximately 5 times the amount of force on the aluminum tipped booms while stowed and spinning at 2Hz. As a solution to this problem the mass of the sphere was increased by switching material from aluminum to tungsten which is more than 7 times denser bringing the weight of the spheres from 1.4g to over 8g each. The result of changing the sphere material lowered the deployment threshold from over well above 4Hz to less than 2Hz. Figure 3.10 shows what spin rate is required based on system friction and sphere weight.

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5 See file “DICE Wire Boom-Dynamics-Rev5.xlsm” on CD-ROM
6 The Small Satellite Test Facility spin table could only spin the system to about 4Hz before the spin rate became unstable. Even at this rate the booms showed no signs of initiating deployment.
Because the centrifugal force acting along the wire booms would be greater with the larger tip masses it was necessary to change the wire to a much stronger 29 gauge stranded copper wire with the same Teflon coating. This increased the tensile strength of the wire booms from 3N to 15N\(^7\). Another reason the wire was changed from 36 to 29 gauge wire was because the 36 gauge wire is so small trying to mechanically attach it to the sphere would be extremely difficult.

![Deployment Force Graph]

Figure 3.10 Sphere Mass Required for Deployment

Integrating the electrical wire to the spheres was a challenging problem from the beginning of the design and a variety of ideas were tested and most had highly variable results and in all cases the connection had low tolerance for fatigue. A tensile test was also conducted with each integration method to determine the reliability of the mechanical connection. The final solution was to use a silver impregnated high temperature epoxy to bond the stripped and tinned wire into the tungsten sphere. The tensile test results were consistent and showed a large safety margin against breaking due to the applied centrifugal force.

\(^7\) See file “calmont-eng-wire-gauge.pdf” on CD-ROM
force. To address the issue of fatigue with the connection, special care had to be taken during assembly to prevent the epoxy from wicking along the wire.

3.4 Corner Mount Design

The corner mounts located on the deck plate (PCB) provide support for the EFP’s while stowed during transportation and launch. They also act as a guide for the wires as the booms are deployed from the spacecraft. The assembly consists of the corner mounts or supports for the spheres and a guide post for the wire as it comes off the deployment spool and through the corner mount.

During early development the corner mounts were nothing more than a simple backstop support for the EFP’s, to help keep them positioned on the deck plate and to act as a stopping point when winding the wire booms onto the deployment spool. A more complete solution was needed in order to protect and contain the spheres during transportation and launch and a cup-like solution was devised and proven to be effective in containing the spheres but also allowing them to deploy when necessary.

Through the course of much of the testing of the integrated system the cup style corner mounts worked well. They allowed for a reliable and repeatable test setup as long as the wire was 1m or less in length. Once the length of the wire booms was increased significantly beyond 1m the ability to end up with wires the same length after winding could not be achieved easily or at all. There are many factors that contributed to the uneven wire lengths such as tension on the wire during winding, consistency and stacking of wraps on the deployment spool, consistency of wraps on the wrapping jig, and imperfections in the wire such as kinks or bends.

Figure 3.11 Picture of the Deployment Wire Wrapping Jig
A wire wrapping jig, shown in Figure 3.11, was developed for the purpose of controlling the variables during winding, however, it became a system full of difficult to control variables. It did, nevertheless aid in managing the 5m of wire that would be wrapped onto the spool. This jig was used along with a procedure for cutting the wire to precise lengths in order to get final length differences that could be dealt with. Once this method was established wire booms could be wrapped to within several millimeters of each other.

Although the four 5 meter lengths of wire could be wrapped onto the spool to within millimeters of each other there was still that small distance that needed to be taken up somewhere in the system so that the spheres with their gold coating would not be damaged during vibration testing or launch. The solution was a spring-loaded cup corner mount that had several millimeters of travel allowing all four spheres to be snuggly secured into their respective corner mounts. The environmental vibration tests confirmed that the corner mounts worked properly and minimal wear was shown on the surface of the spheres. The various corner mount designs can be seen in Figure 3.12 with details showing the static cup and spring-loaded cup designs.

Figure 3.12 Evolution of the Corner Mounts from Left to Right: Trifold Design, Static Cup Design, Spring-Loaded Cup Design
3.5 Brake Design

Because the wire booms are deployed using the angular momentum of the spinning satellite and because there is a large change exchange of energy from the spinning spacecraft to the wire booms as they deploy it is necessary to control the deployment rate very precisely. The brake system allows the centrifugal force generated by the spinning motion to deploy the wire booms but only in very small increments, about 1/32 to 1/64 of an inch at a time. This controlled deployment allows the spacecraft to remain stable, spinning about the correct axis, during the entire wire boom deployment. With regard to the controlled deployment a few requirements were developed in order to aid in the design:

1) The wire boom deployment rate shall not exceed 1cm/s

2) The braking force shall be greater than 1.2N with a goal of 2.5N

A few other requirements were also developed concerning the deployment of the wire booms, however, they were requirements related to the spacecraft and it’s affect on the deployment and will not be discussed in this chapter of the report. The requirements listed above will be discussed in later portions of this section.

While the brake system does not contain as many pieces as the spool assembly it is the most complex and sensitive mechanism in the wire boom deployer. The brake system is made up of 9 pieces in total but can be categorized in to 3 groups: the squiggle motor, the brake arm and brake ring, and the brake linkage.

Because the brake system is so complex this section contains a brief discussion about each of the three groups and their design and evolution.

3.5.1 The Squiggle Motor

The squiggle motor, shown in Figure 3.13, is the actuator that actually turns the brake on and off. It is a small piezoelectric motor that can provide up to 20gf (gram force)⁸ and has a footprint of 12mm x 2.8mm with the drive screw installed. Apart from its size this motor was chosen because of its non-magnetic characteristics and its linear motion capabilities and is the only off-the-shelf component in the whole deployer assembly. It is driven by a tiny controller chip that is mounted on the wire boom deployer deck plate.

⁸ See file “SQL-RV-1-8_datasheet.pdf” on CD-ROM
The squiggle motor is the only component in the whole deployer assembly that only ever changed once. The original choice of motor was the Squiggle motor 1.8 which was sold as an OEM part for very custom applications. Mounting the motor proved to be very difficult to do and if done incorrectly would cause the motor to stall or fail.

The first mounting design was simply a piece of stainless steel foil that the Squiggle motor was intended to snap into. This mounting design was unpredictable and allowed for excessive slop in the motor position relative to the other brake components. Finally a more permanent and secure mount was designed based on the development module offered by Newcale Inc. While both mounting methods were recommended methods of mounting the 1.8 Squiggle motor the latter design resulted in much more consistent and measurable performance.

As the development of the brake continued it became apparent that using the unpackaged 1.8 Squiggle motor would not suffice. In the way the motor was implemented into the design, shown in Figure 3.14, the lead screw from the motor would make physical contact with the brake arm/linkage and as the arm rotated the forces in the lead screw would change from purely axial to a combination of axial and lateral forces. The manual for the 1.8 Squiggle motor states that “The exposed screw shaft should not be subjected to any lateral forces...Side-loading the screw may impede motor performance and/or permanently damage the motor.”
Alternatives to the 1.8 Squiggle motor were explored including its big brother, the 3.4, which could produce 2N of pushing force. The available pushing force of the 3.4 motor alone would have greatly simplified the brake system design since all of the built-in mechanical advantages could have been eliminated. However, once a concept mount was designed for the 3.4 it was discovered that it wouldn’t fit in the very small envelope available inside the DICE spacecraft. The second alternative was to use another version of the 1.8 Squiggle motor, the M3-L module. The M3-L module has the same performance specs as the 1.8 Squiggle motor but comes in a clean package with all necessary control electronics and the ability to push or pull and track the position of the lead screw. The favorable design of the M3-L module also isolated the lead screw from all lateral forces. A diagram illustrating the M3-L module’s components and design is shown in Figure 3.16. In the end the M3-L performed extremely well and the tracking system became an essential part of the braking system. For comparison the various Squiggle motor models can be seen in Figure 3.15.
3.5.2 The Brake Arm and Brake Ring

The pieces of the braking system that physically provide the stopping capability required for a controlled deployment are the brake arm and the brake ring. The two parts need to be discussed together because they are two mating pieces that were specifically designed for each other. In fact in the final stages of implementation on the two DICE spacecraft there were specifically designated sets of brake arms and brake rings that were paired and assigned to each spacecraft. The brake arm is defined as the piece that makes direct contact with the brake ring which is attached to the bottom of the deployment spool assembly. Analogous to the brake pad and brake rotor on a car, they provide the stopping ability needed to control the wire boom deployment. They are made from 7075 aluminum for increased strength and are located on the top of the deck plate.

While the entire braking system went through a number of revisions this section of the report will focus on the final solution. Notes on other brake designs can be found on the CD-ROM paired with this report. The final design is a combination of many prior designs which were flawed but which also had some good components. These pieces that were adopted from previous designs will be discussed throughout this section.
The idea of an impinging brake design was one of the original design concepts proposed in the early stages of development but a lack of understanding concerning the Squiggle motor and its limitations put the idea on the shelf for much of the rest of the design. The original impinging brake design placed the brake arm inside the envelope of the brake ring under the deployment spool and when actuated made contact with the inside cylindrical surface of the brake ring. A number of materials and brake arm shapes were used but none of them could produce the required frictional force to control the deployment. This problem was a combination of a lack of friction between braking materials as well as poorly utilized mechanical advantage.

As the deployment spool matured so did the brake system design and for much of the deployer development the brake system was a strap wrench type brake, meaning that there was a material, a string made of various materials, wrapped around the outside circumference of the brake ring which would tighten when the brake was actuated. There were a number of problems with this design suffice it to say that there was no way to meet the minimum friction for deployment requirement and the braking friction requirement with this design. When this was realized the idea to turn back to the impinging brake arm was proposed. In light of all the design and test data that was collected and an overall better understanding of the motor and brake system limitations and requirements the impinging brake concept seemed like a workable solution.

The first update to the impinging brake arm was the concept of using small teeth, cut into the contact surface of both the brake arm and brake ring via wire EDM, to provide a greater amount of friction while in contact with the brake ring. This interface can be seen in Figure 3.17. During the assembly and initial testing of this idea it seemed to work better than expected providing a holding force of up to 15N depending on the adjustment of the braking system. Unfortunately, the system could not hold consistently around a full revolution of the deployment spool. Extensive testing was carried out with this braking system to try and characterize its behavior and understand the inconsistencies in its braking ability. Inconsistencies were present in tests of all the previous braking systems but only in testing with this new brake design was it noticed that the inconsistencies were actually occurring in the same location around the brake ring and that the location of the inconsistencies was based on how the deployment spool was assembled. It was determined through precise measurement with a Coordinate Measurement Machine
(CMM) that the deployment spool, when assembled and installed, was not exactly concentric but had a tolerance of about ±.010” in either the X or Y directions. In addition to the concentricity tolerance there was also a .005” gap around the bearing, hereafter referred to as bearing slop, bringing the total variance to .015”. Even though this variance is about 4 times thicker than typical copy paper the fact that the brake arm and brake ring only had a .010” engagement the system was bound to fail with this relatively large variance.

![Image](image.jpg)

Figure 3.17 Fine Tooth Impinging Brake Ring and Brake Arm Contact through Microscope

One of the corrective actions taken to tighten the tolerance of the deployment spool assembly was to create a very precise and thorough assembly procedure⁹. This procedure prescribed the use of 4 precision alignment pins that would align the many pieces of the deployment spool with better accuracy shown in Figure 3.18. The result was a decreased assembly tolerance of ±.005”. This decreased tolerance improved the brake system performance but there were still inconsistencies in the braking force around the ring due in part to the bearing slop.

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⁹ Assembly and test procedures are found on the companion CD-ROM
Given the unexpected potential of the new braking system a revision was approved with the objective of eliminating the effects of the remaining concentricity tolerance and the ever-present bearing slop. This revision specified longer-deeper teeth in the brake ring and brake arm that had the potential to engage up to .025”. The performance of the braking system improved again providing large enough holding forces to break the boom wire during tensile tests. Despite the improved performance the system still did not meet the deployment rate requirement of less than 1cm/s. There were still sections around the brake ring that would sporadically slip considerably, deploying up to 2 inches at a time.

It became apparent that there was another factor present that was causing the brake system to underperform. This factor was never clearly identified but was assumed to be tolerance stack up. Because tolerances were going to be a difficult variable to overcome another brake arm design change was made. This change led to a ratcheting system that would control the deployment with each stroke of the motor, shown in Figure 3.19. Basically the brake arm would have two engagement points, one at each end of the brake arm that would work in synchronization to let out a single metered amount of wire boom. This was accomplished when one end of the brake arm was engaged the other was clear of the brake ring but as the motor went through a stroke there was a point where both contact points would engage the brake ring. Then, the end that was engaged would be clear and the one that was previously clear would be engaged.
Along with the redesign the test data taken from the past several iterations was reanalyzed and it was confirmed that there was always only one section around the brake ring that was slipping but there was no way to predict beforehand where along the ring it would happen. Upon closer inspection of the brake and spool components and of the deck plate it was discovered that the holes in the deck plate used for mounting the brake components were as much as .010” from their called out positions. This clearly explained why even after extensive analysis with mathematical and 3D solid models the brake system was still not functioning correctly. The new brake components were immediately adjusted before manufacture to accommodate the incorrect hole positioning on the deck plate.

The new components were installed and performed well. Each set for each deployment system was tuned by hand with a file to ensure proper engagement all the way around the brake ring. The result was a perfectly controlled deployment of the wire booms 1/32” at a time. Because the components were fine-tuned by hand, each wire boom deployment system had a specific set of brake components assigned to them. The final design can be seen fully functional in final stages of testing in the videos on the CD-ROM.
3.5.3 The Brake Linkage

The brake linkage is the portion of the brake system that provides the transmittance of force from the Squiggle motor to the brake arm. Even in the earliest stages of design it was known that the squiggle motor by itself would not be able to produce enough force control the deployment. Every design of the brake system included some means of mechanical advantage to increase the amount of force transmitted to the brake arm. Because the amount of torque on a pivot point can be increased by using a longer lever arm a careful balance had be kept so that the necessary displacement of the longer lever arm would not exceed the travel range, or stroke, of the motor. Most of the early designs used a single lever to rotate a small spool or drum as shown in Figure 3.20.

![Figure 3.20 Two Early Designs Showing a Short Simple Lever Arm which Rotates a Spool/Drum](image)

When it was determined that the brake designs using a rotating spool or drum would not work a study was conducted to investigate the maximum potential force the squiggle motor could provide at the braking surface. This study showed that by careful design a maximum of 2.4 Newtons could be applied to the brake ring by the brake arm. This force met the requirement DR4\(^10\) which states, “The braking force shall be greater than 1.2N with a goal of 2.5N”. The final design that made this force output possible is called the brake linkage and is made up of the six pieces shown in Figure 3.21 and Figure 3.22.

\(^{10}\) See Appendix B
To be as efficient as possible at transmitting the force the rotating joints needed to be precisely fit together with little to no slop. It was observed that if the system had a noticeable amount of slop it significantly affected the performance of the brake. The travel distances of the brake linkage components and contact interfaces of the brake ring and the brake arm are so small that an average machining tolerance stack up of .005 inches between 2 or 3 parts could cause the entire brake system to fail. The slop in the brake linkage was noticeable because it would twist out of plane which prevented the brake arm from traveling fully through its intended range of motion. Similarly to the brake arm and brake ring this linkage had to be hand tuned with material such as Kapton and Teflon. Strips of these materials were used as shims.
and bushings at the points of rotation on both the pivoting link and the rotating spool. Each of these tuned linkages was specific to a miniature wire boom deployment system and the final performance of each linkage was such that the components would remain planar as intended and the maximum range of motion was utilized.
Chapter 4

Testing and Results

The chapter discusses the various tests that were conducted on the miniature wire boom deployment system. These tests were developed in order to characterize the system and better understand its behavior so that effective and meaningful design changes could be made. There are a few tests that were left out of this chapter either because the test was obsolete or because the test was simple enough that no detailed discussion is necessary. The final section will discuss the overall system performance and final results achieved with the completed and integrated wire boom deployment system and more complete records of test data along with videos and pictures of tests can be found on the CD-ROM.

4.1 Braking Force vs. Brake Arm Position

The Braking Force vs. Brake Arm Position test was developed to determine the brake positioning and settings for optimal brake performance. With the use of a frictional brake it was important to find the brake set point at which minimum system friction and required braking force could both be met. This test was designed to find that point and help define the assembly procedure for the braking system. It should be noted that this test is only valid for a purely frictional brake. In other words it is not possible to perform this test with the final brake system design which is finite-length release mechanism.

![Figure 4.1 Test Setup for Braking Force vs. Brake Arm Position](image)
The concept behind this test is to measure the system friction at the “brake off” and “brake on” positions and compare those measurements with the minimum friction and required braking force respectively. The completely assembled deck plate and deployment spool with approximately 1m of wire are mounted to a static test fixture. With the brake on the off position a highly sensitive force gauge was used to pull on the wire which is loaded onto the deployment spool and record the peak value through 1/16 of the deployment spool rotation. After one complete revolution of the deployment spool the brake is set to the on position and again the peak value is recorded through 1/16 of the deployment spool rotation. This test setup is shown in Figure 4.1.

This test or a variation of it was conducted dozens of times during the development of the spool. The test procedure became a formal document when the development of the brake system intensified and it became more critical to have test data that could be directly compared across multiple design iterations. It was the first test to be conducted on a new brake design and it became the indicator of whether or not a particular brake design would be workable. It was used to test many different brake materials and assembly procedures and variations. In the end it helped determine that with the available motor pushing force it would not be possible to meet the braking requirements. There was either too much system friction for the centrifugal force on the wire booms to overcome or there was not enough motor travel apply the necessary stopping force. This discovery led to the design of the final brake system solution. A series of these test results can be found on the CD-ROM.

4.2 Brake Push Bar Force vs. Brake Push Bar Position

The brake push bar force vs. brake push bar position test better characterizes the wire boom deployment system by directly measuring the force that the brake push bar applies on the squiggle motor lead screw with respect to its position. This force relates directly the speed at which the motor is able move and to react. The ability with which the motor is able to react controls the minimum deployable boom length which has an enormous effect on the spacecraft stability. This force is not only important to quantify to characterize the motor speed but also because the force applied back to the squiggle motor lead screw cannot exceed 20gf otherwise the motor will fail.

The basic concept of the test is to measure the reaction force of the brake push bar on the Squiggle motor lead screw by placing a force gauge in place of the Squiggle motor. The complete deck plate and
deployment spool assembly are mounted to a static test fixture which also has a sensitive force gauge attached to it. The force gauge has a long narrow stainless steel shaft mounted on the transducer which simulates the lead screw of the squiggle motor. The force gauge is mounted directly to an optical micrometer stage for very precise advancement of the force gauge. The force gauge is initially set to a predetermined zero position and then advanced 1/4mm and the force measured by the gauge is recorded. The test setup is shown in Figure 4.2.

The results of this test were particularly useful in redesigning the brake mechanism. More specifically the test helped to understand the forces that were occurring throughout the braking system and that the reaction force was not purely axial along the lead screw. Because the stainless steel shaft was very long and somewhat flexible it illustrated that there were components of lateral force in the system as well. The Newscale Technologies Squiggle motor data sheet warns that any lateral forces applied to the motor lead screw could lead to motor failure. This proved to be true. During system testing the squiggle motors that were subjected to lateral forces would easily stall and those that were subjected to excessive lateral forces were permanently damaged. Figure 4.3 shows an example of the test results from which it can be seen that the maximum travel of the motor lead screw is just more than .5mm.
Figure 4.3 Test Results for Brake System with Dyneema Fiber as Braking Material

4.3 Brake Force Anomaly Test

The Brake Anomaly test is very similar to the Brake Push Bar Force vs. Brake Push Bar Position test in that it serves to investigate reaction forces onto the Squiggle motor lead screw. This test further characterizes the reaction forces in the deployment spool and braking systems by measuring the force that the motor experiences as the deployment spool rotates to deploy the wire booms to ensure that that force does not exceed 20gf otherwise the motor could stall and become non-functional. Because the deployment spool and braking system are so complex it is very difficult to do a thorough study of their dynamic forces analytically therefore the Brake Anomaly test measurement these dynamic forces due to the wire boom deployment can be quickly measured. It should be noted that this test is only valid for a purely frictional brake. In other words it is not possible to perform this test with the final brake system design which is a locking brake mechanism.

This test has a similar setup to the Brake Push Bar Force vs. Brake Push Bar Position test. The micrometer stage is moved until the brake is almost completely engaged and the force gauge is then zeroed. The spool is then moved with two different methods, in the first method the spool is rotated by hand, in the second method a wire is attached to the spool and pulled with approximately 1N of force. The peak force is recorded through each 1/4 turn of the deployment spool. This test setup is the same as that shown in Figure 4.2.
The usefulness of this test was very limited but helped to further show that there were what appeared to be “high” spots and “low” spots around the brake ring. It was later determined through the use of a Coordinate Measurement Machine (CMM) that these “high” and “low” spots were from the eccentricity of the deployment spool components.

4.4 Spool Friction Test

The purpose of the Spool Friction test is to characterize the wire boom deployment system by directly measuring the frictional forces in that system. Because the wire boom deployment system requires very low frictional resistance for deployment it is important to identify the sources of friction within the system. If singular sources of large frictional forces are identified then a redesign could take place to eliminate the problem.

This test simply measures the friction of the deployment spool assembly which is complete with the boom wires. The test is conducted without any brake components installed and with the deckplate mounted on the static test stand in a either a vertical or horizontal configuration depending to see the effects of different gravitational forces on the bearing. The spool friction is then measured with a force gauge at every 1/16 increment of the deployment spool. This test setup is shown in Figure 4.4 and a diagram of known friction forces is shown in Figure 4.5.

Figure 4.4 Spool Friction Test Setup in Vertical Configuration
Figure 4.5 Diagram of the Frictional Sources in the Deployment Spool

This test was particularly crucial in identifying at least one concentration of frictional forces. The guide posts installed on the deck plate which were designed to guide the wire off the deployment spool and through the corner mount were large sources of friction. It was decided that one post would be removed to reduce the number of sharp turns the wire had to make. This change decreased the system friction by almost half as shown in Figure 4.6. The remaining guide post at each corner was adjusted so that each could spin freely as the wire traveled around the plastic sleeve. This adjustment further reduced the system friction so that it was as low as 4gf. This achievement made it possible to meet the requirement for minimum friction which was established by the maximum spin rate of the spacecraft.
4.5 Encoder Reliability Test

The Encoder Reliability test verifies the encoder’s ability to accurately track the rotational position of the deployment spool. The primary function of the encoder is to track the released length of the wire booms. This position information is used to account for the location of the measurements made by the EFP’s and is also used in processing those measurements. The reliability of the encoder is critical in tracking the position of the wire booms. Each encoder ‘tick’ represents about .006” of boom wire per revolution of the deployment spool which after the 22 revolutions of the deployment spool required for 5m of boom wire equals about .100”. The knowledge requirement for position of the wire booms is 1/1000 of the boom length which at 5m is 5mm or about .200”. Thus encoder reliability is important for accurate measurements.

While the setup of the Encoder Reliability test is complicated the concept is simple, count the number of encoder ticks missed over one revolution of the deployment spool. The basic setup consists of monitoring the two signal channels of the optical encoder chip using a logic analyzer and catching the data packets from the Field Programmable Gate Array (FPGA). This setup is shown in Figure 4.7. The signals from the encoder through the logic analyzer show the quadrature encoding high and low signals while the data packets from the FPGA provide the absolute encoder position data in hexadecimal format. By visual
inspection of the logic analyzer channels the number of counts around the deployment spool can be
determined albeit tediously. By taking the position data in the first packet and subtracting it from the last
data packet received the number of encoder counts can be determined. The total quadrature encoder counts
around the deployment spool is 1680.

This test was performed extensively both to design the optimal encoder system and to qualify the
encoder for flight. During the first Encoder Reliability tests the number of counts being missed through
one revolution of the deployment spool was large, about 75% of the counts were being missed. This was
due to two factors, the encoder backing material and the gap distance between the optical encoder chip and
the reflective encoder ring, both of these issues are talked about in more detail in Section 3.2. The Encoder
Reliability test enabled the optimization of the encoder system by varying the encoder backing and the gap
distance methodically. In the end the encoder system performed perfectly counting 100% of the counts in
most tests. A sample of test results can be seen in Figure 3.8, Figure 4.8, and Table 4.1, and more detailed
test data for this test can be found on the CD-ROM.
Figure 4.8 Encoder Test Data Showing Quadrature Signal with Few Missed Counts (See Table 4.1)

Table 4.1 Encoder Test Data

<table>
<thead>
<tr>
<th>Test Date</th>
<th>Encoder Start (hex, dec)</th>
<th>Encoder Stop (hex, dec)</th>
<th>#Missed Counts</th>
</tr>
</thead>
<tbody>
<tr>
<td>8/30/11</td>
<td>00 64</td>
<td>100</td>
<td>0</td>
</tr>
<tr>
<td>8/30/11</td>
<td>00 63</td>
<td>99</td>
<td>0</td>
</tr>
<tr>
<td>8/30/11</td>
<td>00 50</td>
<td>80</td>
<td>0</td>
</tr>
<tr>
<td>8/30/11</td>
<td>00 69</td>
<td>105</td>
<td>1</td>
</tr>
<tr>
<td>8/30/11</td>
<td>00 66</td>
<td>102</td>
<td>0</td>
</tr>
</tbody>
</table>

Setup:
- Baffle (Aerodag-G coated tape over encoder eye) on
- A “used” encoder ring
- Aerodag-G Backing to encoder
- Aluminum Spool Top Plate

Setup:
- NO Baffle (Aerodag-G coated tape over encoder eye) on
- A “used” encoder ring
- Aerodag-G Backing to encoder
- Aluminum Spool Top Plate

8/30/11 00 65 101 06 F5 1781 0
8/30/11 00 5D 93 06 EC 1772 0
4.6 Dynamic Deployment Spin Test

Because the wire booms would be deployed on orbit using the centrifugal force provided by the spinning spacecraft a small scale Dynamic Deployment Spin test was developed. The objective of the test was to observe the potential differences between a 'simulated' static deployment and a spinning deployment. The static deployments used for comparison were the integrated system tests discussed in Section 4.7.

Because of the complexity of the 6 DOF spinning problem the test was restricted to rotation about 1 axis using a DC motor powered spin table in the NOVA facility at the Space Dynamics Lab in Logan, Utah. Due to the size of the facility the length of the deployed wire booms was restricted to 1-1.5 meters. The basic test approach was to slowly increase the spin rate of the mock spacecraft with the attached wire boom deployer assembly from about .5Hz until the wire booms began to deploy. The spin rate increase was controlled manually by the operator and the spin rate was only increased once the spin rate had stabilized at the target value.

The results of this test were extremely helpful in the design of the entire deployment spool and it components. The spin tests revealed that the internal friction and its effects were detrimental to the performance of the system. During the initial spin tests the wire booms did not deploy at the maximum speed of the spin table which is 4Hz with the mock spacecraft attached. It was clear that they would not deploy unless the spin rate was much greater. These results led to the change of material for the EFP sensors from aluminum to tungsten and to the reduced friction design discussed in the Spool Design and Brake Design sections of Chapter 3. Once the changes had been made to the system the performance increase was dramatic. The wire booms deployed at spin rates as low as 1.2Hz but averaged at about 1.7Hz which was below 2Hz, a target handed down from the team designing the Attitude Determination and Control System (ADCS) for the spacecraft. A series of deployment videos can be found on the CD-ROM.

4.7 Integrated System Test

This test was used to verify all of the previous tests conducted during the design phase of the instrument. The Integrated System test was a full test of the assembled deployment spool and spacecraft with all control electronics running the system. These tests were used to verify the mechanical as well as the electrical systems for the wire boom deployment system.
The Integrated System test was performed in two different configurations. The first was in the transition phase from design to implementation. The second was during the spacecraft assembly and system testing. Testing in the first configuration was extensive as small bugs were worked out of both the mechanical and electrical systems. Once the major bugs were worked out of the deployment spool mechanism this test was very useful in proving the performance capabilities of the deployment spool and its components such as the braking system. To simulate the deployment a schedule of different weights was used in order to simulate different spacecraft spin rates and wire boom deployment lengths as shown in Table 4.2. A test was conducted at each weight set in the schedule to determine if the braking system and control electronics could actually control the deployment within the deployment requirements. The test setup is shown in Figure 4.9.

![Figure 4.9 Static Deployment/Integrated System Test Setup](image-url)
Table 4.2 Weight Schedule for Static Deployment

<table>
<thead>
<tr>
<th>Step No.</th>
<th>Components</th>
<th>Individual Boom Weight (N)</th>
<th>Total Boom Weight (N)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1.0</td>
<td>Tungsten Spheres</td>
<td>0.085</td>
<td>0.34</td>
</tr>
<tr>
<td>2.0</td>
<td>2 Nuts</td>
<td>0.185</td>
<td>0.74</td>
</tr>
<tr>
<td>3.0</td>
<td>4 Nuts</td>
<td>0.275</td>
<td>1.1</td>
</tr>
<tr>
<td>4.0</td>
<td>1 Bolt and 2 Nuts</td>
<td>0.385</td>
<td>1.54</td>
</tr>
<tr>
<td>5.0</td>
<td>1 Bolt and 4 Nuts</td>
<td>0.485</td>
<td>1.94</td>
</tr>
<tr>
<td>6.0</td>
<td>2 Bolts and 2 Nuts</td>
<td>0.595</td>
<td>2.38</td>
</tr>
<tr>
<td>7.0</td>
<td>2 Bolts and 4 Nuts</td>
<td>0.695</td>
<td>2.78</td>
</tr>
<tr>
<td>8.0</td>
<td>2 Bolts and 6 Nuts</td>
<td>0.785</td>
<td>3.14</td>
</tr>
</tbody>
</table>

In both test configurations mentioned above the test process was the same. The data gathered during these tests was particularly helpful in refining the braking system's mechanical components and because this test provided repeatable results from a given build of the deployment spool and braking system it was the single most helpful test in converging on the final braking system design, the ratcheting brake arm design.
Chapter 5

Conclusion

As mentioned through this paper, the miniature wire boom deployment system was intended for integration on a small satellite as a primary science instrument. This chapter discusses the overall performance of the miniature wire boom system. The miniature wire boom deployer is currently installed on the two DICE spacecraft, Yahtzee and Farkle, which were launched in October of 2011 as a secondary payload to the NASA NPP satellite. As of the printing of this paper the wire booms have not yet been deployed. The satellite is currently gathering science data from other on-board instruments and communications and other systems are still being tested in preparation for the deployment.

5.1 Performance and Results

The only results to speak of are those gathered from the various tests performed on the system during testing and integration. The most significant of these tests, however, was the Integrated System test which was a test performed to verify the performance of the entire miniature wire boom deployer system, both as a standalone unit and fully integrated into the DICE spacecraft.

While most of the requirements for the miniature wire boom deployer could be verified by inspection or directly satisfied by design there were a few that were performance related. These requirements will be discussed along with the related performance of the instrument.

The spin rate of the wire boom deployer shall be controlled independently of the spacecraft/spinning body. This requirement refers to the braking system and the control electronics. Because of centrifugal force caused by the spinning of the spacecraft the wire booms could deploy freely causing the spacecraft to become unstable and enter into a flat spin it was therefore critical to have a safety in place which would control that deployment rate and keep the satellite in a stable, even spin. The braking system is the means of controlling that deployment. During the Integrated System tests the braking system performed well controlling the deployment of the wire booms by allowing the deployment steps to 1/32 inch. This length increment translates into less than 1mm/s when considering the switching speed of the motor. Results of an analysis of the spacecraft spinning dynamics showed the amount of free energy in the wire booms as they
deploy. To keep the spacecraft stable the analysis revealed that the deployment rate should be less than 1cm/s. The deployment test results show a safety factor of 10 against unstable spin.\footnote{See file “Wire Boom Dynamics.pptx” found on the CD-ROM}

The electrical connection from the wire booms to the onboard science electronics shall be a point-to-point connection. The use of brushes and slip rings etc. shall not be allowed. Due to the sensitivity of the measurements being made by the EFP’s a continuous reliable connection was desirable to reduce the noise in the signal. This requirement was fulfilled by design. A lot of work went into the design and redesign of the deployment spool assembly to allow for the necessary point to point connection. The most integral pieces of this design were the inner and outer spool sub-assemblies, the latter of which could rotate independently of the former, see Figure 5.1. This independence in rotation was made possible by the bearing which went through extensive testing and design. The overall performance was good and after deployment tests the signal in the wires were strong, as expected.

![Figure 5.1 Diagram of the Spool Relative Motion](image)

The location of each of the electric field sensors in the spacecraft body coordinate shall be known to within 1/1000 of the boom length when the sensors are fully deployed. Again, because of the sensitivity of
the signals and the accuracy desired in the science measurements from the EFP’s, it is important to know very accurately where the sensors are with respect to the spacecraft. While there is not yet data available on the actual on-orbit deployment of the wire booms the test results were extremely good. The encoder system makes the fulfillment of this requirement possible by tracking the length of the wire booms with a resolution of about .006 inches or .15 millimeters. During many of the tests the encoder system worked flawlessly, counting each and every tick for more than a full revolution of the spool. With these results it will be possible to fulfill this requirement.

By the time the miniature wire boom deployment system was ready to launch aboard the DICE spacecraft almost all of the requirements (see requirement 2 in Table 2.1) established in the beginning were met along with all of the other derived requirements that came about during the testing phase of the program. With all these requirements having been met great confidence can be had in stating that the wire boom deployers will perform properly.

5.2 Recommendations

The first problem identified was the unevenness in the wires after winding 5 meters of wire onto the deployment spool for each wire boom. The unevenness in the wires causes one or more of the EFP sensors to not be seated properly or securely in its respective corner mount. If these sensors are not seated securely during transport and launch they could easily become damaged and likely break or at the very least perform unreliably.

The proposed solution was to put a small amount of tension on the wires when winding them onto the spool using the winding jig with precision weights for tension. It is also important to measure the wires precisely before loading them onto the winding jig so that the exact length is known after the wires are cut to their final length. It would also be useful to use rubber bands with wire hooks to maintain tension on each boom wire while removing the weights and in preparation for removing the spool from the winding jig. This still remains one of the most difficult problems because any imperfections in the wires or inconsistency in the wire wraps on the winding jig cause uneven wrapping onto the deployment spool. An example of this problem is shown in Figure 5.2.
A large source of problems during both the design and testing phases was the lack of really tight manufacturing tolerances. The standard manufacturing tolerance for the machine shop at the Space Dynamics Lab is ±.005 inches which is very good in most cases. It was discovered however, that the concentricity of all of the deployment spool parts and alignment of their many holes was critical for proper brake performance. Also, the PCB, or deckplate which functioned as both the mechanical and electrical interface for the wire boom deployer was not manufactured with very tight tolerances. Some of the holes that were used for mounting various parts of the braking system were out of position by as much as .017 inches. These poor manufacturing tolerances contributed greatly to the underperformance of the braking system and also cost a lot of money in the form of design time and manufacturing costs which were necessary for the many design iterations.

For all of the deployment spool components tighter concentricity tolerances need to be specified, something on the order of ±.002 inches would be appropriate. The tolerances for the PCB manufacture also need to be specified to at least the machine shop standard of ±.005 inches and if surface mount solder nuts are used on the PCB they need to be accurately placed. It was learned that in most cases the relative positions of the holes punched into a PCB are correct but when referenced to an edge they can be out of specification. The solution in this case would be to trim the PCB edge, if possible, until the dimensions fall within specification.
While the lubricant used on the bearing, Black Magic dry moly lubricant, was sufficient and provided great lubricity there is perhaps a better option. A thermal set version of the same lubricant is more stable in a space environment and would be more robust for long term use. Caution should be taken against using a graphite lubricant, it does not function in the space environment, its lubricity depends on moisture and air layers in its molecular structure. Also, its physical properties change and its coefficient of friction increases in a vacuum environment. It can also be corrosive to aluminum.

It was determined late into the development of the wire boom deployer that the wires used for the booms would be exposed to external sources of electron charging that would disrupt the measurement signal. In an attempt to solve this problem a graphite spray was used to shield the wire as shown in Figure 5.3. This application process was time and resource intensive and ultimately produces marginal results. Because the wire is encased in Teflon, which is meant to be a non-stick low COF product, the wire shielding was spotty and easily flaked off the wire as shown in Figure 5.4. The attempt was abandoned because of the high risk of contamination to the other spacecraft components.

![Figure 5.3 Graphite Coating on Teflon-Encased Wire](image-url)
A more permanent and robust solution was devised by Dr. Charles Swenson in conjunction with Calmont Wire Technologies. This solution would provide a small gauge wire with exceptional flexibility and the necessary EMI shielding. A diagram of this concept is shown in Figure 5.5. This solution, however, was never implemented due to the long lead time required for manufacture.
While the Squiggle motor chosen for the final design was adequate for overall performance the capabilities of the Squiggle motor itself left little margin for braking ability. As a suggestion the 3.4 Squiggle motor, just larger than the 1.8, could be used to increase the performance margin of the braking system. Another option would be to use a piezoelectric rotary motor that could not only deploy the wire booms but also eliminate the need to have a separate braking system.

The deployment spool bearing is another critical component for proper functionality of the wire boom deployment system. The bearing affects the braking ability as well as the deployment dynamics based on the support it gives to reduce tipping of the deployment spool. The final design used in the deployment spool assembly was a larger in diameter than previous revisions and was made from a thicker piece of aluminum in order to maintain better flatness tolerances. However, because the diameter of the bearing was still small compared to the diameter of the outer spool assembly and because the lip that the outer spool assembly rests on was so small the spool assembly was allowed to tip out of plane considerably, as illustrated in Figure 5.6. A solution to this problem could be an even greater diameter bushing creating an actual bearing that would sustain side loading as well as restrict tipping out of plane.

![Figure 5.6 Picture (Top) and Diagram (Bottom) Showing Spool Tipping Issue](image-url)
Because the miniature wire boom deployer is such a complex and sensitive system, it isn’t perfect. The above sections are the result of an evaluation of the miniature wire boom deployment system which was performed to identify the most critical and worthwhile revisions that could be made to the wire boom deployer and its subsystems and components. As a conclusion to this paper each of the identified issues should be considered when redesigning the wire boom deployer for another application.
References


Appendices
Appendix A

Dice Science Requirement

Mission Objectives

MO1 Investigate the physical processes responsible for formation of the geomagnetic storm enhanced density bulge in the noon to post-noon sector during magnetic storms.

MO2 Investigate the physical processes responsible for the formation of the geomagnetic storm enhanced density plume which forms at the base of density bulge and the transport of the high density plume across the magnetic pole.

MO3 Investigate the relationship between the penetration electric fields and the formation and evolution of the storm enhanced density bulge and plume.

Science Requirements

S01 Return continuous observations of the ionosphere using two spacecraft that are within the same orbital plane and within 1-min to 6-min of each other.

S02 Return observations of the ionosphere from ≥ 55 degrees latitude with a preference of observations ≥ 80 degrees latitude.

S03 Return 90 days of observations of the ionosphere from the 13 to 17 local time sector using two spacecraft that are within the altitude range of 350 km to 800 km with a goal of 180 days. Run

S04 Return observations of co-located electric fields, magnetic field fluctuations, and plasma density at a ≤ 10 km on-orbit spatial sampling with a ≤ 0.1 km goal and that are absolute time located to within ≤ 1ms UT.

S05 Return observations of the presence of electric fields fluctuations in the 10 to 1000 Hz range at a ≤ 10 km on-orbit spatial sampling.
Instrumentation Requirements

**Electric Field Instrument**

E1 The spacecraft shall return post-flight, post-analysis, observations of the ambient electric fields perpendicular to the local magnetic field with an accuracy of \( \leq 2 \) mV/m and with a goal of \( 0.1 \) mV/m over a range of \( \pm 200 \) mV/m.

E2 The electric field instruments shall provide two double probe observations with sensors deployed 90 degrees \( \pm 0.1 \) degrees relative to each other and simultaneously sampled.

E3 Each electric field sensors shall be spherical, \( \geq 0.75 \) cm diameter and deployed \( \geq 2 \) meters from the spacecraft with a goal of 10 meters tip to tip distance.

E4 The location of each of the electric field sensors in the spacecraft body coordinate shall be known to within 1/1000 of the boom length when the sensors are fully deployed.

E5 The electric field sensors shall be rotated about the axis perpendicular to the deployed plane of the sensors with a frequency, \( f \), such that \( 0.1 \leq f \leq 10 \) Hz and at the geometric center point of the sensors to within 1/1000 of the boom length.

E6 The electric field booms shall be deployed to be perpendicular to the Earth’s geodetic axis of rotation to \( \leq 30 \) degrees.

E7 The post-flight attitude knowledge of the electric field boom orientation shall be \( \leq 0.1 \) degree with a goal of \( \leq 0.01 \) degrees.

E8 The electric field instrument shall provide measurements under the environmental conditions for ambient plasma density of \( 10^2 \) to \( 10^8 \) cm\(^{-3}\).

E9 The electric field instrument shall provide measurements under the environmental conditions for temperature ranging from -10 to 30 degree C.

E10 The electric field instrument shall operate from power supplies of +5V and the unregulated spacecraft battery bus (7.4 ±0.8V). It shall have a total power usage of \( \leq 50 \) mW (TBR).

E11 The electric field instrument shall require no more than 4.0 x 4.0 cm of circuit board space.

E12 The input impedance of the electric field instrument shall be \( \geq 10^{13} \) Ohms and the instrument shall be in a short to ground configuration until deployed.

E13 The electric field probe shall be able to accommodate an induction field, VxB, plus ambient field of \( \geq 600 \) mV/m without saturating.
E14 The electric field instrument shall provide two 16-bit two's complement values to the telemetry system representing the double probe measurements, V12 and V34, of the observed fields.

E15 The electric field instrument shall provide at least 3 points of electric field spectral information from the frequency range of 10 to 1000 Hz at a ≤ 10 km on-orbit spatial sampling rate.

Magnetic Field Science Instrument
M1 The science magnetometer shall return post-flight, post-analysis observations of ambient magnetic fields with an accuracy of 2 nT (TBR) and a signal to noise ratio of ≥ 3 over a range of ≥ ± 50,000 nT.

M2 The science magnetometer z-axis shall be aligned with the spin axis of the spacecraft to within 2 degrees.

M3 The science magnetometer x and y-axis shall be aligned to the electric field boom axis to within 2 degrees.

M4 The spacecraft shall have residual and stray magnetic fields of rms amplitude of ≤ 2 nT (TBR) at the location of the science magnetometer.

M5 The science magnetic field instruments shall operate with a power supply of +5V with a total power usage of ≤ 100 mW (TBR).

M6 The science magnetometer shall provide measurements under the environmental conditions for temperature ranging from -10 to 30 degrees C.

M7 The science magnetometer shall require no more than 4.0 x 4.0 cm of circuit board space.

M8 The science magnetometer shall provide three 18-bit two's complement values to the telemetry system representing Bx, By and Bz of the measured field.

Ion Langmuir Probe
L1 The ion Langmuir probes shall return post-flight, post-analysis plasma density observations over the range of 2x10^3 to 2x10^7 cm^-3 with a resolution of 350 cm^-3.

L2 The voltage on the sensor shall be ≤ -5 volts relative to the spacecraft structure with a goal of -8 volts.

L3 The ion Langmuir probe shall be aligned with the spacecraft spin axis and return observations of the ion ram current in wake free regions.

L4 The ion Langmuir probe shall provide two observations with sensors deployed 180 degrees ± 0.1 degrees relative to each other and simultaneously sampled.
L5 Each ion Langmuir probe sensor shall be spherical, ≥ 1.27 cm diameter and deployed ≥ 5 cm from the spacecraft such that one of the sensors shall be outside of the spacecraft wake at all times.
L6 The ion Langmuir probe shall provide measurements under the environmental conditions for temperature ranging from -10 to 30 degrees C.
L7 The ion Langmuir probe instrument shall operate from power supplies of +5V and the unregulated spacecraft battery bus (7.4 ±0.8V). It shall have a total power usage of ≤ 50 mW (TBR).
L8 The ion Langmuir probe shall require no more than 4.0 x 4.0 cm of circuit board space.
L9 There shall be no exposed potentials on the spacecraft to the space environment aside from the ion Langmuir probe. All solar panel interconnects shall be isolated from the plasma and all connectors shall similarly be covered during operation of the spacecraft.
L10 The surface of the ion Langmuir probe shall be cleaned prior to flight.
L11 The surface of the ion Langmuir probe will be gold (TBR) coated.

**Science Instrument Interface**

SI1 The science Instruments will interface to the rest of the spacecraft using a single SPI interface
SI2 The maximum sampling rate for any science data channel shall be 100 Hz
SI3 Science preamplifiers must be located as close as possible to the sensor mechanical interfaces.
SI4 Digital control lines may not be routed through the areas allocated to the Electric Field Instrument, Science Magnetometer, and Ion Langmuir probe.

**Derived Instrument Requirements**

**Electric Field Instrument**

ED1 The input voltage range of the Electric field instrument shall be ±8V for a 10 meter tip to tip boom length. This requirement is derived from E1, E3, E13 and includes a 30% margin.
ED2 The telemetry rate for the V12 and V34 channels shall be 80Hz giving a spatial resolution of 93m at 350km altitude and 96m at 800km. This requirement is derived from SO3, SO4, SI2.
ED3 The input bias current of the instrument shall be lower than 1pA and input impedance more than \(10^{13}\) Ω. This requirement is derived from E8 and E12
ED4 The frequency response for the channels V12 and V34 shall be a DC low-pass response with a near linear phase response within the pass band and with at least a 20 dB/decade roll-off for out of band signals. The 3 dB cutoff frequency shall be 35 Hz. This requirement is derived from ED2.
ED5  The electric field instrument response shall be calibrated over the temperature range of -10 to 30 C using a polynomial. This requirement is derived from E1 and E9

ED6  The temperature of the electric field instrument electronics shall be known to within 0.2 C while collecting science data and telemetered at \(\geq 17\) mHz (TBR). This requirement is derived from ED5

ED7  The electric field spectrometer shall consist of four channels which cover the spectral ranges of 16-32 Hz, 32-64 Hz, 64-128 Hz and 128-512 Hz. The gain for these channels shall be 500 (TBR). This requirement is derived from SO5 and E15

ED8  The electric field spectrometer shall employ a high pass filter to reduce the VxB spin induced signal. The roll off of the filters shall be at least 60dB per decade for the out of band signals with a cutoff frequency of no more than 12 Hz.

ED9  The insulation on the electric field wire booms shall have a volume resistively of more than \(10^{17}\) Ω/cm.

**Ion Langmuir Probe**

LD1  The input current range of the ion Langmuir probe shall be from 700 pA to 7uA referenced as positive current to the probe surface from the space environment. This operating range shall be quantized with at least 16 bits. The operating range is calculated for a 1.9 cm diameter spherical sensor. This requirement is derived from L1 and L5.

LD2  The telemetered rate for the ILP1 and ILP2 data shall be 80Hz. This rate results in a spatial resolution of 93m at 350km altitude and 96m at 800km. This requirement is derived from SO3, SO4, SI2.

LD3  The frequency response for the channels ILP1 and ILP2 shall be a DC low-pass response with a near linear phase response within the pass band and with at least a 60 dB/decade roll-off for out of band signals. The 3 dB cutoff frequency shall be 35 Hz. This requirement is derived from LD2

LD4  The ion Langmuir probe response shall be calibrated over the range of -10 to 30 C using a polynomial. This requirement is derived from L6

LD5  The ion Langmuir probe electronics temperature shall be known to within 0.2 C while collecting science data and telemetered at \(\geq 17\) mHz (TBR). This requirement is derived from LD4

**Magnetic Field Science Instrument**

MD1  The science magnetometer sensor head shall be kept at least 15 cm (TBR) away from the power supply conditioning electronics. This requirement is derived from M1 and M3.
MD2  The telemetered rate for the Magnetic Field Instrument shall be 80Hz giving a spatial resolution of 93m at 350km altitude and 96m at 800km. This requirement is derived from SO3, SO4, SI2.

MD3  The Magnetic Field Instrument response shall be calibrated over the range of -10 to 30 C (TBR) using a polynomial. This requirement is derived from M6.

MD4  The Magnetic Field Instrument electronics temperature shall be known to within 0.2 C while collecting science data and telemetered at ≥ 17 mHz (TBR). This requirement is derived from MD3.
Appendix B

Deployment Requirements

DR1  The boom deployment rate shall not exceed 1 cm/s.

DR2  The rotational velocity to initiate boom deployment shall be a minimum of 1.6 Hz.

DR3  The maximum centrifugal force contributed from all four sensors during deployment shall not exceed 80% of the braking force.

DR4  The braking force shall be greater than 1.2 N with a goal of 2.5 N.
Appendix C

Letter to New Scale Technologies

This letter was written to request permission to use certain figures contained in this report. This letter, send via email was sent on November 29, 2012, no response has yet been received.

Dear New Scale Technologies,

I am a student at Utah State University finishing my Master’s Degree in mechanical engineering. In October 2011 Utah State University and the Space Dynamics Lab in Logan, Utah in conjunction with NASA and Cal Poly launched two small satellites into orbit that are currently making measurements and collecting data that will help scientist better understand certain phenomena in the Earth's upper atmosphere.

The development of these satellites began in 2009 and continued until launch in 2011. A crucial component to the satellites' design was one of New Scale Technologies' Squiggle motors. Each satellite contained an M3-L module that served as an actuator for the deployment of 4 sensors used for these scientific measurements. Our mechanism development actually began with the use of the Squiggle 1.8 and 1.8 RV motors but when we learned about the release of the M3-L unit and its capabilities we decided to use it. The M3-L unit simplified not only the mechanical design of our mechanism but also the electronics and control programming.

I am currently finishing a thesis as partial fulfillment of my degree which discusses the development of this sensor deployment mechanism that was designed around New Scale Tech's squiggle motors. Naturally there are sections of this report that highlight the use of this motor and pictures and diagrams are helpful in describing the design. I have chosen a number of pictures that I would like to include in this Thesis but it would only be right to ask permission to use them. These are pictures that are found both on New Scale Tech's website and in the Squiggle motor data sheets. I am writing this email to ask for written permission to use these pictures and diagrams. If you would prefer a more formal written request please let me know I will be happy to submit one. If permission is granted to use these items New Scale Technologies will be properly referenced both within the paper as well as in the references section of the report.

Thank you so much for your time and consideration.

Sincerely,

-Keith Bradford