Mechanical Subsystem Development for the CanX-7 Nanosatellite, the NEMO-HD Microsatellite, and the XPOD Mass Dummy

by

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A thesis submitted in conformity with the requirements for the degree of Masters of Applied Science
Graduate Department of Aerospace Engineering
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Abstract

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The proliferation of small satellites has ushered in new challenges in satellite mechanical development. The pursuit for accessibility and low cost has resulted in industry-wide implementation of standardized form factors. Yet, advances in payload technology continue to drive demands for new bus designs with significant growth potential. The Space Flight Laboratory is currently developing satellites across a wide spectrum of mission scopes. Amongst these are a challenging technology demonstration nanosatellite with a 3U CubeSat form factor, and an Earth observation microsatellite to fly with a next-generation large-scale platform. Presented herein are the advances made in mechanical engineering for these two satellites in particular, as well as other ground support equipment. Significant work is done towards the detailed design and analysis of the bus structures. As well, contributions are made in support of the assembly, integration, and testing activities on both the satellite and subsystem levels.
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To The Glory Of God. Amen.
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Chapter 1

Introduction

The proliferation of small satellites has been significant in the past decade and a half [36]. A large number of programs in the micro (<100 kg) and nano (<10 kg) categories are now being developed by universities and industry-based entities alike. The Space Flight Laboratory (SFL) at the University of Toronto Institute for Aerospace Studies (UTIAS) is actively developing micro and nanosatellite across a wide range of mission scopes. Of these, the CanX-7 (Canadian Advanced Nanospace eXperiment) and NEMO-HD (Nanosatellite for Earth Monitoring and Observation - High Definition) are at the two ends of the spectrum in terms of size and capabilities. The former, at 100 mm x 100 mm x 340.5 mm and a mass of 3.7 kg, is a technology demonstrator with a primary objective to deploy a passive de-orbit technology for nanosatellites [7]. The other, a microsatellite measuring 600 mm x 600 mm x 300 mm, and weighs 65 kg, is an Earth observation platform that will provide optical and spectral images and videos in very high resolutions [37]. In light of these are the on-going mechanical developments particular to these programs. The CanX-7 is a milk carton sized nanosatellite designed to accommodate new payloads while building on proven heritage and leveraging current subsystem technologies. NEMO-HD is a newly developed large-scale platform designed to cater towards more ambitious and larger flagship missions of the future. Concurrently, SFL continues to evolve its support infrastructure including the SFL-developed family of XPOD (eXoadaptable PyrOless Deployer) systems, to introduce program efficiencies wherever possible.

1.1 Perspective on Microspace

The catalysts for the growth of microspace programs are well documented [1, 38]. The growth in highly reliable, inexpensive, and miniaturized Commercial Off-The-Shelf (COTS) parts has pushed the boundary for increased functionality and reliability, delivered within reduced ensemble complexity and in smaller form factors [41]. Traditional military-grade hardware including S-Class components have been equaled or bettered by civilian technology capable of achieving market competitiveness and productivity [17, 45]. Manufacturing approaches including the use of rapid prototyping are enabling much cheaper and rapid fabrication, while allowing ambitious design possibilities not achievable with traditional machining methods [23]. Finally, standardization such as the CubeSat initiative introduced universal formats that are now ubiquitous amongst nanosatellite builders [9].
There is now a thriving community and industry around the development and operation of small satellites. Educational institutions spearhead many of these initiatives to mentor students in capacities previously reserved for those fortunate enough to enter the Space field. Industry participation has also seen exponential growth with prospects of achieving access to space on smaller scales and at reduced risks than with traditional commercial and government avenues that remain prohibitively expensive and sparse. The microspace initiative has become a disruptive technology by displacing certain limitations associated with big-space programs and enabling technologies that would otherwise be difficult or impossible without the initial exposure through Space-based platforms. An ever-increasing demand for cost-effective, responsive Space is now forecasted for the near future.

To support these small satellite programs, mechanical developments in the field have evolved in almost lockstep fashion with each succeeding mission. Existing satellite buses are continually optimized to support the wide breadth of dissimilar payloads, evolving mission parameters, and past lessons learned, ensuring that almost no two structures remain identical. New bus form factors are also constantly being developed to accommodate newer hardware as technologies are proven and confidence grows. As payload capabilities mature it is often accompanied by a rise in their need for bus-side resources. Pragmatically this translates into larger volume and mass allocations, and even indirectly such as an increase in power demands, which drives greater solar cell quantities and, in turn, solar panel dimensions.

1.2 The Space Flight Laboratory

The Space Flight Laboratory (SFL) was founded in 1998 under the directorship of Dr. Robert E. Zee at the University of Toronto Institute for Aerospace Studies. SFL is a vertically integrated entity with end-to-end development capabilities, which is foundational in its success in achieving the breadth of its many missions and capabilities. In 2003, the MOST (Microvariability and Oscillations of STars) microsatellite, Canada’s first space telescope and jointly developed by SFL, was successfully built and launched. It was followed in 2007 by CanX-2, a 3U CubeSat platform that validated many of the SFL-developed technologies for use in nanosatellites. The CanX (Canadian Advanced Nanospace eXperiment) program is a highly successful initiative to provide cost-effective access to space for research and development programs in and outside of SFL. Present on CanX-2 were a novel cold gas system, on-board computers, communication systems, and attitude determination and control software and hardware.
Chapter 1. Introduction

To build upon the initial pioneering missions, SFL took to develop an adaptable, modular bus platform with greater growth capabilities than those offered by the CubeSats. The result was the GNB (Generic Nanosatellite Bus) which is a 200 mm x 200 mm x 200 mm form factor with a mass of around 8kg [32]. Significantly more capable, the GNB is a platform with much-expanded capacities and is the basis for many of the SFL’s on-orbit successes, including: NTS, AIISSAT-1, and AIISSAT-2 (on-orbit ship tracking) [19, 31], the BRITE Constellation of satellites (space-based astronomy) [22], and CanX-4 and 5 (active nanosatellite formation flying). Further evolutions of the GNB resulted in the still larger Nanosatellite for Earth Monitoring and Observation (NEMO) bus, with the same frontal footprint and double the length. It employs many of the GNB’s subsystem architecture, hardware, and software, and affords a large increase in payload volume and generated power. The NEMO family has joined the range of next-generation form factors for a number of future missions, including the NEMO-AM and GHGSat-D missions for evaluating terrestrial aerosol and greenhouse gases, and the NORSAT-1 next-generation Maritime traffic monitoring platform.

The CanX-7 and NEMO-HD are the next steps in the evolution of satellite buses at SFL, albeit in very different ways. By returning to a 3U form factor, the CanX-7 is able to build upon the familiarity and experience gained from the CanX-2 mission. Since 2008, there have been many advances in both software and hardware technologies, which are now incorporated to arrive at a much more capable satellite. CanX-7 is the quintessence in aggressive utilization to achieve significant payload potential within a limited bus volume at a very low cost. The NEMO-HD in contrast is in the microsatellite territory with an ambitious capability of accommodating very large payloads with extreme resource demands and performance requirements. With NEMO-HD, SFL continues to push its structure’s know-how in areas such as the use of honeycomb sandwich panels as its primary structural material in lieu of machined alloy structure. The investments in its development will therefore form a new foundation for future design possibilities.

1.3 Motivations

This thesis describes the author’s contributions toward the CanX-7 and NEMO-HD satellites during phases C and D of their program developments. Work on the CanX-7 satellite saw its mechanical subsystem mature from post-PDR configurations to detailed design and analysis. The manufacturing and testing of satellite structure have been completed together with the development of the satellite’s assembly processes. Finally, GSE (Ground Support Equipment) has been developed to support the integration efforts and also the various system-level testings. In contrast, the work done in support of the NEMO-HD program focused primarily on advanced structural analysis to validate the mechanical performances of the satellite bus. This included analysis on localized assemblies and complex, full satellite validation done with finite element techniques. Finally, the author was instrumental in the development of the XPOD Mass Dummy, which benefited the XPOD program by achieving significant cost savings and risk reductions during launch campaign. It is hoped that the knowledge gained here may continue to foster the foundational capabilities at the Space Flight Laboratory; as well, that the scope of work presented herein helps to shed a small light on the facet of mechanical engineering particular to the field of microspace.
Chapter 2

Canadian Advanced Nanospace eXperiment - 7

CanX-7 is SFL’s second satellite that utilizes the 3U CubeSat bus form factor. Its sole precursor is the CanX-2, which was launched in April 2008 and remains fully operational to date. Refer to Figure 2.2. As of writing, CanX-7 is the smallest satellite under development at SFL. Nevertheless, it is a technology demonstrator whose objectives are by all means ambitious, including:

- To demonstrate a modular, bolt-on drag sail technology to achieve passive de-orbiting of micro- and nanosatellites from LEO (Low Earth Orbit).
- To validate the on-orbit use of a space-based ADS-B (Automatic Dependent Surveillance - Broadcast) aircraft tracking technology on the nanosatellite scale.
- To employ a SFL-developed miniature VGA Inspection Camera (mVIC) imager with the primary goal of the visual inspection of drag sail performances.

The primary payload on CanX-7 is a set of four sail modules developed at SFL, each of which deploys a 1.0 m$^2$ trapezoidal film of Upilex. Refer to Figure 2.1. When deployed together the sails induce sufficient overall surface area to induce atmospheric drag to effect orbital decay and re-entry [7]. In light of the growing concerns for space debris in Low earth Orbit (LEO), the Inter-Agency Space Debris Coordination Committee (IADC) has suggested de-orbit guidelines for all future satellite missions [3, 21]. The SFL takes its responsibility towards Space environment stewardship very seriously. The successful on-orbit demonstration of the drag sails will grant space heritage to this technology for use on future micro- and nanosatellites. The secondary payload on-board CanX-7 is an ADS-B receiver, developed jointly with the Royal Military College (RMC) of Canada. ADS-B is a GPS technology that is gradually replacing traditional ground-based radar as the primary means in tracking aircraft. ADS-B is particularly superior where terrestrial radar networks are ineffective or non-existent such as over irregular terrain features and vast oceanic routes, or incongruous as over political borders. Space-based ADS-B overcomes these limitations and are already being fielded in newer generations of SATCOM satellites [30]. CanX-7 will demonstrate cost-effective, responsive employment of ADS-B on a nanosatellite scale. Lastly, the CanX-7 will carry the mVIC, a SFL-developed technology comprised of three miniature COTS camera
The mVIC will be used to inspect and validate the sail’s on-orbit performance, amongst other scientific activities [33].

Figure 2.1: CanX-7 Satellite Showing Deployed Sails (L) and Close Up Of Satellite Bus

The definitive design of the CanX-7 structure has evolved into a highly capable payload platform. What has transpired is a bus with an approximate 35% of continuous usable payload volume, greater than a single 1U nanosatellite, attributed to the primary and secondary payloads located on one end of the bus. The bus itself possesses an advanced two-axis stabilized attitude determination and control capability using state-of-the-art magnetometer and magnetorquer technology with ±3° accuracy tracking the local magnetic field. A centralized power distribution unit system is installed, which through its body mounted solar cells is capable of producing approximately 6 W of generated solar power, and can store approximately 18.75 Wh using a single Lithium Ion battery. The communication suite on-board consists of a UHF receiver and provides a 4 kbps command uplink, while an S-Band transmitter provides downlink at between 32 kbps and 1 Mbps. The command and data handling is achieved with a single GNB on-board computer, with storage capacity of up to 1 GB and processor memory of 1.5 MB, all of this in a bus that has a passive thermal capability that is capable of maintaining system-wide operational temperatures of between +60°C and −20°C. In other words SFL has produced a highly capable and modularized 3U bus platform that, with minor customization, has considerable payload flexibility and accommodation capacities.

The author’s contributions to the development of the CanX-7 bus structure span several aspects. The design of the bus and its internal subsystem components were finalized followed by detailed structural analysis of the satellite bus. As well, significant work has been done towards the assembly and testing of the satellite and its subsystem components. This includes the design of an optimized integration approach, and the development of ground support hardware of various functions.
2.1 Design

The CanX-7 has a geometry of 100 mm x 100 mm x 340.5 mm, and a mass of 3.7 kg. Much of the preliminary mechanical development work on the CanX-7 was accomplished by Fiona Singarayar [35], from initial concepts to post-PDR (Preliminary Design Review) time frame. The author’s work as presented herein builds upon this foundation, bringing the CanX-7 mechanical subsystem through CDR (Critical Design Review) and to its present state.

The general layout of the CanX-7 bus is compatible with the specifications for a 3U CubeSat form factor [9]. This consists of four launch rails along the edges of the rectangular prism with cross braces to form a trussed structure and body panels for covering and to carry shear loads. The launch rails interface with the satellite’s deployer unit, in this case SFL’s XPOD (eXoadaptable PyrOless Deployer), which in turn interfaces with the launch vehicle.

Refer to Figure 2.3; CanX-7’s primary structure is designed around the +Z and −Z trays, which are the main elements that support the majority of the subsystem components. The interior of the +Z tray accommodates the UHF receiver and the S-Band transmitter of the communication subsystem, and also the battery and the BCDR (Battery Charge-Discharge Regulator). Two magnetic reed switches used for satellite deployment indications are also mounted on the inside of the +Z tray, while its exterior supports two sets of solar cell coupons and two deployable UHF antennas. The interior of the −Z tray supports the PDU (Power Distribution Unit), the OBC (On-Board Computer), Z-magnetorquer, and the satellite’s test port. The −Z tray has one solar cell coupon on its exterior, the two other deployable UHF antennas, and also the deployable imager boom. The +Z and −Z trays are joined by the +X and −X panels, which are the two longitudinal body panels. They are distinct such that the +X panel is longer, with the X-magnetorquer on the interior and two solar cell coupons on the exterior, and the −X panel is shorter to make room for the secondary payload enclosure, and supports just one solar cell coupon on its exterior. The +Y and −Y panels cap the small ends of the satellite bus and both support one solar cell coupon on their exteriors. The −Y panel also supports the Y-magnetorquer mounted to its interior. In addition, the distinct primary payload stack consists of two structural mounting plates and four sail modules. The secondary payload is also a separate entity with an electronics enclosure and an exterior surface-mounted patch antenna. These are adjacent to each other and are located by the +Y end of the bus.
2.1.1 Structural Optimizations

Mass Reductions

One of author’s initial efforts in optimizing the CanX-7 structure was to reduce its total satellite mass fraction. The material of the primary structure had been changed from magnesium AZ31B-H24 to aluminum 6061-T6 for thermal reasons, as the aluminum’s greater specific thermal conductivity was advantageous. The resulting increase in material density from 1740 g/m$^3$ to 2700 g/m$^3$ was significant and warranted mass reduction wherever possible. The author evaluated all structural elements to aggressively reduce material margins, including reductions in stiffener dimensions, panel and wall thicknesses, and extraneous geometries that were artifacts of previous design iterations. Refer to Figure 2.4 and Figure 2.5.

The author also took the opportunity to address manufacturing considerations of the structural components. The solid model for the CanX-7 satellite at the time had undergone numerous design alterations,
and many of its component details were haphazard throughout. These were standardized to eliminate unnecessary stand-alone features. Hole geometries were also scrutinized to ensure they conform with established industry standards for drill sizes and depths, which are particularly important for tapped holes meant to accept threaded fasteners and helicoils. Corner fillets and mounting bosses geometries were made uniform to reduce tool changes during fabrication. These efforts contributed towards reducing potential confusions and errors during manufacturing. The pursuit of general cleanliness and consistency amongst part details is also a good design practice.

As evident in Table 2.1, the total primary structural mass of the spacecraft decreased 7% from 767 g at post-PDR to 711 g at present, normalized for magnesium density. This directly translates into just under 90 g of mass savings with the current aluminum frame, a not-negligible amount for a nanosatellite structure. This is evidence for how the CanX-7 structural mass was optimized by pursuing design efficiencies and keeping structural elements only where needed.
### Table 2.1: Primary Structure Mass Reduction Summary

<table>
<thead>
<tr>
<th>Structure</th>
<th>Post-PDR Mass (g)*</th>
<th>Current Mass (g)*</th>
<th>Current Mass (g)**</th>
<th>Delta***</th>
</tr>
</thead>
<tbody>
<tr>
<td>+X Panel</td>
<td>94</td>
<td>93</td>
<td>145</td>
<td>-1%</td>
</tr>
<tr>
<td>-X Panel</td>
<td>59</td>
<td>59</td>
<td>92</td>
<td>0%</td>
</tr>
<tr>
<td>+Y Panel</td>
<td>35</td>
<td>33</td>
<td>51</td>
<td>-6%</td>
</tr>
<tr>
<td>-Y Panel</td>
<td>34</td>
<td>34</td>
<td>53</td>
<td>0%</td>
</tr>
<tr>
<td>+Z Tray</td>
<td>252</td>
<td>231</td>
<td>359</td>
<td>-8%</td>
</tr>
<tr>
<td>-Z Tray</td>
<td>208</td>
<td>180</td>
<td>280</td>
<td>-13%</td>
</tr>
<tr>
<td>UHF Enclosure</td>
<td>85</td>
<td>80</td>
<td>124</td>
<td>-6%</td>
</tr>
<tr>
<td><strong>Total</strong></td>
<td>767</td>
<td>711</td>
<td>1,104</td>
<td>-7%</td>
</tr>
</tbody>
</table>

*Magnesium AZ31B-H24  
**Aluminum 6061-T6  
***Normalized for magnesium

#### Secondary Payload Enclosure

The secondary payload enclosure on CanX-7 is an independent unit that supports the ADS-B payload electronics and antenna. The electronics consist of a COTS (Commercial Off The Shelf) board assembly measuring 88 mm x 53 mm and with a mass of 60 g, a RS-486 circuit board developed by SFL, and a pre-amp unit for the antenna. The antenna is of a flush-mounted patch design with a footprint of 75 mm x 75 mm, and is bonded to the exterior surface of the enclosure’s cantilevered panel. Finally, a 9-pin micro-D connector is present, which is the data and power interface to the satellite.

Refer to Figure 2.6; the design of the secondary payload enclosure was complex due to several factors. The primary flight electronic of the payload is a commercial Beaglebone stack. Its longest dimension is 88 mm, which needs to fit inside the enclosure whose length along that axis could not exceed 91.5 mm. This was constrained by the satellite’s exterior face on one side and the clearance needed from the +X panel magnetorquer on the opposite side. This resulted in minimum wall thicknesses of down to 1 mm at places that needed to be selectively reinforced to satisfy strength and natural frequency requirements. The breakdown of the enclosure also proved to be challenging. The ideal assembly process would be to
integrate the compartment into the satellite first followed by the outer antenna panel to allow access and visual inspection of the interior components. However, the configuration of the payload, including the presence of a rigid coax cable between the antenna and the electronics, meant the exterior panel must be secured before integration with the satellite. This necessitated distributing the enclosure fasteners in such a way to allow the desired assembly procedure to be carried out. Finally, to reduce machining costs the various features in and outside of the enclosure had to be kept to a minimum which invoked some creativity.

2.1.2 Component Design and Placements

The CanX-7 inherits a number of the hardware concepts from its CanX-2 predecessor. This includes the deployable UHF monopole antennas and the magnetometer boom. The CanX-7 also benefits from much recent GNB-era subsystem hardware and technologies including the OBC (On-Board Computer), communications suits, attitude control subsystem, and the power system components. The use of ready-made parts augmented the CanX-7 development progress by enabling a compressed manufacturing schedule while achieving significant capabilities and flexibilities with matured hardware. Mechanically this also helped to solidify design definitions early on, albeit with a bit of thoughtfulness integrating into a different bus form factor.

Power System Interface Board

The CanX-7’s PDU (Power Distribution Unit), shown in Figure 2.7, is derived from SFL’s MPS (Modular Power System) developed for larger buses [8], taking elements including two μSPN (Micro Switch Power Node) and one IFN (Interface Node) cards. The μSPN cards distribute power to the various loads in the spacecraft, provide over-current protection, and monitor consumption of each load. The IFN card on the other hand provides firecode detection and system reset functionality, solar array interfacing and temperature telemetry. To configure and adopt these for low-power use, a POSIF (Power System Interface) backplane unique to the CanX-7 was designed to integrate the PDU cards into one system. The design of the POSIF board was therefore critical, both mechanically and electrically, a process that the author worked closely to achieve alongside SFL’s engineers. Specifically, the author was responsible for the mechanical aspects of the POSIF design including geometry, mounting, and connector placements.
The design of the POSIF board was a challenge given the internal space constraints of the then-relatively mature interior layout. Refer to Figure 2.8. The configuration of the PDU cards within the CanX-7 are such that they mount on the –Z tray and underneath the OBC board. The connectors on the PDU cards are situated such that the POSIF board must be designed to mount orthogonally to them and parallel with the interior of the +X panel. The resulting available space was limited given the proximity of the bulky battery enclosure and the radio enclosures opposite to the OBC board. Working with these constraints, the author ensured that the POSIF geometry adequately encompassed the PDU cards and had sufficient surface area for board-mounted connectors and printed circuit routes. The definitive design of the POSIF board including its distinctive L-shaped footprint was the result.

**Printed Magnetorquers**

The design and definition of the printed magnetorquers component was another that the author contributed towards. The CanX-7 ADCS (Attitude Determination and Control System) includes three powered magnetorquers mounted mutually orthogonal within the satellite bus, i.e., one for each X, Y, and Z-axis; refer to Figure 2.9. These magnetorquers are essentially a set of coils that generate a magnetic field when current is applied, and function by interacting with the Earth’s magnetic field. The CanX-7 magnetorquers are ‘smart torquer’ technology that embeds the control circuit with the coils using PCB (Printed Circuit Board) technology, in lieu of external electronics [10].
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Figure 2.10: Mechanical Mock-Up Showing Tight Magnetorquer Clearances with Surrounding Structures

The geometry design of the printed magnetorquer was influenced by several factors, as shown in Figure 2.10. To achieve adequate performance, a minimum number of coil turns and lengths were required in the PCB. To achieve commonality, all three magnetorquers needed to be identical. Further complications were brought about by their placement locations, which must accommodate existing structures that were already defined. The X-magnetorquer went between the +X panel and the secondary payload enclosure, which constrained the maximum thickness and width of the magnetorquer. The Y-magnetorquer must be low in profile to clear the radio enclosure and the significant amount of wiring in the –Y end of the bus. The Z-magnetorquer was placed adjacent to several mounting bosses for the power and computer boards, which limited the length and minimum interior width and radius of the PCB. The author was able to achieve an optimized solution by iteratively working through the magnetorquer sizing with the ADCS engineer to meet performance needs, and also altering the satellite bus structure wherever possible to achieve positive clearances.

2.1.3 Wiring Harness Development

The design of the CanX-7 wiring harness constituted a critical aspect of the satellite’s development. Its objectives were to:

- Optimize the harness layout for assembly efficiency during satellite integration
- Validate connector assignments on electrical components
- Minimize complexity and points of failure by reducing connector counts, wire lengths, and routes
- Ensure adequate wiring support and connector strain relief

Wiring Architecture

The CanX-7 harness design achieves these objectives by attributing the majority of the wiring onto a master harness attached to the PDU. This centralizes the wiring so as to reduce the quantity of
separate harnesses. As designed, the CanX-7 power and data wires originate from the two μSPN boards of the PDU and the OBC, respectively. As both the PDU and OBC are located on the –Z tray, this effectively centralizes the majority of the harness and reduces in-line connectors. From the PDU and OBC, the wirings branch out to connectors a majority of which mate directly to their respective subsystem components. The result is a simplified harness architecture that enables a ‘plug and play’ approach to satellite assembly. This allows most components and parts to be pre-assembled into discrete sub-assemblies and brought together only at the last moment, maximizing access before and minimizing exposure time during bus integration when damage risks due to handling is high.

Figure 2.11: Simplified CanX-7 Wiring Architecture

Figure 2.11 is a simplified layout of the CanX-7 wiring architecture. Each path represents a set of wires terminating at subsystem components. This diagram is also representative of the harness paths of the final design.

Connectors

Several different connector types are used in CanX-7, as shown in Figure 2.12. The diversity is primarily a result of mechanical decisions. This includes considerations such as the need for repeated cycling without performance degradation, and also size and geometries where volume and access limitations dictate. Connectors that are present in CanX-7 include:
• Hirose DF-11 Series: This is the preferred connector on SFL electronics, having achieved flight heritage since the CanX-2 spacecraft. The DF-11 has gold-plated metal contacts that are crimped onto wires and inserted into plastic housings. DF-11s are simple, reliable, inexpensive, are easy to work with, and prevent mis-insertion with handed tabs on the housings. The DF-11s are used extensively within the CanX-7, populating most of the subsystem PCBs and constitute a majority of the wiring harness connections.

• Omnetics Polarized Nano Series: These are ultra-small connectors that are used where size and space are a premium. On CanX-7 these are limited to the sail module payload stack and the miniature camera payload in the imager boom. The electronic components of both payloads are extremely small and the harness routing passageways and feed-thrus are limited in geometries. The use of polarized nano series is needed to fit and navigate the available volume within the structure and to connect to the PCBs.

• Glenair Micro-D M83513 Series: This series of connectors conforms to military standard MIL-DTL-83513 for quality control and reliability. The Micro-D connectors are durable, high-precision connectors used where a high number of cycling is anticipated. On CanX-7 a 15-pin Micro-D is used for the test port, which sees a significant number of mechanical cycles during the assembly, integration, and testing phases. Although not expected to see significant cycling once installed into the bus, the secondary payload also employs a 9-pin Micro-D connector. This allows it to undergo thorough bench top testing as a stand-alone unit prior to satellite integration.

• Harwin L-Tek Datamate M80 Series: A high reliability connector series, the Harwin L-Tek is preferred for high vibration and temperature environments. The connectors also have metal clasps to prevent from coming undone. On CanX-7, these are chosen for the +Y panel solar cell coupon and thermistor connectors. They replace the DF-11 connectors used throughout the bus given their low profile, which is needed to nestle within the sail module adapter plate cavity and navigate through the sail stack down into the bus.

Figure 2.12: CanX-7 Connectors (L-R): DF-11, Polarized Nano, Micro-D, L-Tek M80s
Design

The definitive design of the CanX-7 wiring harness consists of:

- Separation Switch Harness
- Test Port Harness
- Imager Boom Harness
- BCDR (Battery Charge-Discharge Regulator) Harness
- +Y Panel Harness
- Primary Bus Harness

Most of these connect between two subsystem components. This BCDR harness, for instance, connects the battery/BCDR unit to the POSIF backplane of the PDU. As well, the +Y panel harness connects the solar cell coupon and the +Y panel thermistor directly to the IFN board. Some of these also constitute part of a separate assembly that is attached to the bus during satellite integration. This includes the imager boom harness that connects the mVIC camera and magnetometer units in the imager boom to the primary bus harness via a single 8-pin DF-11 connector. Similarly the primary payload wiring harness is integral to the sail modules stack and interfaces with the primary bus harness during payload integration via a 12-pin polarized nano connector. Figure 2.13 shows the primary bus harness, the +Y panel harness, and the BCDR harness. Not shown are the separation switch and the test port harnesses, which are on components mounted to the bus –Z panel, and also the imager boom harness and primary payload harness, which are part of their respective sub-assemblies.

Figure 2.13: CanX-7 Wiring Harness Mock-Up

The CanX-7 wiring harness design was carried out by the author using a full-scale mock-up of the structure (ref Section 2.3.3). The use of a physical mock-up benefited the harness design in several ways. Wiring harness design has traditionally been done digitally using 3D computer modeling, which has
certain limitations. This includes hard to capture physical aspects including harness twists and bending stiffness. Likewise the ergonomics of physically handling the harnesses, connectors, and tie-downs cannot be practiced. The use of a mock-up augmented the computerized design process by catering to these limitations. As the CanX-7 mock-up was of sufficiently high fidelity, it further allowed the installations of dummy mechanical hardware including spare PCBs and other components. This resulted in a very comprehensive model that permitted the wiring harness to be designed in conjunction with the bus integration procedures.

Figure 2.14: Primary Bus Harness assembly to +Z and -Z Trays During Bus Integration

Dry-runs of the satellite integration were conducted as shown in Figure 2.14. This shows the +Z and -Z tray sub-assemblies integrated and just prior to the start of bus integration. Notice the ergonomics of the wiring harness design in relation to the bus trays and subsystem components. As the primary bus harness is pre-integrated to the PDU and OBC, much of the harness routing can be done at the -Z tray sub-assembly level, taking advantage of the PCB metal standoffs for wires tie-downs. The radio harness branches off from the primary bus harness, and is seen installed to the S-Band and UHF components on the +Z tray. The cleanliness and simplicity of this layout is evident. From this point the bus assembly proceeds rapidly with minimal wiring connections remaining. A comprehensive discussion of the satellite assembly procedure is found in Section 2.3.3 herein.
2.2 Structural Analysis

The author conducted the structural analysis of the CanX-7 bus using the FE (Finite Element) analysis method. The CanX-7 solid model assembly was imported from its Solid Edge format into Siemens’ NX8 software for meshing. Analysis and post-processing of the FE results were also done in NX8 with the NX Nastran solver. Finally the results were gaged against program requirements to ensure a satisfactory design. This section provides an in-depth exposition of this work.

2.2.1 Modeling

Structural Elements

All of the CanX-7’s structural components were modeled using 3D hexahedron elements, using either the 8-node Hex-8 element or the 20-node Hex-20 element. Hexahedron elements were used throughout as they are preferred over the 4-node tet-4 and 10-node tet-10 tetrahedrons. Linear triangular and tetrahedron elements are prone to being overly stiff resulting in drastically inaccurate deformation and stress predictions [16]. Better performing second-order triangular elements can be used with quite acceptable results, but they are also much more computationally expensive and not preferred for large assemblies. Further, element shape and distributions for triangles and tetrahedron are much more prone to random distributions leading to a higher number of poorly shaped elements such as those with undesirable high aspect ratios. Figure 2.16 is an example of the quality differences seen using auto-generated tetrahedron vs. manually defined hexahedron elements on the CanX-7 launch rails.

Figure 2.16: Poor Auto-Mesh (L) and Manually Swept-Meshed (R) of CanX-7 Launch Rails
Tetrahedron elements are used, however, under very select circumstances, one of which is to exploit their capabilities in defining complex features and geometries that are difficult or impossible to accommodate with hexahedron elements. Tetrahedron elements are also associated with auto meshing, which is used when expedited meshing of a part is required. Often this is in support of detailed local analysis of a part where the corresponding solver time is not prohibitive. For instance, this was used to good effect in validating the quality of the manually meshed parts. Mesh verification can be done using built-in software checks to assess certain critical parameters. Alternatively, it can also be achieved through a series of convergence tests. This involves evaluating a number of dissimilar meshes of the same part to check for the convergence of results when subjected to the same boundary conditions. This is a standard FE methodology and is a good gage on the quality of a particular mesh in relation with others. The author used this technique to validate the fidelity of a number of structural meshes prior to integrating them into the assembly FE model. For a comprehensive exposition of these fundamental FE concepts, the author encourages the reader to reference the Introductory and Advanced Finite Element courses given by Dr. Carlos A. Felippa of the Aerospace Engineering Department at the University of Colorado-Boulder. Additionally, the Advanced Aerospace Structures course by Dr. Craig A. Steeves at UTIAS should be visited which provides the opportunity to code actual Finite Element solvers.

The CanX-7’s structural FE model also strikes a balance between the fidelity of the model and associated effort to achieve the mesh. Figure 2.18 shows the meshing process for the CanX-7 +X panel that uses 2D shell elements for the thin body panels and 3D hexahedrons for the structural beams and component mounting bosses. This is a four-step process involving the solid model, a simplified and discretized ‘idealized’ part model, a composite model with isolated 2D surfaces and 3D polygon bodies, and finally a meshed part with the appropriate 2D and 3D elements. The amount of manual work entailed in this...
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The process varies according to part complexities and feature geometries. Often it is much more desired to forgo the use of different element types and use exclusively 3D elements, for instance, to achieve drastic time and effort savings in manual meshing without suffering in accuracy of results.

Components and Connections

A majority of the subsystem hardware were created as individual unit-level meshes and imported into the assembly FE model. This includes the battery, printed circuit boards (OBC, power systems, radios), magnetorquers, magnetometers, test port connectors, and separation switches. Parts that are non-load bearing and whose geometries have no implications to the analysis results were represented as manually defined 0D lumped mass elements with spider elements to their mounting fastener locations. An example of this is the printed magnetorquer part mesh as shown in Figure 2.19.

One-dimensional RBE2 and RBE3 rigid spider elements are used to connect the 0D lumped masses to the nodes at the component fastener locations. RBE2 have infinite stiffness between the daughter nodes and does not interpolate and redistribute weighted loads that are manifested when a part deforms [28]. This is representative of rigid components such as compartments and enclosures that contribute in stiffness. RBE3 elements allow the daughter nodes to deform with respect to each other and are used where distributed loading between the element legs are needed, such as around screw holes where stresses are non-uniform due to fastener bearing. Hardware such as PCBs are also modeled using RBE3s between the lumped masses and fastener points to ensure no rigidity, and therefore loads are attributed through the fragile components. Fastener modeling specifically involved using RBE2 elements to model the fastener shanks and load paths between parts, in addition to using RBE3s for the screw heads. An alternative method was also used using beam elements to model the shank to allow information such as cross-sectional stresses to be extracted. These are shown as examples in Figure 2.20.
Assembly Models

The NX8 finite element environment provides the ability to generate models and assemblies in the same hierarchy as its solid model counterparts. This is beneficial in that component level meshes may be generated and imported for use in multiple instances under different assembly level meshes. The author made use of this feature to reduce the time and effort in building the assembly FE model, and to reduce modeling error by ensuring equal meshes for the same parts. Figure 2.21 is a visual representation of CanX-7 assembly FE meshes, both for the satellite and the +Z tray sub-assembly. Notice the three magnetorquers that are embedded in separate sub-assembly FE models but are the same part mesh.

2.2.2 Boundary Conditions

Loads

The CanX-7 bus is assessed against the following structural requirements [6]:

[CX7-MEC-8] The spacecraft shall be designed to withstand launch loads 25% in excess of those on the launch vehicle with the highest quasi-static (static + dynamic) accelerations

[CX7-MEC-9] When in their launch configuration, all spacecraft units shall have a minimum first natural frequency of 100 Hz in any axis

The reference launch loads for the CanX-7 structure analysis was based on that of the Boeing Delta II rocket. At the time of analysis the Boeing Delta II was the potential launch vehicle with the most severe launch load spectrum. Delta II has since been ruled out as a potential launch vehicle, however, given the lack of available launch slots. Nevertheless, its launch requirements continue to exceed those of future potential rockets.
The worst-case quasi-static load for the Boeing Delta II is 18.12G, or 22.65G with a 1.25 safety factor [6], along the worst-case axis and less for the other two. To be conservative the same 22.65G is applied to all three satellite axes simultaneously for a maximum composite load of 40G RSS (Root Sum Square). This ensures the satellite and its XPOD deployer are orientation-agnostic in their mounting on the LV, which is not absolute. Typical launchers would support an unspecified number of secondary payloads of various geometry and masses. These are mounted to the LV payload adapter ring as stipulated by the LV provider. It is therefore important to ensure that the satellite analysis covers all conceivable load cases and orientations. Lastly, a further safety factor of 1.25 was added on top to capture errors and uncertainties in the finite element modeling process. This brought the maximum acceleration along any axis to 29G and a RSS value of 50G. This last safety factor is a rule of thumb, as recommended in Space Mission Engineering [44], for passing structural designs by analysis only.

<table>
<thead>
<tr>
<th>Case</th>
<th>+X Axis</th>
<th>+Y Axis</th>
<th>+Z Axis</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>+29G</td>
<td>+29G</td>
<td>+29G</td>
</tr>
<tr>
<td>2</td>
<td>−29G</td>
<td>−29G</td>
<td>−29G</td>
</tr>
<tr>
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</tr>
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<tr>
<td>8</td>
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<td></td>
<td>−50G</td>
</tr>
</tbody>
</table>

Table 2.2: Quasi-Static Analysis Cases Applied Loadings

Table 2.2 outlines the eight different load cases used in the CanX-7 quasi-static analysis. Cases 1 and 2 were analyzed with positive and negative 29G loads applied simultaneously along all three axes, respectively. This was to simulate the composite load vector at an angle with respect to all three of the primary satellite body axes. Cases 3 through 8 were to apply the composite load vectors along the satellite primary axis in both positive and negative orientations. In addition to these inertial loads, a 130 N compressive loading is applied to the ends of all four launch rails on both ends of the satellite. These simulate the XPOD pre-load forces on the CanX-7 bus and were applied to all eight load cases.

Constraints

Refer to Figure 2.22; in the launch configuration the CanX-7 bus is firmly constrained in the satellite longitudinal axis (Y-axis) by the XPOD pusher plate on the satellite −Y side and the XPOD door on the +Y side. As previously alluded to, a 130 N pre-load is acted in the satellite longitudinal axis. This was by design and is imparted by the XPOD’s pre-load washer assembly acting through the pusher plate. The XPOD triple also has four internal Delrin rails corresponding to the four launch rails of the CanX-7. These constrain and support the bus laterally about the satellite’s X and Z-axes. The width of contact between the Delrin rails and the satellite launch rails is 7.5 mm out of the 8.5 mm width of CanX-7’s launch rails. In addition, the Delrin rails support only 292.5 mm of the entire 340.5 mm length of the satellite’s sides, with a constant cross section. The remainder are a tapered section that opens outwards towards the XPOD’s door. This provides an increased clearance as the satellite is ejected from the deployer.
Constraints were applied to the FE model to best simulate the interactions between the CanX-7 bus and the XPOD triple. A unique set of constraints was applied to each of the eight load cases outlined in Table 2.3. Altogether there were ten individual constraints that were selectively applied in combinations to reflect the contact supports between the XPOD interior and the satellite’s launch rails. Most of these consist of simply supported constraints to disable the out-of-plane nodal displacements while keeping the other DOF (Degree Of Freedom) free. Eight of these constraints are applicable to the launch rail longitudinal faces (two per rail) to simulate contact supports against the XPOD’s Delrin rails. The two remaining sets are applied to the +Y and Y ends of the launch rails to represent contacts against the XPOD pusher plate and the door. Shear relationship between the satellite and the XPOD was not modeled and effects due to friction were neglected. It was calculated that the inertial loads on the bus would overcome the friction between the aluminum satellite structure and the Delrin polymer.
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Composite applications of the constraints are outlined in Table 2.3. In analysis Case 1, for instance, a +29G inertial load is applied in each of the +X, +Y, and +Z directions simultaneously. Constraint set were therefore applied to the +Y face to simulate the satellite resting against the XPOD door, and on Rail B +X face, and Rail D +Z face to represent supports by the Delrin rails on the +X and +Z faces. Refer to Figure 2.23 for an illustration. Constraints on Rail A +X and +Z faces were ignored, however, for conservatism. In this manner the satellite was allowed to rotate about rails B and D, which allowed for potential bending of the satellite structure. Although conservative, this was not entirely inconceivable should the XPOD itself deform and become misaligned under launch loads.

<table>
<thead>
<tr>
<th>Case</th>
<th>Rail A +X Face</th>
<th>Rail A +Z Face</th>
<th>Rail B +X Face</th>
<th>Rail B -Z Face</th>
<th>Rail C -X Face</th>
<th>Rail C -Z Face</th>
<th>Rail D -X Face</th>
<th>Rail D +Z Face</th>
<th>+Y Rail Feet</th>
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Table 2.3: Simply Supported Constraint Applications by Analysis Case

2.2.3 Results

Quasi-Static

Quasi-static stress results are tabulated in Table 2.4. The quasi-static failure criterion is yield, therefore the Von Mises stress was used rather than Max Principal, which is generally suited for analyzing ultimate failure scenarios. As well, elemental-nodal stresses were observed instead of element stress. Elemental-nodal stress determines the nodal stress by averaging the adjacent elemental stresses at the
node location, and will produce higher stress contours than element stress, which is an average of the
element stress at the element nodes. In reality the actual stress should converge somewhere in between.
However for conservatism the higher of the two was taken. Safety margins are computed against the
tensile yield strength of aluminum 6061-T6, which is 276 MPa.

The highest stress is seen in load case 8, which was a local stress concentration around one of the
four fastener mounting holes of the UHF radio cover. The margin of safety in this case is 6.82 and
corresponds to a stress of 35.3 MPa, well below the material yield strength. The applicable load case
corresponds to an inertial load of 50G acting in the –Z direction. This resulted in the cover bowing away
from the UHF enclosure in the direction of the load.

<table>
<thead>
<tr>
<th>Case</th>
<th>Max Stress [MPa]</th>
<th>Safety Margins</th>
<th>Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>20.0</td>
<td>12.8</td>
<td>-Z Panel, +X Panel Mounting Fastener Hole</td>
</tr>
<tr>
<td>2</td>
<td>15.7</td>
<td>16.6</td>
<td>-Z Panel, Sail Module Mounting Fastener Hole</td>
</tr>
<tr>
<td>3</td>
<td>7.4</td>
<td>36.1</td>
<td>-X Panel, +Z Tray Launch Rail Fastener Hole</td>
</tr>
<tr>
<td>4</td>
<td>8.3</td>
<td>32.3</td>
<td>+X Panel, -Z Tray Launch Rail Fastener Hole</td>
</tr>
<tr>
<td>5</td>
<td>8.7</td>
<td>30.7</td>
<td>+Z Panel, Sail Module Mounting Fastener Hole</td>
</tr>
<tr>
<td>6</td>
<td>6.9</td>
<td>38.7</td>
<td>+Z Panel, S-Band Radio Enclosure Wall Corner Hole</td>
</tr>
<tr>
<td>7</td>
<td>31.4</td>
<td>7.8</td>
<td>-Z Panel, Power Subsystem Mounting Bosses</td>
</tr>
<tr>
<td>8</td>
<td>35.3</td>
<td>6.8</td>
<td>+Z Panel, UHF Radio Enclosure Fastener Hole</td>
</tr>
</tbody>
</table>

Table 2.4: CanX-7 FE Analysis Stress Results, Quasi-Static Load Cases

As a sanity check, a rudimentary analysis using the Euler-Bernoulli bending theory was used to ensure
the FE stress was reasonable. Refer to Figure 2.25. The UHF radio cover has a mass, m, of 51.7 g. The
total inertial load, W, it imparts under 50G is,

\[ W = mg \times 50 \]

\[ = 0.0517 \text{kg} \times 9.81 \text{N/kg} \times 50 \]

\[ = 25.37 \text{N} \]
The four panels screws have a footprint, $L \times w$, of 113.5 mm x 79.5 mm. By idealizing the plate as a simple supported beam with two screws on each of the ends, the moment, $M$, generated at each screws, $S$, can be determined as,

$$M = \frac{WL}{8S}$$

$$= \frac{25.37N \times 0.1135m}{8 \times 2}$$

$$= 0.179N.m$$

Finally, the second moment of area, $I$, for the screw mounting tabs is calculated for its cross section $b \times h$ of 5 mm x 2 mm. With this, the cross sectional stress, $\sigma$, within the mounting tabs is determined at a thickness $y = 1$ mm from the neutral axis.

$$I = \frac{bh^3}{12}$$

$$= \frac{(0.005m)(0.002m)^3}{12}$$

$$= 3.33 \times 10^{-12}m^4$$

$$\sigma = \frac{My}{I}$$

$$= \frac{0.179N.m \times 0.001m}{3.33 \times 10^{-12}m^4}$$

$$= 54.0MPa$$

Note that this was conservative given the actual axis of bending is actually at an angle to the edges of the plate. This gives an effective section width greater than the 5 mm of the corner tab. If the bend line is 45-degrees to the corners, a revised total stress is:

$$I = \frac{(0.005m)(0.002m)^3}{\sin(45 \text{ deg}) \times 12}$$

$$= 4.71 \times 10^{-12}m^4$$

$$\sigma = \frac{0.179N.m \times 0.001m}{4.71 \times 10^{-12}m^4}$$

$$= 38.0MPa$$

The stress value of 38.0 MPa is slightly higher than the 35.5 MPa from the FE result, but is quite comparable given the simplified assumptions of the hand calculations. The high-stress result on the UHF radio cover in analysis case 8 was therefore validated. Similar checks were also performed on other high-stress area as part of the post-processing validation effort.
Modal Analysis and Component Deflections

Results of the satellite modal analysis showed the first body mode at 424 Hz, as shown in Figure 2.26. This is well above the 100 Hz required to meet launch vehicle compatibility. This mode involves the bending of the satellite structure along its longest length and oscillating along the X-axis. Noteworthy is the fact that this mode is much higher than the requirement as the primary CanX-7 structure is extremely stiff by design, including panel stiffeners along the longitudinal length of the thin body panels. Also note that being a modal analysis, the deflection magnitudes generated are immaterial.

![Figure 2.26: CanX-7 First Body Mode](image)

Lastly, the predicted maximum displacements for any of the components in all of the analysis cases are very small. Refer to Table 2.5. This is indicative that no components, in particular PCBs, will be damaged due to excessive deflections causing the components to experience high stresses or come in contact with adjacent hardware.

<table>
<thead>
<tr>
<th>Case</th>
<th>Max Deflection [mm]</th>
<th>Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>0.049</td>
<td>+X Panel, longitudinal edge, towards +X direction</td>
</tr>
<tr>
<td>2</td>
<td>0.038</td>
<td>-X Panel, longitudinal edge, towards X direction</td>
</tr>
<tr>
<td>3</td>
<td>0.026</td>
<td>-X Panel, center of panel, towards +X direction</td>
</tr>
<tr>
<td>4</td>
<td>0.028</td>
<td>-X Panel, center of panel, towards +X direction</td>
</tr>
<tr>
<td>5</td>
<td>0.010</td>
<td>-Y Panel, center of panel, towards +Y direction</td>
</tr>
<tr>
<td>6</td>
<td>0.007</td>
<td>Secondary payload enclosure, lower face, center, -Y direction</td>
</tr>
<tr>
<td>7</td>
<td>0.014</td>
<td>-Z Panel, center of panel, towards +Z direction</td>
</tr>
<tr>
<td>8</td>
<td>0.057</td>
<td>UHF antenna cover, center of panel, towards Z direction</td>
</tr>
</tbody>
</table>

Table 2.5: CanX-7 Maximum Component Deflections
2.3 Assembly, Integration, and Testing

The AIT (Assembly, Integration, and Testing) phase of the CanX-7 development is synonymous with project phase D of a program's life cycle, as described in the NASA system engineering handbook [29]. Phase D focuses primarily on the building and assembly of the engineering models and flight hardware. Part of this process is to identify and wring out potential defects in the system either by design or through inadequate processes and to implement corrective measures. Phase D also entails the establishment of ground support aspects for the mission. Engineering tools including bench top ‘flat-sat’ setups are complimentary to the satellites, and some will continue to be even long after launch as ground-based surrogates in support of orbiting spacecrafts. The end goal of this mission phase is to culminate with an integrated system ready to enter FRR (Flight Readiness Review).

As mechanical lead, the author contributed significantly towards the AIT phase of the CanX-7 mission. During this phase the satellite structure was procured, fit-checked, and assembled. Mechanical aspects of the bus structure were tested to ensure fit, form, and function, including the installation of dummy and flight subsystem components. The structure was also used to support performance testing of a number of subsystems. This included the RF (Radio-Frequency) and EMC (Electro-Magnetic Compatibility) testings in anechoic chambers. Equally important, mechanical GSE (Ground Support Equipment) used in these activities were developed and implemented. Finally, the satellite integration processes were designed, documented, and vetted. This was used in the assembly of the DirtySat, which was the first complete integration of the flight structure and electronics hardware.

2.3.1 Procurement of Structure

Figure 2.27: CanX-7 Flight Structure: –Z Trays, UHF Radio Enclosures, and Deployable Parts

The structure for the CanX-7 bus was made during the summer and fall of 2013. Two sets of hardware were procured, one as an engineering model for integration and system-level testings, and another as the flight structure of the final satellite. The structures were fabricated locally in the Toronto Area after quotes were solicited from a number of prospective vendors. The author examined several criteria during the quoting process that included cost, lead time, and quality (with reference to prior experiences).
Local vendors in the Toronto vicinity were also preferred to reduce shipment cost and time and eliminate cumbersome international custom procedures. Local facilities also allowed for the flexibility of on-site visits and ease of back and forth shipments between the machine shop and facilities used for plating and chemical conversion coating, as necessary. Figure 2.27 shows several of the CanX-7 structural components laid out on the work bench.

Once the structural parts had been received, the author conducted fit checks on all of the machined hardware to ensure fit and finish were within tolerances. Helicoils were installed to facilitate the installation of threaded fasteners, as necessary, to ensure assembly could be done properly. Finally, a complete run-through of the assembly procedure developed by the author was done with the structural components and mock-ups of the subsystem components. Once fit checks were complete, the parts were sent for chemical conversion coating for protection of the bare aluminum structure. Figure 2.28 shows the helicoils installed in the satellite-Z tray and also fit checking of the UHF antenna rotator assembly.

![Figure 2.28: Structural Panel Helicoil Installations (L) and UHF Rotator Assembly Checks (R)](image)

On previous SFL missions, Alodine was the preferred chemical conversion coating for aluminum structures. For CanX-7, the author elected to apply Iridite instead after experiencing difficulties locating local vendors for Alodine application that could deliver consistent batch qualities at reasonable costs. Iridite and Alodine are different product brands both of which meet the MIL-STD-5541F standard for aluminum chemical coating [42]. Unlike Alodine, Iridite may be applied free of chromium, which gives Alodine the characteristic yellow tint, and is also therefore more environmentally friendly. Iridite performs equally well as a treatment for aluminum with adequate surface protection and electrical conductivity characteristics [25]. Iridite application has since been used on a number of other SFL projects including flight structures for the GHGSat-D mission.

### 2.3.2 Ground Support Equipment

The author developed a number of mechanical GSE (Ground Support Equipment) in support of the CanX-7 AIT activities. The GSEs developed by the author accomplished several purposes, including: to assist in hardware and components during assembly and integration; to provide mechanical support and protection during storage, handling, and transportation; and to enable successful testings on satellite and subsystem levels.
Lunch Box

The lunch box for the CanX-7 satellite provides several functions. SFL has traditionally built lunch boxes to offer protection to satellite buses during storage and handling [11, 18]. The lunch boxes are designed to accomplish this by forming an enclosed box around the satellite such that the satellite may be safely handled and placed on surfaces without direct contact with the external environment, including with the handler. Figure 2.29 shows the CanX-2 and GNB satellites in their lunch boxes.

![Figure 2.29: CanX-2 (L) and GNBs (R) in Protective Lunch Boxes](image)

The design of the lunch box is inherently simple and consists of Delrin rails that interface and support the satellite inside the enclosure, and protective Lexan sheets that form the outer panels of the box. The interior volume of the lunch box replicates that of the satellite’s deployer units. This ensures that the satellite will fit with the same clearances as it would on the launch vehicle. The CanX-7 is designed to be supported via integral launch rails on the bus while inside their deployer units during transit and launch. The Delrin rails function in the same way in interfacing and supporting the bus structure by forming direct surface contacts between the Delrin and launch rails of the satellite bus. Figure 2.30 shows the CanX-7 lunch box Delrin rails and Lexan panels prior to integration.

![Figure 2.30: CanX-7 Lunch Box Components Showing Delrin Rails and Lexan Panels](image)
Delrin and Lexan are used for their availability, low cost, and ease of machining. Delrin is a much softer material than aluminum and will not damage the satellite structure in direct contact. The polymer material is also self-lubricating and is ideal to allow smooth insertion and removal of the bus from the enclosure. Clear Lexan is ideal in permitting visual inspection without having to remove the panels and risk exposing the satellite’s exterior faces. Lexan is also resilient to damage and can be handled without concern. The only drawbacks to Delrin and Lexan are their inability to dissipate static charge. Ionizing fans are therefore directed at the enclosure whenever a satellite is present.

For CanX-7, the author also approached the lunch box design as an opportunity for improvement, which are depicted in Figure 2.31 and Figure 2.32. These include:

- Optimizing the hardware configurations associated with attaching the Lexan panels by replacing cumbersome fasteners with captive thumb screws. It was found that the screws on previous SFL lunch boxes were tedious to work with in the clean room environment. Loose screws were not permanently associated with the panels and were an issue if accidentally dropped near the spacecraft or onto the floor. They also introduced potential hazard by requiring tools that were constantly maneuvered around and over delicate satellite hardware. The thumb screws alleviated these concerns and are significantly more efficient to operate.

- Allowing for the local access of the sail module stack to benefit primary payload integration and testings. The Lexan panels on previous SFL lunch boxes were of single-pieces, and access to any part of the satellite entailed exposing the entire face of the spacecraft under the removed panel. The longitudinal body panels on the CanX-7 lunch box is sectioned between the primary payload and the rest of the satellite. This allowed selective access to the payloads and/or the bus structure for servicing and testing.

- Enabling the deployment of the satellite UHF antennas and imager booms for functional testing and other activities while within the protection of its lunch box. The lunch box for CanX-2 was
fully enclosed and to deploy the UHF antenna and booms required liberating the satellite from its protective casing. The GNB lunch boxes do have openings for the UHF antenna, which are fixed and cannot be stowed otherwise. The CanX-7 lunch box was designed to incorporate cutouts in the Lexan cover adjacent to the base of the deployables. This allowed the deployables to be situated in either the extended or stowed positions.

- Tailoring the versatility of the lunch box by constituting its components to support the satellite and sub-assemblies during integration activities. Lunch boxes for the CanX-2 and GNB satellites were used partially as assembly jigs to support the body panels and trays during integration. Lunch box components by design provide a stable support platform and offer stand-off protection for delicately mounted components including solar cell coupons. The CanX-7 lunch box has been designed with the same purposes in mind and is reconfigurable in part and in whole to accommodate this need.

The author completed the lunch box design and manufacturing prior to the integration of the CanX-7 DirtySat (Ref Section 2.3.5). The majority of the lunch box was manufactured at an outside machine shop, while selected components were made on a mill by the author at the SFL facility. Once the lunch box was ready, it was thoroughly tested using the full-scale rapid prototype mock-up to validate that all intended functions are met.

**Assembly Ground Support Equipment**

In addition to the lunch box, assembly GSEs were designed and fabricated by the author to support the various panel sub-assemblies during integrations. The design philosophy for these GSEs are to maximize the use of off-the-shelf hardware and minimize part count, associated machining, and introduction of GSE-specific design alterations to the structural panels. The author was able to achieve these objectives by borrowing GSE design concepts from previous SFL panel GSE designs [13].

The four CanX-7 body panels (+X, −X, +Y, −Y) were designed to be supported with two Delrin rails each, mounted to their longitudinal edges. Hex standoffs were incorporated to secure the panels to the Delrin rails via existing panel mounting holes. These also provide added support to elevate opposite faces of the panels off the assembly benches. Figure 2.33 shows the +/-Y panels mounted to their
support GSEs. The +Z and –Z trays do not have Delrin rails, however. They are supported simply with threaded standoffs mounted to existing fasteners through holes in the launch rails. The standoffs elevate the tray exterior faces off the work bench while components are integrated, after which they will be transferred to the lunch box GSE trays for the integration of the complete satellite assembly; refer to Figure 2.34.

Figure 2.33: +/-Y Panels Mounted To Support GSEs

Figure 2.34: +/-Z Trays Showing Threaded Standoffs

Radio Frequency Pattern Testing Support Equipment

The author was responsible for developing the mechanical GSE in support of the CanX-7 RF (Radio Frequency) testing activities (Ref Section 2.3.4). These GSEs required the following attributes:

- To securely locate the EM at the correct height with respect to the transmitting antenna, located
at opposite ends of the anechoic chamber.

- To allow (re)positioning of the bus in several favorable orientations necessary for data collection.
- May be disassembled and transported between SFL and the test facility.
- Maintain RF-transparency with minimal impact to test results.
- Be cost-conscious and require minimal fabrication effort and time.

The author’s design of the GSE was largely influenced by the required positions and orientations for EM during data collection. The RF testing was done at the University of Toronto St. George campus, Galbraith building, where an anechoic chamber of sufficiently large size was rented. Figure 2.35 shows the CanX-7 satellite Engineering Model (EM) within the chamber during testings. The test setup at the Galbraith facility required the EM and the transmitting antenna to be set up on two pillars on opposite sides of the room. The transmitting antennas, of which there were two types for S-Band and UHF frequencies, were set up on a fixed pillar and directed at the EM. The EM, on the other hand, was mounted on a commercial rotating pillar manufactured for RF testings. The specialized rotator unit permitted a 360-degree yaw along the vertical axis necessary for measurement sweeps.

![Figure 2.35: CanX-7 Engineering Model Inside Galbraith Building Anechoic Chamber](image)

The design of the RF testing GSE was very involved, for several reasons. The correct setup required the EM to be positioned such that the satellite’s antenna bore sight axis be in line with the center of the transmitting antenna. Due to the geometry differences between the S-Band and UHF antennas, GSEs were made to situate the antennas at the correct heights. The heights of the satellite also mattered. For the UHF pattern testings this entailed aligning the center of the four UHF antennas, which in this case is the satellite longitudinal axis coming out of the small -Y panel, along the transmitting antenna line of sight. For the S-Band system, this corresponds to having the satellite longitudinal panels facing the transmitting antenna to align the S-Band patch antenna normal axis down range with the transmitter. The configurations of the GSE needed to ensure that all of these were taken into account.
Finally, a number of different satellite orientations had to be catered to. The CanX-7 RF testing was based on the circular polarization methodology [40, 2]. To obtain the required measurements, the satellite was oriented uniquely for both the UHF and S-Band tests. Total spherical coverage of the satellite was obtained through several 360-degree sweeps conducted at 45-degree increments about the antenna bore sight axis. A total of four orbits were used to achieve the required data points for the complete spacecraft, as shown in Figure 2.36. The resulting requirement therefore called for four distinct attitudes for each of the UHF and S-Band tests.

The author satisfied the position and height requirements with a GSE design that consists of three distinct support blocks. These blocks were rearranged to constitute two different assemblies that allowed for eight unique support orientations. Refer to Figure 2.37 and Figure 2.38. One of the blocks is the support base on which the two others are interchangeably mounted in a vertical orientation. The geometries of all three pieces were carefully designed to allow maximum usability while ensuring angular orientation of the satellite EM and position of the antenna bore sight axis. In addition, the setups were tailored for the Galbraith anechoic facility with considerations for the position and heights of the rotating pillars.
The GSE mounting blocks were fabricated by the author at the SFL facility. Urethane foam was used for the GSE material for its low cost, general availability, and good RF transparency. Foam is also low in mass and could be easily transported and mounted atop the anechoic chamber support pillars without concern. Two-inch thick urethane foam sheets were procured by the author and manually cut using a commercially available hot wire cutter. Due to the thickness of the foam material and the flexibility of the wire, jigs were devised to ensure proper alignment and angular precisions of cuts. Once the blocks were fabricated the complete setup was tested with the full-size CanX-7 mock-up, shown in Figure 2.39, prior to being used by the EM. This GSE mounting setup was successfully used in the RF testing and demonstrated the ability to be rapidly reconfigured by repositioning the satellite and vertical blocks without the need to remove the base. This negated the need for time-consuming re-calibration in between tests, which contributed significantly to both test efficiency and measurement confidence.
2.3.3 Integration Procedures

The development of the CanX-7 assembly procedure focused on efficient and concise assembly flow, optimal layout of the wiring harness and components, minimized risks to the spacecraft, and implications for testing, handling, and storage of the bus. This section presents the results of the assembly and integration development efforts accomplished by the author.

Full-Scale Mock-Up

The CanX-7 team elected to develop the integration procedures through the use of a mechanical mock-up, which allowed for the evaluation of the mechanical and procedural design on a full-scale bus structure. The former was achieved through the use of spare components such as blank, non-flight electronic circuit boards. These ensure the high fidelity of the mock-up, which was instrumental in the development of the wiring harness assembly (Section 2.1.3 herein). The ergonomics of the design was validated by a walk-through of the complete assembly steps using representative hardware and tools to ensure sufficient access and minimized risks to component damage due to human interactions.

The mechanical mock-up, shown in Figure 2.40, was made in 1:1 scale using fused deposition modeling (FDM), a subset of 3D printing technology. This was a result of early evaluations done by the CanX-7 team [35], which found several advantages with the approach, including:

- The capacity to produce very high-quality mock-ups, which greatly increases the confidence in the fidelity of the exercise.
- Significantly lower cost and turn-around time against other manufacturing methods such as machining for comparable products.
- An ability to quickly export machine-readable formats from 3D solid models, circumventing the time and effort required to generate production prints and engineering drawing.

![Figure 2.40: Full-Scale Rapid Prototype Mock-Ups (L), Showing High Fidelity of FDM 3D Printing (R)](image)

The recent proliferation of commercially available 3D printing services have contributed significantly to the above factors. As such, rapid prototyping is now a preferred way to evaluate design concepts and prototype across many industries. The use of 3D printing as a tool for mechanical design by the
CanX-7 team is a first in SFL with demonstrated benefits. The same technology is also being used to fabricate flight structure of the CanX-7 sail module housings and enclosure, which would be impossible to fabricate with traditional machining due to its complex geometries.

An earlier copy of the CanX-7 mock-up was fabricated during the PDR phase of the CanX-7 development, and an updated copy was commissioned by the author post-CDR as the design had matured sufficiently to render the first set outdated and no longer representative. The new mock-up was produced in M30i plastic, which is essentially a branded ABS material. It was printed by Cimetrix Solutions in Oshawa, Canada, using a new line of Fortus printers, which has an advertised resolution of down to 0.005” (0.127 mm). The new mock-up was used to great benefit in both the assembly and integration design, and developing the final wiring harness configurations. As an addendum, it was also used to assist in other program-level development work, including validating the lunch box and the RF testing support stand (Ref Section 2.3.2).

Assembly Concept

![Satellite Sub-Assembly Breakdown](image)

It was decided early on that the CanX-7 +Z and –Z trays should be the first to be integrated. These panels were designed as the primary mechanical members of the bus and form the basis onto which all other components are assembled. As such the majority of the subsystem hardware were located directly to these two trays to allow for straightforward load paths between components and the primary structure. From an integration standpoint this drastically simplified the assembly process to a decision of when and in what order major sub-assemblies are married, as the components and wiring harnesses are already installed to the integrated +Z & –Z tray assemblies. It was found that regardless of the
assembly flow thereafter, the bus +X panel, which is the long body panel with the +X magnetorquer, must be integrated next. This is to protect the delicate backplane of POSIF board of the PDU, and also to access the magnetorquer connector, which could not be done on any other sequence. From there, the various assembly concepts diverged.

Initially the bus and payloads were regarded as distinct, isolated entities. The desire was to modularize the two to allow flexibilities in integration and testing. Ideally this benefits efficiency by allowing parallel development with reduced interdependency. Four ‘body panel priority’ and ‘payload priority’ approaches were proposed as presented in Figure 2.42. Sequences (1) and (4) were disposed of as the secondary payload must be installed before the primary payload due to mechanical interferences. Sequence (3) was also deemed undesirable as the –X panel must be installed after the secondary payload to gain access to connectors during the latter’s installation. Procedure (2) was preferred as it did not have such limitations, and the body panels were also integrated last. It was also ideal as closing the –X and –Y panels last permit inspection and access to critical IFN connectors throughout integration.

Figure 2.42: Integration Concept - Payload & Body Panel Priority Approaches
Other concepts were explored including hybrid approaches where the payloads and panels are installed in lockstep during integration, as shown in Figure 2.43. This does not permit as much independence between the payloads and the bus, however the desire to maintain access under the −X and −Y panels were greater. Concept (5) was ruled out for the same reason that the secondary payload must be integrated before the primary payload stack. Due to delays in the development of the sail modules, concept (6) was eventually selected. This allowed the rest of the satellite to be integrated to support the DirtySat and RF testing activities without the need for primary payload integration. Concept (6a) was initially chosen at the end of the assembly procedure development phase, however concept (6b) was ultimately decided upon.
Integration Overview

As alluded to earlier, the CanX-7 assembly procedure developed by the author follows a hierarchical framework where the satellite components and sub-assemblies are built first, which are then integrated to form a complete satellite bus. The design of the sub-assemblies was intentionally modularized to allows final integration and close-out at the satellite-level. This section provides an overview of integration on both the bus and sub-assembly levels.

The assembly procedure breaks the satellite down into the following sub-assemblies:

- 4x UHF and 1x Image Boom Deployables
- +X and –X Body Panel Sub-Assemblies
- +Y and –Y Body Panel Sub-Assemblies
- +Z and –Z Tray Sub-Assemblies
- Secondary Payload Enclosure Sub-Assembly
- Upper and Lower Primary Payload (Sail Stack) Assemblies

Deployables

The deployables are unique in that they generally remained permanently integrated once assembled. The CanX-7 has five deployables: four UHF monopole antennas and one imager boom unit. Assembly of the UHF deployables, shown in Figure 2.44, consists of integrating the rotator arms to the mounting bases along with the respective mandrels and actuating springs. The antennas are pre-fabricated by the author from 0.041” diameter music wire crimped into D-sub connectors. These are installed into the free end of the rotator and the corresponding coax cable connection is inserted into the opposite end. When fully integrated, the UHF antennas and their corresponding coax cables are permanently adhered to the rotator with 3M 2216 B/A epoxy adhesive to prevent from detaching.

Figure 2.44: UHF Antenna Assemblies with Rotator Unit Detailed Exploded View
The imager boom integration is much more involved, due to the presence of the magnetometer and the mVIC payload inside the enclosure at the tip of the boom. Refer to Figure 2.45. There are three smaller sub-assemblies that must be integrated for the boom. First, the rotator is assembled. This involves installing the rotator into the base, together with its spring, mandrel, and a single #0-80 socket head screw. Second, the mVIC payload is installed into the imager case. Care is taken to ensure the minuscule imager payload is not damaged. Finally the imager case cover, which resembles more of the case than the cover, is integrated with the magnetometer and the deployment runners.

Once the rotator unit, the imager case, and the imager case covers are assembled, they are integrated together first by closing the imager case and imager case cover together, following which the imager case cover and the rotator unit are attached to the imager boom. The imager boom is secured in place with #2-56 set screws designed to ensure both proper alignment and shear locking capacity. See Figure 2.46. The integration of the imager boom deployable is now complete.

Figure 2.45: Imager Boom Rotator, Imager Case, and Imager Case Cover Assemblies

Figure 2.46: Image Boom Integration
+X and –X Panel Assemblies

Figure 2.47: +X Panel Sub-Assembly Shown Installed with Delrin Rails Support GSEs

The +X and –X panels line two of the longitudinal sides of the bus. However, they are relatively un-populated by components, with only the +X printed magnetorquer mounted to the interior of the +X panel. However, they do support 2x (+X panel) and 1x (–X panel) solar cell coupons on their exteriors, and each of the panels also has one S-Band patch antenna bonded to the outside. Refer to Figure 2.47.

+Y and –Y Panel Assemblies

Figure 2.48: –Y Panel Sub-Assembly Shown Installed with Delrin Rails Support GSEs

The +Y and –Y panels are similar to the +X and –X panels in terms of sub-assembly preparations. Figure 2.48 shows the –Y panel integrated and installed in its support GSE rails. The +Y panel is identical except it does not have a printed magnetorquer on its interior. Details of the assembly GSEs for the +Y and –Y panel, as well as those for the +X and –X panels, are found in Section 2.3.2 herein. These rails make use of the existing structural mounting locations to hold the parts, and secure the parts with hex standoffs that double as legs to elevate the assembly above the bench surfaces.

+Z Panel Assembly

The +Z tray is one of the two main structural trays on the CanX-7, and holds the battery unit including the BCDR (Battery Charge-Discharge Regulator), as well as both UHF and S-Band radios. The integration of the +Z tray is therefore more involved than the X and Y-panels. See Figure 2.49
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Figure 2.49: +Z Tray Sub-Assembly Shown with Aluminum Standoff Support GSEs

The installation of the radio units are identical to many of the SFL satellites, which use versions of the same S-Band and UHF radios in the same radio compartment layout and internal geometry. This is a delicate and time-consuming operation involving RF shielding measures, not shown in Figure 2.50 but can be referenced in Section 2.3.5 herein. The radio units are installed first followed by the separation switches, which are permanently bonded down with Scotch Weld 2216 RTV. The battery and BCDR unit is then integrated to complete the +Z tray sub-assembly.

Figure 2.50: Integration of S-Band and UHF Radios Into Their Respective Enclosures

–Z Panel Assembly

The –Z tray is by far the most involved of the sub-assemblies. It supports a significant number of hardware components including the Z-magnetorquer and the test port connector. It is also integrated with large unprotected electronics of the PDU (Power Distribution Unit) and the OBC (On-Board Computer), which must be handled delicately. See Figure 2.51. The wiring harness is also part of the –Z tray assembly, as it is permanently installed to the PDU hardware. As such, the –Z tray assembly procedure includes several harness tie-downs to pre-secure the wiring assembly both to alleviate downstream workload and to aggressively pursue assembly tidiness.
Primary and Secondary Payloads

The secondary payload enclosure consists of a machined enclosure to accommodate the ADS-B Beaglebone electronics unit; a connector panel on which the payload Micro-D connector, the pre-amp, and the RS-485 are mounted; and an outer panel that supports the ADS-B L-Band patch antenna. These are assembled to form an enclosed unit that is then integrated to the satellite bus. Figure 2.52 is an exploded view of the assembly.

The upper and lower sail stacks are identical and consist of a machined mounting plate and two triangular sail modules. See Figure 2.53. Each sail module is secured to the under side of its mounting plate with three socket head screws through the top of the plate. The modular design of the sail module stack ensures that the mechanical mounting interfaces between the plate and the sail modules, and between the satellite bus and the plate, are straightforward.
Bus Integration

The majority of the design for the bus integration procedure was accomplished with the full-scale rapid prototype mock-up. In accordance with the integration concept, the bus assembly starts with the +Z and –Z tray sub-assemblies, which are placed adjacent to each other. This allows the radio harness attached to the primary bus harness of the –Z tray to be connected to the radios on the +Z tray. The UHF and S-Band radio coax cables are also installed at this point to ensure optimal access of the SMA connectors. See Figure 2.54. The +Z tray is then inverted over the –Z tray with Delrin support rods firmly secured to both trays to provide support, alignment, and proper spacing. Several component and wiring connections are made at this point, after which the majority of the remaining wire supports and tie downs are installed along the –X side of the satellite bus. Once the +Z and –Z trays are integrated, the +X panel is installed and its relevant connections are mated, as shown in Figure 2.55.
The deployables are attached at this point, however, their presence was deemed unnecessary during the full-scale mock-up phase. For details see the DirtySat integration as described in Section 2.3.5 herein.

The secondary payload installation is next, which is accomplished with care to prevent damages to the delicate wiring harnesses. The secondary payload enclosure is situated adjacent to the OBC and PDU units of the satellite bus and there exists minimal clearances between the enclosure sidewalls and the wiring harness along the computer and power boards. The payload enclosure is also in close proximity to the magnetorquer on the +X panel and the wiring harness routing toward the primary payloads. Figure 2.56 is an indication of the installation process.

The last of the panels to be installed are the –X panel and then the –Y panel, as shown in Figure 2.57. The installation of these panels is straightforward, with the exception of the concentration of wiring and coax cables on the –Y end of the bus. Wiring and harness tie downs are checked at this point to ensure proper strain relief and clearances with respect to the –Y panel magnetorquer when closed. Noteworthy is that the initial assembly procedure used for the mock-up was to install the –Y panel first and then the –X, consistent with procedure (6a) in Figure 2.43. It was through the combined experience gained on both the mock-up and the DirtySat that procedure (6b) was deemed a better approach. This change largely affects the S-Band antenna coax cable connection to the –X panel through the –Y side of the bus. To do so with the original assembly procedure would require the –Y panel to be removed and reinstalled again. The process (6b) negates this need.
The above steps encompass the results of the integration process as designed with the full-scale mock-up. Notice in Figure 2.58 that installation of the primary payload stacks was simulated only superficially with mock-ups of the mounting plates due to lack of available sail mock-ups. As well, the sail stack wiring harness was also being finalized during this time.

2.3.4 Radio Frequency Pattern Testing

SFL satellites go through RF (Radio Frequency) pattern testing as part of the overall development process. Pattern testing provides empirical test data of the satellite RF pattern for validation against predicted HFSS (High Frequency Structural Simulator) performances \[2\]. For CanX-7 this included testings of both the satellite’s UHF and S-Band systems with the satellite bus assembled in a representative state. In addition to developing the mechanical GSE (Section 2.3.2), the author prepared the EM (Engineering Model) of the satellite bus for use in this effort. See Figure 2.59. Specifically, the EM was assembled using the spare structure, which included the primary spacecraft panels and trays and the secondary payload enclosure to complete the spacecraft exterior shell. Additional aluminum plates were machined by the author to close off the open cavities of the missing sail module stack, to simulate the electromagnetic properties of the module exterior with thermal tapes applied. To complete the EM in a flight-representative configuration, the imager boom was installed and secured in the deployed position during testings to capture the effects of the metallic boom on the overall pattern signature.
During RF testing the satellite EM was not fitted with an operational S-Band transmitter or UHF receiver. In lieu is a splitter for each of the UHF and S-Band frequencies that takes an incoming signal from an external source through a connected coax cable and feeds the signal directly to the antennas. The author was able to accommodate the splitter units into the EM in a manner to allow the use of spare antenna coax cables without modifications. This was important as signal attenuations through these coax cables are particularly sensitive and are captured as part of the test results. The external coax cable was accommodated through an existing feed-through in the secondary payload enclosure.

The EM was also prepared with spare antenna hardware on its exterior. The spare set of the CanX-7 S-Band patch antenna was taped to their positions on the EM in preparation for the RF testing. Care was taken to ensure proper electrical isolations between the patch antenna and the metallic structure underneath. The UHF antennas were also attached and trimmed down to length. RF isolation tests were then conducted on the lab bench, shown in Figure 2.60, to ensure that both the UHF and S-Band systems perform as expected prior to packaging and transfer to the test anechoic facility.
2.3.5 DirtySat

![Figure 2.61: CanX-7 DirtySat Undergoing Testing During Integration](image)

The CanX-7 DirtySat is the first flight integration of the satellite. It was built using the spare structure and a combination of flight and spare electronics, the latter including the mVIC camera EM where the flight copy was not yet available. Figure 2.61 shows the integrated DirtySat on the test bench. The integration of the DirtySat was the first time that flight electronics, such as shown in Figure 2.62, were married with flight-representative structure, and was a significant milestone in the development phase of CanX-7. The goals for the DirtySat included the following:

- To ensure the fit and form of the mechanical aspects of satellite flight hardware and spare structure through fit checks of all sub-assemblies.
- Validate the complete CanX-7 integration procedures and ensure the assembly steps may be accomplished as intended. This includes ensuring the clarity and completeness of the assembly procedure such that it may be performed by individuals unfamiliar with developing the integration process (i.e., not the author).
- Ensure EMC (Electromagnetic Compatibility) of the satellite that all systems and parts are properly grounded and there is no adverse affect between subsystem hardware in close proximity.
- Perform LFFT (Long Form Functional Tests) to ensure the health of the satellite. This includes the general functionality of all the subsystems to validate that they power on, exhibiting nominal power consumption and data acquisition, and operate as expected.

Preparation for the DirtySat integration was completed in spring 2014. The spare structure and assembly GSE including lunch box were cleaned and fit checked. The only exceptions to this were a number of assembly GSEs that were already fitted to the flight panels that were situated in the clean room environment. Specifically these were the X and Y-panel GSE rails that were needed to elevate the flight panels to prevent contacts of their solar cells and the bench top. As such their omission during the DirtySat integration was not a detriment to the fidelity of the exercise. Lastly, the wiring harness assembly was manufactured and installed to the flight electronics and safe-to-mate activities were performed.
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Figure 2.62: Flight PDU with Primary Bus Harness (L) and Magnetorquer with Wiring(R)

The CanX-7 DirtySat integration was completed by the author during May 2014. No major modifications were made aside from a number of minor design changes, such as adjustments to wiring harness tie-downs due to the use of lacing chords in lieu of zip ties on the wiring mock-up and the bulk of the ubiquitous kapton labels on the flight harness. No significant issues arose with the assembly procedure also, a tribute to the successful design and preparation efforts leading up to the exercise.

Figure 2.63: Initial Boom Harness, Attached To Bus (L) and Modified As Stand-Alone Unit (R)

The only mechanical change to the wiring harness is an additional in-line connector to segregate the imager boom wiring from the primary bus harness. During CanX-7 harness development the author designed the component and wiring interfaces to aggressively minimize in-line connectors. The intent was to reduce the associated quantity of parts and also potential failure points with each added connection. With this in mind the imager boom wiring was initially made a permanent part of the primary bus harness. Integrating these into the imager boom entailed awkwardly threading the harness through the base of the boom, the rod, and connecting it to the disassembled and exposed imager and magnetometer PCBs, both of which are delicate. This was found to be more cumbersome than envisioned. The imager boom harness was subsequently made a unique unit that could be pre-integrated with the boom and the PCBs, greatly simplifying the entire process. Refer to Figure 2.63 for the pre and post-modification boom configurations.
Figure 2.64: CanX-7 Dirty Sat Sub-Assembly Integrations

Figure 2.65: Dirty Sat Bus Integration

Figure 2.66: Completed DirtySat Inside Lunch Box Protective Enclosure
2.4 Future Work

As of writing, the author has made strides towards preparation of the flight hardware. All flight structures have been fit checked, cleaned, and mechanical GSEs for the panels are made and fitted to the individual panel assemblies. Thermal tapes have been applied to the exterior of the body panels, and surface-mounted components are bonded. Lay down of the surface-mounted components were done in accordance with existing SFL procedures [12] for preparations, adhesive application and bonding, final curing, and cleanups.

Nevertheless, several significant milestones remain. The flight assembly of the satellite has yet to take place, after which, the satellite is to undergo key system-level testings in the thermal-vacuum chamber and with the vibration test facility at SFL. These and the remainder of the AIT activities should occur throughout the rest of 2014, with the aim to have the satellite enter Flight Readiness Review (FRR) by the end of the calendar year.
Chapter 3

Nanosatellite for Earth Monitoring and Observation - High Definition

The Nanosatellite for Earth Monitoring and Observation - High Definition (NEMO-HD) satellite is the largest SFL satellite currently under development. NEMO-HD is a microsatellite with approximate external dimensions of 600 mm x 600 mm x 300 mm and a projected mass of 65 kg. The satellite is designed to obtain high-definition imagery and video capture of at best 2.8 m Ground Sampling Distance (GSD) from an orbit altitude of 600 km [37], including high-definition 1080p video streams. As such, the satellite will deliver unprecedented performance, flexibility, and future capabilities in a compact form factor.

Mechanically the NEMO-HD bus is very ambitious, which is a monocoque structure design using aluminum skin, aluminum core honeycomb panels. This is the first time that a SFL-developed bus employs a significant amount of honeycomb panels as the primary structural element, whereas previous buses were of machined alloy structure or had limited use of composite panels only as solar cell panel backplanes. The large amount of structural honeycomb panels enables the bus to achieve significant mass savings while retaining strength and stiffness. The only other exception being the Microvariability and
Oscillation of STars (MOST) microsatellite, which was launched in 2003. Its structural design was based largely on existing buses by Surrey Satellite Technology, Ltd. (SSTL) in the United Kingdom, and those of Radio Amateur Satellite Corporation (AMSAT).

The size of the NEMO-HD bus was required to accommodate the primary payload, which consists of a single machined barrel with a maximum diameter of 188 mm and a length of 445 mm, with an additional electronics enclosure measuring 183 mm x 200 mm x 205 mm. The large volume of the bus also allows the various subsystem components to be distributed and housed in machined enclosures throughout the interior of the satellite bus. Externally the bus solar cells are bonded to panels that are stood off of the primary structural panels, in order to achieve the required thermal decoupling from the bus. External mounted components also comprise various types of antennas.

Figure 3.2: NEMO-HD ‘Butterfly View’ Showing Distributed Modularized Interior Compartments

The author’s contributions towards NEMO-HD consisted of two facets, which were in the mechanical analysis of the NEMO-HD structure and also the design and fabrication of ground support equipment. The analysis done for NEMO-HD included those on the component and subassembly levels. Of those, the alignment assessment between the primary telescope and the satellite start tracker assembly due to thermal-elastic effects is presented herein, the scope of which is sufficiently representative of all others given its complexity. Next, the full satellite structural analysis is presented which includes the author’s approach in evaluating the honeycomb sandwich structure and mesh optimizations for large satellite assemblies. Finally, a brief discussion is given regarding the details of the various mechanical ground support equipment.
3.1 Assembly-Level Analysis

The author has made significant contributions in the component and sub-assembly level analysis for the NEMO-HD satellite. One of these is the alignment analysis between the primary telescope and the two star tracker units under on-orbit thermal elastic effects. This analysis is presented as it encompasses aspects of the many different types of analysis done. It is also one of the assessments done involving both the satellite’s structure and another subsystem’s components that are critical in determining the satellite’s operational performances.

3.1.1 Overview

The NEMO-HD spacecraft has two star trackers for precise attitude determinations. The star trackers protrude from the –Z panel through an opening in the panel structure. As designed, they are orientated at 45° from the satellite’s –Z axis and at ±30° from the satellite’s X-Z plane to achieve a 60° separation between the bore sights of the two units. Mechanically the star trackers are mounted on an adapter bracket, which is attached directly to the telescope cylinder. This minimizes mechanical separations between the telescope and the star trackers, which reduces alignment uncertainties that contribute towards the satellite’s attitude control errors. Finally, the telescope assembly including the optical bench on the end of the telescope cylinder is mounted by a forward and an aft mounting bracket to the spacecraft –Z panel. Refer to Figure 3.3.

To ascertain the thermal elastic alignment implications of the star trackers and the telescope, it is reasonable to reduce the problem down to the determination of their bore sight axis at conditions of interest. The bore sight of the primary telescope can be defined in two ways, which are the normals to the front face of the telescope barrel and also that of the optical bench interface. These represent the locations where the light enters into the telescope barrel, and where it exits into the imager sensors. They are different, however, due to the deformation of the barrel itself under thermal loads. The bore sights of the star trackers are defined by their fastener mounting positions on the star tracker bracket. Each star tracker is mounted via three M3 screws, which can be used as three points that define a plane, the normal of which represents the line of sight of the star tracker unit.
3.1.2 Finite Element Analysis

The thermal-elastic analysis is done using FE (finite element) analysis. An assembly FE model was constructed to include detailed meshes of the telescope cylinder, the forward and aft telescope brackets, and the star tracker bracket assembly. The telescope cylinder is a large structure with significant thermal mass and geometry. Therefore, it is important to capture the variations in thermal gradient and the mechanical displacement inherent of the barrel under the thermal inputs. Similarly, the two telescope mounting brackets are also considered for their complex geometries and size. The –Z structural panel was not considered as its sandwich structure is relatively stable both thermally and mechanically.

The FE analysis was accomplished in a two-stage process. First, a thermal analysis was conducted to solve for temperature gradients across the entire assembly. Temperature loads generated by the SFL thermal engineer for various on-orbit scenarios were provided to the author who then used them as inputs to the system. Second, a structural FE analysis was done to solve for mechanical deformations using the gradient map generated from the thermal FE analysis. The final result was used to extract bore sight axis information, which was used in the alignment calculations.

All machined components are meshed with properties of aluminum 7075-T6 material using auto-meshed quadratic 10-node tetrahedron (tet-10) elements. Auto meshing of the elements was done to expedite the modeling process and to allow complicated geometries and details to be accurately captured that would otherwise not be possible with manual meshing techniques. This is critical as thermal elastic modeling requires very high accuracy in part details. Element quality was manually enforced by discretizing the parts into optimal sections for meshing based on geometry and feature sizing. The element distribution within these sections was manually enforced to constrain node distributions along edges and surfaces. Finally, the resulting meshes were checked for quality and refined as necessary.

A number of RBE2 and RBE3 rigid elements were used throughout the thermal-elastic FE assembly. These include manually defined spider elements to simulate fastener connections between component meshes. RBE2s were also used to simulate the stiffness of the two star trackers and the optical bench by constraining their mounting fastener locations. This effectively models the star trackers and the optical bench as perfectly rigid parts, which are sufficient for the intent of the analysis.

Surface contact constraints are enforced between corresponding parts to ensure mechanical loads are accurately distributed across mated surfaces. Lastly, the forward and aft telescope brackets are secured to simulate bolted connections to the –Z panel.
### 3.1.3 Alignment Determinations

As previously alluded to, the alignment analysis was conducted by calculating the relative states of the telescope and star tracker bore sight axes. The states of four axes were considered, two on the telescope barrel and one for each of the star trackers.

Any arbitrary Cartesian local frame in space, $F_{arb}$, with mutually orthogonal unit axes $\hat{a}_1, \hat{a}_2, \hat{a}_3$, such that,

$$F_{arb} = [\hat{a}_1 \hat{a}_2 \hat{a}_3]^T$$

(3.1)

can be determined through a set of three non-coincident points $P_1, P_2, P_3$ in the global inertial frame $F_1$. These points define two local vectors $\xi_1, \xi_2$ where

$$\xi_1 = P_2 - P_1$$

(3.2)

$$\xi_2 = P_3 - P_1$$

(3.3)

with which the components $\hat{a}_1, \hat{a}_2, \hat{a}_3$ of $F_{arb}$ can be deduced through vector analysis. The author used this approach to define local reference frames for both the telescope and star tracker units from their respective reference points.

The defining points used in the analysis are shown in Figure 3.6, and are as follows. The points on the front end of the telescope (Tf) are $P_{1Tf}, P_{2Tf}, P_{3Tf}$, and are three nodes on the forward rim of the barrel. The three points on the aft end of the telescope (Ta) are $P_{1Ta}, P_{2Ta}, P_{3Ta}$, which are defined by the center of the optical bench interface and two of the peripheral fasteners. These were chosen corresponding to where the light enters the telescope and where the light exits the barrel and enters the optical bench. The points for the star trackers are the three mounting fastener locations on the star tracker bracket, which are $P_{S1Tf}, P_{S2Tf}, P_{S3Tf}$ for star tracker 1 and $P_{S1Ta}, P_{S2Ta}, P_{S3Ta}$ for star tracker 2. Note that star trackers 1 and 2 correspond to the $-Y$ and $+Y$ star side trackers with respect to their positions on the spacecraft. The star tracker points were chosen to align the local frames of the star trackers to the local coordinates of the actual units.
Given the geometry of the points, the components of the forward and aft telescope frames, \( F_{T_f}, F_{T_a} \), were calculated as,

\[
\begin{align*}
    a_1 &= \frac{r_2 \times r_1}{\| r_2 \times r_1 \|} \\
    a_2 &= \frac{a_3 \times a_1}{\| a_3 \times a_1 \|} \\
    a_3 &= \frac{-(r_2 + r_1)}{\| r_2 + r_1 \|}
\end{align*}
\]  

and the components for the two star tracker frames, \( F_{S1}, F_{S2} \), were defined as

\[
\begin{align*}
    a_1 &= \frac{a_2 \times a_3}{\| a_2 \times a_3 \|} \\
    a_2 &= \frac{r_1}{\| r_1 \|} \\
    a_3 &= \frac{r_1 \times r_2}{\| r_1 \times r_2 \|}
\end{align*}
\]

For a comparative assessment, local frames for the initial (undeformed) and final (deformed) states were defined. As an aside, the local frames will not become non-orthogonal, due to the employment of fixed rigid elements to ensure stiffness between the nodes.

The alignments between the local frames were defined in terms of rotations, \( \delta \varphi \), such that

\[
\delta \varphi = [\vartheta_1 \ \vartheta_2 \ \vartheta_3]^T
\]
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The transformations between any two arbitrary reference frames $F_1, F_2$ can be defined by a rotation matrix, $M_{21}$, such that,

$$F_2 = M_{21} \cdot F_1$$  \hspace{1cm} (3.11)

The requirements for this exercise were to determine the alignment between the telescope and star trackers 1 (S1) and 2 (S2), as well as the alignment between the two star tracker units. Five corresponding rotation matrices are therefore needed, which are $M_{S1Tf}, M_{S2Tf}, M_{S1Ta}, M_{S2Ta}$, and $M_{S2S1}$. Again, an initial and a final set of matrices were determined.

Finally, to determine the misalignment between the initial and final states, transformation matrices, $R$, were calculated given the rotation matrices, such that,

$$M_{\text{final}} = R \cdot M_{\text{ini}}$$  \hspace{1cm} (3.12)

To extract the angular values of the misalignment, the 3-2-1 Euler rotation matrix relationship was invoked, where,

$$R = R_{\theta_3}R_{\theta_2}R_{\theta_1} = \begin{bmatrix} c_2c_3 & c_1s_3 + s_1s_2c_3 & s_1s_3 - c_1s_2c_3 \\ -c_2s_3 & c_1c_3 - s_1s_2s_3 & s_1c_3 + c_1s_2s_3 \\ s_2 & -s_1c_2 & c_1c_2 \end{bmatrix}$$  \hspace{1cm} (3.13)

Where $s_i = \sin(\theta_i)$, $c_i = \cos(\theta_i)$. The Euler angles for the deformation rotation vector are then extracted, as follows,

$$\theta_1 = -\text{atan2} \left( \frac{R_{3,2}}{R_{3,3}} \right)$$  \hspace{1cm} (3.14)

$$\theta_2 = \text{asin} \left( R_{3,1} \right)$$  \hspace{1cm} (3.15)

$$\theta_3 = -\text{atan2} \left( \frac{R_{2,1}}{R_{1,1}} \right)$$  \hspace{1cm} (3.16)

Note that for 3-2-1 Euler transformation, singularity is experienced only when $\theta_2 = \frac{\pi}{2}$. As the expected angles are merely on the order of arc seconds, this is not an issue.

A MATLAB script was written by the author to compute the above analysis. The program takes in the coordinates of the defining points for both initial and final states, and generates the resulting deformation angles. The results from the analysis are favorable with the highest star tracker to telescope misalignment of $-11.4$ arc seconds and the highest star tracker to star tracker misalignment of $-8.1$ arc seconds. These are well within the requirements of $\pm 36$ arc seconds stipulated by the NEMO-HD attitude determination and control subsystem.
3.2 Full Satellite Structural Analysis

The full satellite structural analysis of NEMO-HD was a very challenging endeavor, in a couple of ways. The satellite is a very large assembly with a significant number of subsystem components. The meshing of these components were laborious and it was necessary to discern what was critical and must be included, what could be simplified and represented differently, and what could be omitted outright. As well, the NEMO-HD bus is only the second satellite for SFL that uses honeycomb sandwich panel as its primary structural material. The previous instance was the MOST microsatellite whose structural design had origins at the Surrey Satellite Technology Ltd. (SSTL) and with the Amateur Radio Satellite (AMSAT) group, as previously alluded to. The analysis of honeycomb sandwich structure is much more involved than with traditional isotropic material used for SFL bus structures. Therefore, it was an area that warranted significant scrutiny both in modeling and in post-processing.

3.2.1 Modeling

Honeycomb Panels

Figure 3.7: Honeycomb Panel Modeling (L-R): Accurate Model, 2D Laminate, and Equivalent Panel

Several methods for modeling honeycomb panels using the finite element methods, including specific techniques within the NX8 software, have been carefully investigated by Diaconu [14]. Three approaches were discussed in particular, as shown in Figure 3.7, including the accurate model where core and face sheets geometries are generated using 2D elements; the 2D mesh composite laminate approach; and the equivalent panel method where 3D elements are used to approximate actual panel geometries. A comparative analysis showed that the results of the three methods generally agree closely with each other. However, a 2D composite shell was chosen as the preferred method due to significantly less modeling and computational efforts. For NEMO-HD, the author has elected to model the honeycomb panels with 2D composite laminate elements for the same reasons.

It is noted that Diaconu baselined the accurate model for comparison against the other approaches. The author took this further by validating the NX8 composite modeling capability against empirical test data. Honeycomb panels are very commonly used in the aircraft industry for a wide variety of applications. As part of their certification process, aircraft structural panels are required to conform to published standards. One of these is the long beam flexure test specified in MIL-STD-401B: Sandwich Construction and Core Materials; General Test Methods (since superseded by SAE AMS-STD-401). Manufacturers of aircraft grade honeycomb panels typically make available panel specifications and mechanical properties including long beam flexural test results. By generating 2D meshes of several aircraft grade sandwich panels and reproducing the test setup in FE, the author was able to make good correlations between the numerical and published panel data.
The test that was replicated was the standard long beam flexural test, which involves coupons measuring 3” x 22”. In a four-point test, the coupon is supported 1” inboard of each end and the loads are applied 10” apart centered about the panel. Figure 3.8 shows setups of the standard three and four-point test setup. Two sets of data are collected as part of the test:

- A face sheet tensile failure test that indicates the ultimate load at failure and the corresponding face sheet stress
- A displacement test that gives the maximum panel deflections under a 100 lbf load as a measure of panel stiffness

The author chose to validate the 2D meshes against data for aluminum face sheet and aluminum core sandwich panels. Aluminum is an isotropic material with well-established attributes and is a part of the NX8 material library and therefore a good candidate material. Core material cards were custom generated based on published information from Hexcel [20], a prime manufacturer of aerospace-grade honeycomb core material. The panels that were modeled included Gillfab 4030 (0.500” thick) by M.C.Gill [24] and two different panels from the AA207 family manufactured by Teklam [39]. An example of the FE stress plot is shown in Figure 3.9. Results of the FE coupon tests is given in Table 3.1.

As can be seen the FE results consistently over estimated both the panel stress and deflection magnitudes. The author conjectures that the thickness of the adhesives in real life was responsible for the discrepancy. This is because the bonding medium adds a finite amount of thickness to the cross section of the facing material, increasing overall stiffness and distributing face sheet stresses. As face sheet materials for these panels are on the order of 0.5 mm, the added adhesive thickness is not insignificant. This is also why the differences were more pronounced for thinner honeycomb panels, as mathematically the stress is affected by the cube of face sheet thickness and only linearly with panel thickness.
Table 3.1: Long Beam Flexure Results, FE vs. Published Data

Nevertheless, the variations between the published data and the FE coupon tests were small with an average of around 6%. The overestimation also entailed a more conservative structural analysis, which is favorable. A more rigorous exercise to reconcile the FE results and published data was not pursued for the interest of time and diminished returns. The composite panel modeling method using 2D shell elements in NX8 was therefore deemed sufficient for use.

The NEMO-HD spacecraft uses three dissimilar builds of honeycomb panels. Two of the panel types are used for the structural panels and are identical except in thickness. The thinner panel with a thickness of 6.35 mm is used throughout the entire structure, with the exception of the –Z (main) panel, which is made from a 30.0 mm” thick sandwich. Both of the structural panels have 0.6 mm thick aluminum 7075-T6 face sheets and an aluminum 5056 foil honeycomb core with a 1/8” cell size and a core weight of 6.1 pcf (pound per cubic foot). The third type of panel is used exclusively for the solar cell coupons. It employs 0.644 mm thick carbon fiber laminate facings made from ultra-nigh modulus pitch carbon laminate, and the same core as the structural panels. Aside from its own weight and those of the bonded solar cells, these panels carry no significant structural loads.

Figure 3.10: NEMO-HD Body Panel Mesh (L) and Panel Sub-Assembly Showing Dissimilar Meshes (R)

The honeycomb panels were meshed using second order quadratic elements to ensure accuracy of mesh. Panel regions were manually discretized to reflect the footprints of panel-mounted components to facilitate contacts between part meshes. Elements around panel mounted inserts were also meshed with aluminum to simulate the metallic insert material. Material and laminate properties for each of the honeycomb panel were manually specified, including carbon laminate ply orientations. Figure 3.10 shows the satellite-level assembly with body panels only, as well as the +X+Y panel assembly with dissimilar panel mesh types.
Components

Figure 3.11: NEMO-HD Telescope: High-Fidelity Mesh (L) v.s. Simplified Coarse Mesh (R)

Component meshes used for the NEMO-HD full-satellite analysis were different than those used for the detailed parts assessment. The latter is intended for detailed understanding of part behavior and focused optimizations, and require higher fidelity models with realistic features and finer mesh discretization. On the other hand, full satellite analysis are used to gain an understanding of the global interactions useful for initial concept design and more generalized assessments. As such they are tailored with coarser meshes and representative but simplified attributes, enabling a more efficient assembly FE modeling process and to ensure a reasonable computational effort. Figure 3.11 shows the difference between the high-fidelity mesh of the NEMO-HD telescope barrel as used for the thermal-elastic analysis, versus the significantly simplified version in the full satellite structural FE assembly.

Figure 3.12: NEMO-AM Structure with Mass Blocks (L) and CanX-2 Mass Dummy (R)

The majority of the subsystem enclosures were modeled as representative mass blocks. Mechanical testing of actual structural test articles often involve mass simulators in lieu of non-primary structure hardware and enclosures. Figure 3.12 shows the mass blocks used in the NEMO-AM spacecraft for its vibration testing, and also the CanX-2 whole-satellite mass dummy used for XPOD validation. The XPOD mass dummy in Section 4.1.3 is another example of a mass simulator albeit on a larger scale.

The author took a similar approach when modeling the full satellite assembly FE for NEMO-HD, where mass blocks were used in lieu of point masses and rigid elements. This allowed for better definitions
of proper footprint, inertia, and surface contact conditions where the resulting stiffness contributions to the body panels are significant. This involved drastically simplifying the components with most of the complicated features removed to save meshing and computational effort. Critical geometries such as general dimensions, wall thicknesses, and mounting features were retained by discretion to allow for reasonable approximations of critical modal and stress data. The overall assembly mass and mass center were enforced by carefully adjusting internal distributions of material. An example of this is the modeling of the payload electronics compartment, shown in Figure 3.13. Figure 3.14 shows the distribution of the representative mass blocks situated in the finalized NEMO-HD FE assembly model.

![Figure 3.13: Payload Electronics Enclosure: Solid Assembly, Simplified Mass Model, Final FE Mesh](image1)

![Figure 3.14: Completed NEMO-HD Structural FE Assembly with Various Mass Dummies](image2)

### 3.2.2 Boundary Conditions

The boundary conditions in the NEMO-HD analysis is drastically less complex than those for the CanX-7 analysis in Section 2.2.2, for two reasons. First, the NEMO-HD does not use an encapsulating deployer system. Its launch vehicle interface is therefore not a set of simply supported surface contacts that are case-dependent and applied in combination to achieve realistic constraints. Rather, the NEMO-HD utilizes an adapter ring that is rigidly mounted to the launch vehicle and mechanically affixes to the satellite via a number of explosive bolts. These bolts are permanent and structural until severed on command to release the satellite from the launch vehicle at the end of the launch phase. In finite
element modeling, these bolts were adequately represented as user-defined fixed constraints at the bolt interfaces on the satellite. Surface contacts between the adapter ring and the satellite are neglected as the quantity of bolts (19 total) and the small pitch in between are sufficient to constrain out-of-plane motions of the satellite mounting plate due to the adapter ring flange. Surface contact constraints are present internally within the NEMO-HD satellite FE assembly, however, which were manually defined between components with direct surface contacts. This was done to ensure that proper stiffness and load distribution characteristics from part-surface interactions were adequately captured. Figure 3.15 shows the NEMO-HD satellite fitted with the Soyuz/Fregat launch adaptor ring. Although specifics between it and other launch vehicle adaptors may differ, their interfaces at the NEMO-HD spacecraft’s mounting plate are identical.

Figure 3.15: NEMO-HD Satellite With Launch Vehicle Adapter Ring

The NEMO-HD FE model was analyzed to eight different load cases. This was to encapsulate all possible loading scenarios and to ensure that mounting of the satellite to the launch vehicle is orientation agnostic. As of writing the author was given the Soyuz/Fregat combination as the prospective launch vehicle. The maximum inertial load applied in the analysis is 43.5G, which is the five-sigma value (99.9997%) of the 8.7G Root Mean Square (RMS), as stipulated in the NEMO-HD qualification vibration test plan [27]. This load was alternately applied to both the positive and negative directions of all three primary axes to arrive at six different analysis cases. The same load is also applied in two additional cases where components of 25.11 G are simultaneously applied in the +X/+Y/+Z and –X/–Y/–Z directions. This is to capture the representative scenarios where the resultant is not aligned with any of the satellite’s body axes. Table 3.2 outlines the eight different load cases used in the NEMO-HD quasi-static analysis.
Table 3.2: NEMO-HD Quasi-Static Load Cases

3.2.3 Post Processing

The results from the FE analysis were looked at in four separate facets, including structural honeycomb panel failures, inserts failures, machined component stresses, and satellite modal characteristics. These four components encompassed the stress and natural frequency aspects of the post processing.

Structural Honeycomb Panels

Unlike isotropic materials, there are several different failure modes applicable to the generic sandwich panel that must be considered. The author conducted an investigation of these failure modes for applicability to NEMO-HD and evaluated the performances of the satellite’s structural sandwich panels where necessary. There are seven unique types of sandwich failure, which include,

- Facing failure - tensile or compressive failure of the face sheets
- Transverse shear - core material shear failure
- Core crush/indentation - failure of the core due to localized pressure or excessive panel deflections
- Buckling - panel buckling due to in-plane compressive loads
- Shear crimpling - when buckling results in additional localized core shear failure
- Face wrinkling - local face sheet buckling outwards due to delamination or inwards due to core compression failure (indentation)
- Face dimpling - multiple inter-cell buckling due to excessively large core material cell sizes

The specifics of these failure modes are covered in depth in existing literature on honeycomb technology, such as the one by Bitzer [5], which the author consulted with extensively. In general, the first four failure modes are grouped as global failures while the remainder are local modes.

The local modes are not a concern for the NEMO-HD satellite. Face wrinkling and dimpling, as well as global modes of buckling and shear crimpling, occur with excessive in-plane compressions. For face wrinkling, the critical stress $\sigma_{wcr}$ is expressed as

$$\sigma_{wcr} = 0.82E_f \left[ \frac{E_c t_f}{E_f t_c} \right]^{1/2}$$  \hspace{1cm}(3.17)$$

where $E_f$ and $E_c$ are the face sheet and core compressive modulus, respectively, and $t_f$ and $t_c$ are the face sheet and core thicknesses, respectively. Using this equation the critical wrinkling stress for the 6.35 mm thick and 30.0 mm thick structural honeycomb panels were evaluated to be 3.36 GPa and 1.42 GPa, respectively. Additionally for face dimpling, the critical stress $\sigma_{Dcr}$ is expressed as

$$\sigma_{Dcr} = \frac{2E_f}{\lambda_f} \left[ \frac{t_f}{S_c} \right]^2$$  \hspace{1cm}(3.18)$$

where $\lambda_f$ is a stiffness term with a value of 0.89 for aluminum and $S_c$ is the core cell size. This allowed the panel critical dimpling stress to be determined at 5.70 GPa for both the thick and thin structural panels as they share the same core material. Both the face wrinkling and dimpling allowables are much higher than can be expected for the satellite under all conceivable loading scenarios. Note that the maximum in-plane face sheet tensile stress numbers shown in Table 3.3 below are absolute stresses, for conservatism. The compressive stress experienced by the face sheets will also be at or below the maximum of 106.0 MPa.

Shear crimpling is a down stream failure mode after panel buckling occurs, the mechanical expression for which is the same as for local dimpling in equation 3.18. These failure modes were also considered not feasible.

Sandwich panels undergo core compression failures at high compressive loads, after which the core settles into a ‘crush’ mode where a lower but relatively constant crush strength prevails until the core is wholly compressed. The failure load is therefore the core stabilized compressive strength, which is 4.69 MPa for the structural panels on NEMO-HD. The most massive component within the satellite is the primary payload by far, at 24.6 kg. It is supported by the front and aft telescope brackets which have a combined footprint of 0.028 m$^2$. With a 43.5 G inertial load, the pressure it exerts on the panel is 376 KPa, well below the stabilized core compressive strength limit. All other panel-mounted components exert significantly lower loads due to their larger footprint-to-mass ratio that evenly and adequately distribute loads across the contacting surfaces.
The core shear strength, $\tau_c$, of the panels is expressed as

$$\tau_c = \frac{V}{hb} \quad (3.19)$$

where $V$ is the shear load, $h$ is the height between the centerline of the face sheets otherwise calculated as the sum of the face sheet and core thickness $h = t_c + t_f$, and $b$ is the width of the shear face. Using the core shear strength of 2.34 MPa in the weaker ‘w’-ribbon direction, which is much weaker than the actual transverse shear strength of a honeycomb core material, the failure shear load per-centimeter width for the body panels was determined. This turned out to be 134.9 N for the 6.35 mm thick panel and 675 N for the 30.0 mm panel. To place this into perspective, these loads would cause the inserts to tear out in tension much earlier than the same load would cause core shear in compression, again due to the large perimeter and footprints of the panel-mounted components.

Lastly, the facing failure as a result of face sheet material tensile yielding is determined by extracting the face sheet stresses from the finite element model. Table 3.3 shows the highest panel face sheet stresses for all eight analysis cases, along with the margins of safety calculated against the yield strength of 434 MPa for aluminum 7075-T6, and the locations of the high stresses. As evident, overall face sheet stresses are acceptable with the highest being 106.0 MPa. This is associated with one of the inserts on the –Z panel for the LV adaptor plate, as shown in Figure 3.17. The same insert also supports a corner bracket that mounts the +X–Y side body panel, compounding the loads through the area. Even so, the margin of safety is a healthy 3.1 with no concerns for face sheet failure. The margin of safety requirements for NEMO-HD is greater than zero (positive) [26].

<table>
<thead>
<tr>
<th>Case</th>
<th>Stress [MPa]</th>
<th>M.S.</th>
<th>Location</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>106.0</td>
<td>3.1</td>
<td>Corner Fastener, –Z panel, with –Y panel &amp; +X–Y panel</td>
</tr>
<tr>
<td>2</td>
<td>105.9</td>
<td>3.1</td>
<td>Corner Fastener, –Z panel, with –Y panel &amp; +X–Y panel</td>
</tr>
<tr>
<td>3</td>
<td>112.5</td>
<td>2.9</td>
<td>Center Fastener, –Z panel with LV Adaptor Plate</td>
</tr>
<tr>
<td>4</td>
<td>113.3</td>
<td>2.8</td>
<td>Center Fastener, –Z panel with LV Adaptor Plate</td>
</tr>
<tr>
<td>5</td>
<td>64.6</td>
<td>5.7</td>
<td>Center Fastener, –Z panel with –X–Y Panel</td>
</tr>
<tr>
<td>6</td>
<td>64.6</td>
<td>5.7</td>
<td>Center Fastener, –Z panel with –X–Y Panel</td>
</tr>
<tr>
<td>7</td>
<td>61.2</td>
<td>6.1</td>
<td>Corner Fastener, –Z panel, with –Y panel &amp; +X–Y panel</td>
</tr>
<tr>
<td>8</td>
<td>61.1</td>
<td>6.1</td>
<td>Corner Fastener, –Z panel, with –Y panel &amp; +X–Y panel</td>
</tr>
</tbody>
</table>

Table 3.3: Structural Panel Face Sheet In-Plane Tensile Stresses
Inserts are the de facto method for attaching threaded fasteners to honeycomb sandwich panels and are a critical aspect that had to be considered. Significant work has been done in the aerospace industry to characterize the performances of inserts. Nevertheless, it has been shown that their exact behavior, like any composite assemblies, is difficult to capture due to its interactions with the sandwich panel whose particulars must also be accounted for. Factors that come into play include insert material, length, diameter, potting adhesive properties and radius, sandwich panel geometries, as well as the mechanical attributes of both its face sheet and core material. Moreover, the performances of insert-honeycomb combinations are often sensitive to factors in the manufacturing processes including potting approach and curing methods. These are reasons for OEMs (Original Equipment Manufacturers) such as Boeing, Airbus, and Bombardier, and also panel manufacturers such as Teklam to pursue individual and often proprietary standards in insert data.

The best method to ascertain the mechanical properties of the insert-honeycomb combination is through batch testing of production-grade coupons. As of writing, the NEMO-HD team is not privvy to this information. Without the benefit of empirical data, the author elected to characterize insert performances by using analytical methods given in the Insert Design Handbook of the European Space Agency [15], in conjunction with material information from the Hexcel honeycomb attributes and properties [20] and the
MMPDS materials database of the U.S. Department of Transportation/Federal Aviation Administration [43]. It has been shown in these literature that the analytical methods presented below are generally more conservative when compared with those from pull test results. Although this could have led to an over design it was used as the baseline in the absence of better data.

Refer to Figure 3.19, the critical insert tensile load, $P_c$, is

$$P_c = 2\pi b_p d \tau_c$$

(3.20)

where $b_p$ is the insert potting radius, $d$ is the distance between the center of the two face sheets, and $\tau_c$ is the critical out-of-plane core shear strength defined as $1.36 \times \tau_w$, which is the core shear strength in the weaker 'w'-ribbon direction. The second diagram in Figure 3.19 illustrates the differences between $b_i$ and $b_R$, which are the insert and “real” potting radius, respectively, the latter of which is irregular. A semi-empirical relationship is therefore used to approximate $b_p$ for a given $b_i$:

$$b_p = b_i + 0.5S_c$$

(3.21)

where $S_c$ is the cell size of the honeycomb core material.

Similarly, a relationship exists to determine the critical shear load, $Q_c$, which is

$$Q_c = 8b_p^2 \tau_w + 2fb_p \sigma_{fy}$$

(3.22)

where $f$ is the face sheet thickness and $\sigma_{fy}$ is the yield strength of the face sheet material. Refer to Figure 3.20. Notice the enclosure which stiffens the honeycomb panel and prevents rotation of the insert, and also translating moment loads into out of plane tensile and compressive loads through coupling of fastener on opposite edges. This is critical as inserts have much lower load performances resisting moment loads.

An edge bearing scenario occurs when insert is close to the edge of a panel, typically at edges distances of less than 2x insert diameter. In this case the panel tear-out load, $Q_b$, is

$$Q_b = 4\sigma_{bf}e$$

(3.23)
where $\sigma_f$ is the bearing allowable of the face sheet material, and $e$ is the edge distance. This is applicable for many of the NEMO-HD panel inserts mounted around the outer edges of the structural panels.

Figure 3.20: Potted Insert (Thru) Shear Load Diagram (L) and Edge Tear-Out Scenario (R)

<table>
<thead>
<tr>
<th>Panel</th>
<th>Insert</th>
<th>$P_c$ [N]</th>
<th>$Q_c$ [N]</th>
<th>$Q_h$ [N]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Structural Panel 30.0 mm</td>
<td>M3</td>
<td>3585</td>
<td>2998</td>
<td>2457</td>
</tr>
<tr>
<td>Structural Panel 30.0 mm</td>
<td>M4</td>
<td>4468</td>
<td>4318</td>
<td>3276</td>
</tr>
<tr>
<td>Structural Panel 30.0 mm</td>
<td>M5</td>
<td>5352</td>
<td>5866</td>
<td>4095</td>
</tr>
<tr>
<td>Structural Panel 6.35 mm</td>
<td>M3</td>
<td>701</td>
<td>2998</td>
<td>2457</td>
</tr>
<tr>
<td>Structural Panel 6.35 mm</td>
<td>M4</td>
<td>874</td>
<td>4318</td>
<td>3276</td>
</tr>
<tr>
<td>Structural Panel 6.35 mm</td>
<td>M5</td>
<td>1047</td>
<td>5866</td>
<td>4095</td>
</tr>
</tbody>
</table>

Table 3.4: NEMO-HD Structural Panel Insert Allowables (Analytical)

Using the above formulas, the author was able to determine the analytical performances of the insert-sandwich panel combination specific to the NEMO-HD structural panels. Refer to Table 3.4. Notice that the tensile loading varies significantly between the 6.35 mm panel and the 30.0 mm panel across all three insert types, due to the significant impact of core height. Conversely, the shear and tear-out allowable are the same between the 30.0 mm and 6.35 mm thick panels as the core height does not factor in at all.

Figure 3.21: FE Plot of +X+Y Panel and Highlighted Insert Nodes for Loads Extraction
From the FE results, tensile and shear loads were extracted for all of the individual inserts on all nine primary structural honeycomb body panels for all eight of the analysis load cases. This was laboriously done by individually selecting the nodes from each insert location and extracting the sum of their nodal grid point forces, which was then used to calculate the in-plane and out-of-plane loads experienced by the insert. Refer to Figure 3.21. The loads were used to determine the margin of safety for the inserts against the calculated insert capacities. Results of this analysis are tabulated in Table 3.5, where the insert with the lowest margin is identified by load cases. Margins for the solar cell panel inserts were not calculated as they carry virtually no structural loads.

Note that the lowest insert tensile and shear safety margins are experienced in cases 3 and 4, which occur when the satellite undergoes 43.5 G lateral accelerations along the Y-axis. The fasteners in question are on two of the satellite’s diagonal bottom edges joining the +X–Y panel and the –Z panel. Refer to Figure 3.22. These two fasteners also share an internal load-bearing bracket tying the panels together. During lateral loadings, the satellite sees global deformations that result in significant loads being transferred via these brackets to the launch vehicle adaptor plate resting on the exterior of the –Z panel. This fastener and its corresponding opposite on the +X+Y panel, experience significant loads to resist the body moment generated by the lateral inertial load.

<table>
<thead>
<tr>
<th>Case</th>
<th>Tensile Margin</th>
<th>Size</th>
<th>Location</th>
<th>Shear Margin</th>
<th>Size</th>
<th>Location</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>2.94</td>
<td>M5</td>
<td>–Z Panel</td>
<td>1.63</td>
<td>M4</td>
<td>+X+Y Panel</td>
</tr>
<tr>
<td>2</td>
<td>2.95</td>
<td>M5</td>
<td>–Z Panel</td>
<td>1.63</td>
<td>M4</td>
<td>+X+Y Panel</td>
</tr>
<tr>
<td>3</td>
<td>1.74</td>
<td>M4</td>
<td>–Z Panel</td>
<td>0.53</td>
<td>M4</td>
<td>+X+Y Panel</td>
</tr>
<tr>
<td>4</td>
<td>1.74</td>
<td>M4</td>
<td>–Z Panel</td>
<td>0.54</td>
<td>M4</td>
<td>+X+Y Panel</td>
</tr>
<tr>
<td>5</td>
<td>2.30</td>
<td>M5</td>
<td>–Z Panel</td>
<td>0.87</td>
<td>M4</td>
<td>+X–Y Panel</td>
</tr>
<tr>
<td>6</td>
<td>2.30</td>
<td>M5</td>
<td>–Z Panel</td>
<td>0.87</td>
<td>M4</td>
<td>+X–Y Panel</td>
</tr>
<tr>
<td>7</td>
<td>5.83</td>
<td>M5</td>
<td>–Z Panel</td>
<td>3.56</td>
<td>M4</td>
<td>+X+Y Panel</td>
</tr>
<tr>
<td>8</td>
<td>1.42</td>
<td>M5</td>
<td>–Z Panel</td>
<td>3.56</td>
<td>M4</td>
<td>+X+Y Panel</td>
</tr>
</tbody>
</table>

Table 3.5: NEMO-HD Structural Panel Insert Minimum Margins

Figure 3.22: NEMO-HD FE Plot (L) and Solid Model (R) of Worse-Case Insert Location
Machined Components

All of the alloy parts on NEMO-HD are machined from either magnesium AZ31B-H24 or aluminum 7075-T6. The majority of the components are fabricated from the lower density magnesium to achieve significant mass savings. The only locations where aluminum is used are areas where higher stresses are expected, including +X Panel, forward and aft telescope brackets, star tracker brackets, satellite honeycomb body panel corner joints, and the satellite’s launch vehicle adaptor plate. The highest stresses and associated safety margins for machined components are tabulated in Table 3.6. Note that the material yield strengths for aluminum and magnesium are 434 MPa and 220 MPa, respectively. As evident, the values all meet the positive margin requirement with the lowest M.S. value attributed to the forward telescope bracket during cases 3 and 4, which are the 43.5 G cases along the +/-Y axis.

<table>
<thead>
<tr>
<th>Case</th>
<th>Aluminum 7075-T6</th>
<th>Magnesium AZ31B-H24</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Stress [MPa]</td>
<td>M.S.</td>
</tr>
<tr>
<td>1</td>
<td>214.1</td>
<td>1.0</td>
</tr>
<tr>
<td>2</td>
<td>213.1</td>
<td>1.0</td>
</tr>
<tr>
<td>3</td>
<td>270.0</td>
<td>0.6</td>
</tr>
<tr>
<td>4</td>
<td>257.6</td>
<td>0.7</td>
</tr>
<tr>
<td>5</td>
<td>118.4</td>
<td>2.7</td>
</tr>
<tr>
<td>6</td>
<td>118.5</td>
<td>2.7</td>
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<tr>
<td>7</td>
<td>123.6</td>
<td>2.5</td>
</tr>
<tr>
<td>8</td>
<td>123.0</td>
<td>2.5</td>
</tr>
</tbody>
</table>

Table 3.6: Machined Component Stresses, Margins, and Locations

Modal Behavior

Results of the NEMO-HD modal analysis showed the first, second, and third modes at 133 Hz, 139 Hz, and 140 Hz, respectively. The first and third are body modes showing deflections of the exterior honeycomb panels. This is due to the inertia from the large number of panel-mounted subsystem enclosures, which have significant mass contributions to the panel assemblies. The second mode shape is a localized deflection of the primary payload back end enclosure. This particular model shape corresponded well with expectations given the cantilevered nature of the enclosure off the rear of the telescope barrel.

Figure 3.23: NEMO-HD First Three Mode Shapes (L-R): 133 Hz, 139 Hz, 140 Hz

All of the first three mode values are much greater than the 35 Hz FNF requirement for Soyuz payloads.
and also the 100 Hz qualification sinusoidal vibration loads of the launch vehicle [4]. It must also be stressed that the FE modal analysis was done without the benefits of surface contact for computational reasons, which would have stiffened the assembly and increase the body modes above the result values.

3.3 Assembly, Integration, and Testing Contributions

The author’s contributions in the AIT phase of NEMO-HD consisted primarily of the development of GSE (ground support equipment). This included GSEs for component testing, handling and integration, and for storage and protection. An example of each is given in this section.

3.3.1 Component Testing GSEs

The author contributed in finalizing the design and performing the in-house machining of the rate sensor calibration jig. Rate sensor calibrations are done with the sensors mounted on a turn table machine. Its primary goal is to determine the measurement error inherent of the sensors themselves and also imperfections in their mechanical interfaces to the rate sensor enclosure. To determine the system biases, rate sensor data are measured over a range of orientations and post-processed. A simple calibration jig was developed consisting of an L-bracket mounted to the turn table and a plate that is mounted to the L-bracket flange. The plate supports the rate sensor enclosure and is allowed to pivot between 0° and 90° in 15° graduations.

The author was tasked with the design of mounting GSEs for the NEMO-HO PAN (Panchromatic) and MS (Multi-Spectral) camera sensors. A need was identified to allow mounting of the camera PCB stacks in the most accessible manner during bench testing with minimal damage concerns to the fragile data ribbon cables. The GSEs were to be used for thermal vacuum testing of hardware and must provide realistic heat sink characteristics. Further, they also needed to support the sensors atop a commercial tripod adjacent to a light sphere for optical testings and calibrations. The author designed the GSEs out of machined aluminum sheets and extrusions for simplicity and to reduce material and machining costs. The GSE for the MS sensors is a simple plate to allow the two sensor stacks to be situated with the data ribbon left unstressed in between. The setup for the PAN camera was more involved with L-brackets to provide proper heat sinks for the electronics and to orient the sensor perpendicular to the base plate.
Commercial hardware and standoffs were used on both GSEs for wire harness tie-downs, and 1/4”-28 threaded holes were provided to interface with standard camera tripods.

3.3.2 Handling and Integration GSEs

During the design phase of the NEMO-HD, it was stipulated that a method was needed to handle the primary telescope as an integrated assembly. It was decided that a set of forward and aft handles would allow the safe handling and positioning of the telescope unit. However, the design of the aft handle was difficult given a lack of positive mechanical interfaces to secure the handles to the telescope assembly. This was complicated by a design freeze on the telescope barrel, which disallowed additional fastener lo-
cations to accommodate the GSEs. The author was able to design a modular handle to clasp around the telescope assembly and mount to existing fastener locations between the barrel and the optical bench. This design alleviated additional rework to the barrel design while still affording secured handling of the telescope assembly.

In addition, the author also assisted in the fabrication of a number of assembly GSEs, including the support leg assemblies for the \(-Z\) panel. These consist of one aluminum cross bar that interfaces with the satellite’s \(-Z\) panel and two vertical Delrin legs that attach to the outside of the cross bar. Four sets of these legs were made, which are used to support and elevate the NEMO-HD satellite above the work bench to clear the star tracker assembly protruding from the bottom.

### 3.3.3 Storage and Protection GSEs

![Figure 3.27: LRI Lens Dirty Cover](image)

One of the storage and protection GSEs designed by the author were the clean and dirty lens covers for the NEMO-HD LRI (Low Resolution Imager). These were made to protect the lens during integration and transit, and should be transparent to allow verification of the imager with the cover on. The dirty lens, used outside of the clean room environment, was designed as a simple lens cap that was friction fitted with a machined internal mohr taper sized for the imager barrel. The body of the cap was designed using Delrin as a soft and self-lubricating material, to prevent abrading the imager. Clear Lexan was used for the transparent material and was adhered to the Delrin with 3M 2216A/B epoxy adhesive.

The clean cover for use in the clean room environment was more involved in that it must be installed on the outside of the assembled NEMO-HD bus. The LRI camera protrudes out of the satellite on one of the side panels with the camera bore sight at 45 degrees to the panel face. The clean cover must interface with the satellite body panel and the lens surround bracket, while providing an unobstructed view for the imager. The final design of the clean cover fit over the imager port and secured into the camera surround bracket with two captive thumb screws. The latter were to allow handling without the use of loose fasteners and hand tools that are undesirable during LV-level integrations. The clean cover design was also of Delrin and clear Lexan, and was fabricated by the author internally at SFL.
3.4 Afterthought

This section on NEMO-HD is focused very much on the analysis of the structure with a token presentation on the ground aspects of the satellite. The intent is to stress upon an understanding of the differences for a large satellite system such as the NEMO-HD from those of a smaller bus like the CanX-7. The thermal-elastic analysis was a subsystem-driven effort unique to a large platform with very stringent subsystem requirements. Likewise, the approach towards the full-satellite FE analysis differed substantially due to the complexities involved, the boundary conditions experienced, and the unique primary structural buildup in both the configurations and materials used. Finally, the implementations of ground support equipment were tailored for very specific subsystem development processes, and in ways that diverged from those applicable for prism-shaped nanosatellites. The work done on NEMO-HD was therefore an example of how its mechanical developments evolved to successfully support and cater to the satellite system as a whole.
Chapter 4

eXoadaptable PyrOless Deployer

Nanosatellites typically interface with and are released from launch vehicles with purpose-built deployer units. Examples of this include the P-POD (Poly-Picosatellite Orbital Deployer) and the T-POD (Tokyo Picosatellite Orbital Deployer) developed by the California Polytechnic University and Tokyo University, respectively [34]. Analogous systems include NASA’s universal launch adaptor system for nanosatellites for use on launch vehicles’ upper stages, and the JASSOD (Japanese Small Satellite Orbital Deployer) aboard the International Space Station. The XPOD (eXoadaptable PyrOless Deployer) system is a unique family of nanosatellite deployers developed by the SFL. The various XPOD designs cater to a range of satellite bus form factors, including CubeSats (1U, 2U, 3U), GNB (200 mm-cube), twin-GNB (200 mm x 200 mm x 400 mm), and a larger 270 mm-cube unit. With the exception of CanX-1, which was launched in a P-POD, all SFL developed missions have been launched using the XPOD system.

The author’s contributions to the XPOD system is in the development of ground support equipment for use during launch preparation and testing. Specifically, a mechanical simulator for the GNB version of the XPOD was designed, built, and used during multiple launch campaigns. The resulting programmatic benefits were significant risk and cost reductions associated with launch-vehicle-level testings by using the mass simulator in lieu of actual engineering models of the XPOD-GNB combination.
4.1 XPOD Mass Dummy

The XMD (XPOD Mass Dummy) was developed under an initiative for a mechanical simulator of the XPOD-GNB combination in support of the Nanosatellite Launch Service (NLS) program at SFL. This was to support launch campaign activities that encompass mechanical validations as part of the LV (Launch Vehicle) stack. These tests include fit and handling around LV structure and adjacent spacecrafts, spacecraft-LV umbilical connection, and complete LV-satellite stack loads testing, amongst others. Prior to the XMD, all of the SFL LV integration activities involved engineering models of the actual satellites and deployers. Figure 4.2 includes images from the NLS-12 (Nanosatellite Launch Services) launch preparations where the AISSAT-2 spacecraft EM was used for mechanical checks at the launch facility in Russia. Typically, these EMs are high-fidelity and are assembled from expensive and hardware that carry certain inherent risks. The use of EMs also require on-site presence of SFL personnel, which do incur significant program costs.

Figure 4.2: AISSAT II During LV Mechanical Fit Checks (L) and Mounted For Centrifuge Tests (R)

Mechanical simulators in contrast are drastically simplified representations that are inexpensive to fabricate, carry minimal risk implications from potential damages, and do not require on-site supervision. Their only requirements are to possess certain mechanical attributes of the units they represent. In the case of integration, the mechanical simulators may be crude bodies that are representative only in mass, center of mass, specific geometries, and mounting interfaces for the handling and integration tests. Mechanical simulators may also be tuned to specific modal characteristics for integrated dynamic tests.

The XMD was commissioned as an in-situ replacement for the XPOD-GNB combination, with the following criteria:

- The mounting interface and hardware shall be identical to the XPOD-GNB.
- The major external dimensions shall approximate the XPOD-GNB as closely as possible to aid in handling tests.
- Protrusions on the GNB, i.e., the magnetometer and UHF antenna, shall be replicated.
- The connector and locations for the XPOD-LV umbilicals shall be replicated.
- The mass and center of mass of the XPOD-GNB should be matched as closely as possible.
- The first natural frequency of the XPOD-GNB should be matched as closely as possible.
Fabrication should make the maximum use of stock material and fasteners while minimizing machining and part count.

The first four criteria relate to the geometries of the XPOD-GNB, which are critical to ensure that the XMD interfaces with the LV and the surrounding payloads. The next two are critical to ensure the mechanical behavior of the unit and are also representative in relation to the satellite-deployer combination it represents. Finally, the last point is a program desire to minimize development cost under a compressed schedule. Together these requirements stipulate a mechanical simulator that performs a dual purpose by acting as both a geometry and mass/dynamic surrogate, an ambitious goal typically done with multiple dissimilar dummies as the two can be difficult to reconcile.

4.1.1 Design and Analysis

The dominant factors that influenced the early designs of the XMD were the geometry requirements. Figure 4.3 is an overview of the XPOD-GNB and its distinctive features. The main body consists of two panels mounted to a base plate and capped by the door on top. The GNB sits within the middle cavity with the UHF antenna and magnetorquer boom protruding out the open front and rear faces. The sides of the XPOD are flanked by beams with flanged ends and braced upper edges, to provide stiffness to the main body panels. An E-Box (Electronics Box) is mounted to one of the body panels and incorporates connectors that interface with the LV umbilical cables. The entire XPOD-GNB assembly is attached to the launch vehicle via both the side beams and the base plate. Given these, the approach was taken to simplify the XMD into three sections. A center portion would represent the many body panels together with the GNB satellite. Then, two side beams would be added to form the XPOD side beams. This general configuration would be adjusted to fit the LV interface of the XPOD-GNB.

Several design iterations were made during the conceptual phase of the development. The flow of the evolution is presented in Figure 4.4. The earliest design was a large aluminum box comprised of
machined body panels to form the central body and extrusions for the outboard legs. Due to the selective availability of extrusion sizes, the legs were recessed into the body panels. This significantly drove up machining effort, in addition to the lightening holes already on the panels. Variations on this design were evaluated to investigate implications to complexity, part counts, and cost, before it was abandoned for fabrication reasons, following which a minimalist approach was considered to aggressively reduce custom features and to maximize the use of locally available stock extrusions and plate materials. This concept incorporated three identical I-beams sandwiching two spacer plates, which were the primary structural members that secured the assembly to the base. The spacer plates also served a dual purpose to widen the assembly to the correct overall width of the XPOD. This second concept, however, was also rejected due to a lack of physical likeness of the center section, which was too flat such that only a superficially resemblance to the XPOD-GNB could be achieved.

The third concept was developed that leveraged on the minimalist design by adopting the two outboard I-beams and the widened vertical spacer plates. The center I-beam was replaced with a central block made sufficiently deep to correspond to a GNB satellite. In this configuration the magnetometer dummy and UHF antennas could also be mounted in appropriate locations. This design retained the simplicity of the previous concept while still making maximum use of stock materials. The central block also incurred minimal machining while achieving very good approximations of the XPOD-GNB shape and form. Once this third concept was solidified, it was adjusted to refine the assembly’s modal characteristics by adjusting part geometries to vary assembly stiffness. The total mass and mass center were also varied by altering the depth and wall thickness of the middle cavity. Finally, a number of last-minute adjustments were made that were unanticipated. A clerical error at the machining facility resulted in a mismatch between the specified and procured stock extrusions. This significantly affected the mass and the stiffness of the assembly, which forced several redesigns including the introduction of lightening holes and angled upper edge on the I-beams. The lower portions of the vertical spacers were also narrowed to achieve the proper body modes. The author successfully addressed these changes within a very short time frame to ensure program schedules were not adversely affected.
The author used finite element modeling to predict the modal characteristics of the XMD in between design iterations. This was accomplished by importing the solid models from Solid Edge into Siemens NX8 software meshing and analysis. Assembly finite element models were constructed for each variation, which were then analyzed. Varying FE meshes were used to leverage advantages with both manual and automated meshing techniques. This allows an optimal balance of mesh effort and quality while meeting the narrow development window.

![Figure 4.5: XMD Solid Model, FE Assembly, and Modal Analysis](image)

The geometry of the definitive XPOD mass dummy is shown in Figure 4.6, juxtaposed against the XPOD-GNB. As evident, the major geometries are consistent between the two, including body measurements, mounting base plate sizing, umbilical connector placement, and the magnetometer boom and UHF antenna details. The final mass difference between the two was within 100 g and the center of gravity variations was within 10 mm.

![Figure 4.6: XPOD-GNB vs. XPOD Mass Dummy](image)
4.1.2 Manufacturing

Figure 4.7: XMD Components (L) and Assembly in SFL Clean Room (R)

The XMD was machined at a local vendor to allow for rapid turn-around of manufacturing and delivery. All of the XMD components were fabricated from aluminum 6061-T6, which is a commonly used aerospace-grade metal with good machinability, availability, and low cost. The base plate and vertical spacers were made from plate stock. The side beams and angled brackets were made using the same I-beam extrusion. The central block, magnetometer boom dummy, and the UHF antenna mounting bracket were the only parts that required machining from larger block material. The XMD was designed to be assembled with standard metric size fasteners made with 18-8 stainless steel. Threaded holes were tapped and helicoils installed to preserve the longevity and integrity of the threads. Finally, the fasteners were torqued per standard torque values during assembly and staked to prevent from coming loose.

4.1.3 Testing

Figure 4.8: Vibration Test Setup, XMD (Left) and BRITE-Canada (Right)

To validate the modal characteristics of the XMD, low-level sine sweep tests were conducted by the author and SFL staff mechanical engineers at the SFL’s vibration test facility. The goal of the exercise was to ascertain the designed modal values and compare it against previous XPOD-GNBs that also underwent the same test parameters. For the latter, the BRITE-Canada satellite was baselined as it had recently
gone through its acceptance testings just prior to the XMD. As such, the XMD was set up to mirror the BRITE-Canada configurations on the test stands, including the positions of the accelerometers to nullify any potential bias. The characteristics of the test cases were a constant 0.5 g acceleration, 5.0 Hz to 2000.0 Hz, harmonic vibration spectrum. This was performed for all three primary axes to capture a global perspective.

Figure 4.9: Vibration Test Results, FNF, BRITE Canada vs. XMD

The results from the vibration tests showed that the BRITE-Canada and XMD have a first natural frequency of 128 Hz verses 145 Hz, respectively. This is a global mode with the unit bending in the Z-axis direction. This was unexpected as the computer simulation of the XMD assembly was intentionally tuned to 128 Hz as per BRITE-Canada data. Mismatches between FE analysis and test results are not uncommon and computer simulations are never to be trusted in their entirety. However, the disparity should be determined for posterity. By revisiting the FE results, the author was able to attribute the root cause to corrupted surface contact constraints that caused the FE assembly to be less stiff and resulted in lower than actual First Natural Frequency (FNF) predictions. This was rectified and a revised FE analysis showed a FNF value of 148 Hz, which was of sufficient agreement to the vibration test data.

To further ensure the results were not fortuitous, two other FE assemblies using dissimilar meshes were analyzed to achieve result convergence. These gave produced results of 147 Hz and 151 Hz and validated that the FE could closely match the vibration test result. The XMDs were never modified to match the FNF of the BRITE-Canada assembly, however. The difference was not deemed significant to warrant the effort, given that the XPOD-GNB combination have shown FNF between 100 Hz and up to 150 Hz across different missions. The XMD is within this spectrum, and future refinements should be done on a mission by mission basis, as needed. Nevertheless, it is important that the disparity between theory and tests have been understood and future confidence can be gained both in process and design.

As of writing, the two XMDs have been used in support of the NLS-11 launch campaign in Yasny, Russia, on behalf of the BRITE-Toronto and BRITE-Montreal satellites. The two XMDs were successfully used for launch vehicle-level fit checks on the Dnepr rocket, which saw them integrated to the launch payload stack alongside mass simulators of companion satellites. Inertial load tests of the launch stack in the centrifuge facility were also accomplished without issue.
Shortly after the NLS-11 fit checks, one of the XMDs was used in India to validate a new integration procedure in support of the NLS-7 launch for the CanX-4 and CanX-5 satellites. The Indian Space Research Organisation (ISRO) had instituted an alternate stacking method for the Polar Satellite Launch Vehicle (PSLV) rocket, which necessitated tests and fit checks in accordance to the new procedures. In preparation, the author was tasked to quickly adapt the existing XPOD-GNB support ring GSE to the XMD, used to handle and crane the assembly. Owing to the similarities between the XMD and XPOD-GNB geometries, as intentional by design, the author was able to accomplish this with only minimal redesign and rework of both the XMD and the GSE. The modified support ring was made to fit the XMD in the same manner as to the XPOD-GNB, using existing hardware and ensuring the same crane setup. Fit checks on NLS-7 with the XMD was successfully accomplished in June 2014.
Chapter 5

Conclusion

Significant work has been done towards the mechanical developments for the CanX-7 and the NEMO-HD satellites. The former, a nanosatellite that will demonstrate a slew of new technologies, is a successful evolution of the 3U bus form factor that incorporates proven heritage hardware and caters to unique payload requirements. The latter, a microsatellite that supports a high-resolution Earth observation payload, advances the knowledge in structures and materials at SFL with its use of honeycomb sandwich panels in an unorthodox monocoque architecture. Across these two platforms, the author elucidated on how aspects and approaches to design, analysis, manufacturing, integration, and testing were successfully implemented. In addition, an initiative to create a mass dummy for the XPOD system was successfully carried out, which made possible considerable efficiencies and flexibilities in the testings of the XPOD-GNB combination. With these three programs, the author has provided a perspective on mechanical developments for microspace applications that spans the early stages of one mission phase through to the continued support for another operational system.

With the CanX-7, NEMO-HD, and the XPOD mass dummy, the author must impress upon the role of the mechanical subsystem as an integral and supportive part of a microspace mission. Conscious efforts were made to cater to subsystem attributes in the design of the satellite layout and integration processes. The structural components of the satellites were designed to offer adequate mechanical support for subsystem hardware during the most rigorous phases of launch. Analyses were done to assess the integrity of each bus, and also to ensure its on-orbit characteristics are complimentary with those of other equipment. Beyond the satellite, the mechanical subsystem helped to enable successful functional testings and system-level validations in the form of ground support equipment. Finally, it was used throughout the entirety of the mission development by providing support and protection during assembly, storage, and transportation.

It is the author’s desire that this thesis may be an inspiration, and offer lessons learned, for future mechanical engineers in the microspace field. It is also the hope of the author that the works contained herein may help to expand the foundation of its sciences, both at SFL and abroad.
Figure 5.1: NEMO-HD (Foreground) And CanX-7 (Background) Under Assembly At SFL
Bibliography


