Payload System Design of a CubeSat Distributed Telescope

AE 8900 Special Problems Report

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The Virtual Super-Resolution Optics with Reconfigurable Swarms (VISORS) mission is a novel CubeSat formation distributed telescope mission that aims to investigate the underlying energy release mechanisms in the solar corona. VISORS is a mission that was initially conceived in the National Science Foundation (NSF) CubeSat Innovations Ideas Lab workshop held in 2019. The mission will observe the corona in extreme ultra-violet (EUV) at an angular resolution of less than 0.2 arcseconds using two 6U CubeSats that align and fly 40 meters apart to form a distributed telescope. Achieving such a mission requires key technologies in the fields of diffractive optics, inter-satellite communication, CubeSat propulsion, and relative navigation. The development of any single one of these technologies is novel but all of them working in conjunction truly enables the VISORS mission. The consolidation of these technologies into the Cubesat form factor poses a mechanical and systems design challenge. This paper focuses on the preliminary payload design of the VISORS CubeSats, the challenges inherent with combining the key technologies into a 6U form factor, and the key next steps to mature the payload design. Working in conjunction with 10 different universities and a projected launch in late 2023, the VISORS mission will demonstrate the capabilities of CubeSats to perform high precision coronal imagery and will pave the path forward for future CubeSat swarm missions.

I. Introduction

A. Mission Overview

While solar physicists have investigated the Sun and have proposed theories on its underlying physics, the question of coronal heating continues to be unsolved. What is unknown despite countless coronal observations is why the solar corona is 1000 times hotter than the visible surface of the Sun and what drives this stark difference in temperature [1]. Based on current observations, the hypothesis is that most of coronal heating is confined to narrow current sheets of filaments on the order of 100 kilometers wide in which energy is dissipated into the coronal plasma. The conjecture related to the underlying release mechanisms is encompassed as a "major outstanding science question" in the National Science Foundation (NSF) Geospace Section planning document: "How magnetic reconnection works and operates in the solar atmosphere, within the solar wind, at the dayside magnetopause, and in the magnetotail to initiate and facilitate energy transfer between the different regions of the space environment" [2]. Coronal heating is best observed in the extreme ultraviolet (EUV) because of the highly ionized atoms that compose the solar corona; however, the challenge of observing the current sheets is the required 0.15 arcseconds of image resolution needed from Earth orbit. Observing at such a high resolution is infeasible with current imagers and with mirror optic technologies. The desired angular resolution is feasible only by using a diffractive optic technique based on the Fresnel zone plate method [3], but the required focal length when using such diffractive optics makes it infeasible to design a conventional space telescope of that size (approximately 40 meters).

The Virtual Super Resolution with Reconfigurable Optics (VISORS) mission is a distributed space telescope that will observe these energy mechanisms by utilizing novel technologies and employing a reconfigurable cube satellite

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(CubeSat) multi-vehicle formation. The mission was conceived at the NSF CubeSat Innovations Ideas Lab in 2019 and initially employed a set of three 3U CubeSats that align along the sun vector to form a distributed telescope in Low Earth Orbit as shown in Fig. 1. The instrument's distributed nature allows the formation to achieve the focal length needed to incorporate a diffractive optic technique that achieves the desired angular resolution required to observe the underlying energy release mechanisms in the solar corona. The VISORS mission proposal was selected by NSF in 2019 to proceed with mission development.

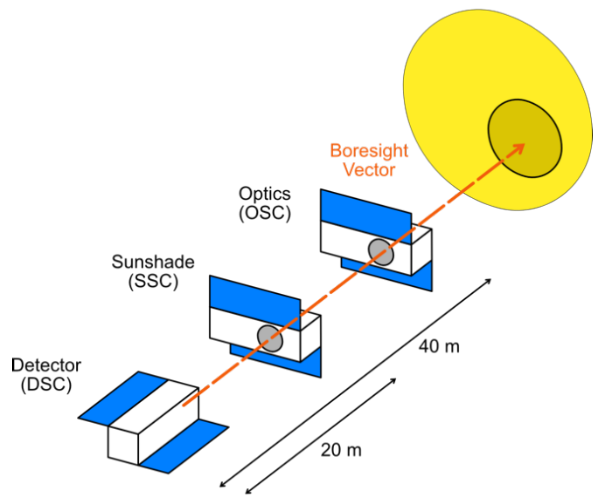

Fig. 1 VISORS Satellite formation when in alignment to collect coronal imagery.

As the design matured, it was realized the initial 3U distributed telescope was infeasible due to constraints inherent to the 3U CubeSat form factor [4]. The constraints manifested in two ways: volume limitation and over-constrained pointing requirements. The volume limitation of the 3U CubeSat form factor reduced the capabilities of key technologies required for the mission. Furthermore, the limited available volume in the spacecraft bus constrained the mass and volume margins early in conceptual design for key technologies, leading to minimal margin for physical growth of subsystems. In addition, the reduced surface area of the 3U CubeSat form factor led to over-constrained pointing requirements. Simply put, not all the pointing requirements could be met during mission operations for the different technologies on board when needed; the ramifications were reduced scientific merit and increased mission safety risk. Overall, both sets of constraints impacted the likelihood of mission success for the VISORS mission. The mission team conducted a mission architecture trade study to analyze alternative CubeSat formations that could resolve the challenges with the initial 3U formation concept. The outcome of the trade study was a new two vehicle 6U formation [4]. With increased surface area and available volume for each spacecraft, the 6U form factor presented a solution to the challenges with the 3U form factor, and the additional space also allowed for increased capability of subsystems [4]. Additionally, the switch to two satellites reduced mission operational complexity [4]. The form factor change did not have a major impact to the mission operations with the main premise of aligning with the sunline remaining the same for coronal observation as shown in Fig. 2. While the trade study required the mission team to redesign key technologies, the architecture change provided a key path to achieve the desired scientific merit and to progress the spacecraft design to what is discussed in this paper.

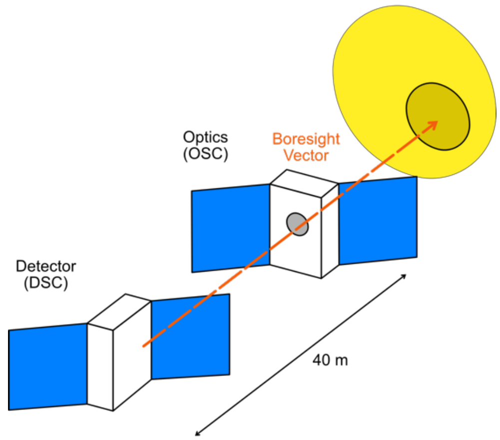

Fig. 2 Updated VISORS satellite formation with a two 6U formation.

B. Concept of Operations

Fig. 2 visualizes how the two 6U satellite formation, when aligned, forms the distributed telescope to capture images of the solar corona. The optics spacecraft (OSC) is the satellite in closest proximity to the sun and houses a photon sieve, which is a diffractive optic technique based upon the canonical Fresnel zone plate [3]. The photon sieve is placed off-center in the OSC which allows the formation to rotate about the boresight vector to image various regions of the sun. The detector spacecraft (DSC) is the second spacecraft and stores the camera, corresponding sensors, and a processor to capture the instrument measurements. The coronal region this telescope can image is highlighted inside the darkened circle in Fig. 2. The OSC serves a dual purpose by also blocking unwanted EUV from entering the detector through the surface area perpendicular to the boresight vector of the detector. The 40 m separation is due to the focal length required to achieve the desired angular resolution using the photon sieve and detector.

The operations for the VISORS mission are governed and stratified by their relative orbit geometries into two operating modes: standby and science operations. In standby mode, the spacecraft are separated further out with their minimum relative distance being 200 meters. In science operations mode, the minimum relative distance is reduced to 40 meters, the required focal length when performing a science observation. The relative orbits are established using a passive collision safety method known as e/i vector separation [5]. This relative orbit technique is defined by the alignment of the relative eccentricity and inclination vectors; the technique is preferred over other methods because it provides passive separation between spacecraft and is robust in the presence of perturbations and uncertainty. The reason the mission is stratified into two distinct relative orbits is to reduce stationkeeping maneuvers required when the spacecraft are closer in science operations and to increase passive collision safety of the formation. When the spacecraft are relatively further apart like in standby mode, e/i vector separation allows for the relative orbits to be passively stable for longer, allowing for multiple orbits of passive collision avoidance as the relative e/i vector separation degrades overtime. Transferring between the two relative geometries requires translational propulsive maneuvers to change relative orbit geometry.

The differing geometries also impact the operations conducted in the two modes. During science operations mode, the primary focus is to align the spacecraft and perform science operations. When in a favorable location along the orbit, the formation will drift into the alignment depicted in Fig. 2 to complete a measurement. The alignment shall be maintained for a minimum of 10 s to obtain the required exposure for the detector. To maintain this alignment, steady communication between the spacecraft is necessary to ensure the spacecraft attitudes and alignment is correct for a science observation at the desired resolution. Several attempts to gather data will be executed before the formation exits the science mode relative orbit and returns to the standby formation. Preliminary Monte Carlo simulations of the formation demonstrate that the spacecraft alignment requirements for observation cannot be met for every observation attempt, so maximizing the number of attempts increases the likelihood for mission success. Standby mode, on the other hand, is focused on routine mission operations such as recharging the spacecraft battery, communicating with the ground, and remaining in standby until more science observations are desired. The spacecraft will downlink the collected science data and perform its maintenance operations before another uplink command is delivered to collect further observations once the downlinked science data is reviewed by the science team. The concept of operations for the VISORS mission is summarized in Fig. 3.

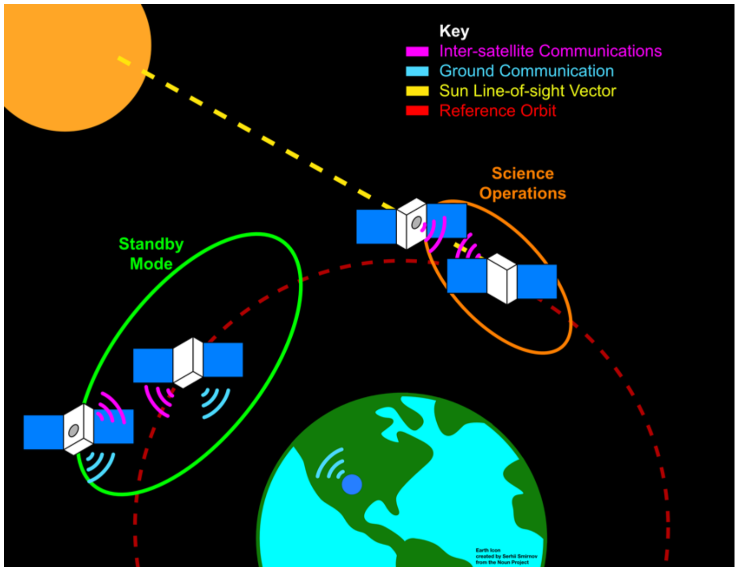

Fig. 3 Concept of operations for the VISORS mission.

II. Advanced Technology Requirements

Achieving the angular resolution desired requires key technologies that must work in conjunction to enable the VISORS mission. Discussed in this section are the advanced technology requirements that provide the foundation for payload component and overall payload design development.

A. High-Precision and High-Accuracy Relative Navigation

While the distributed telescope can provide the angular resolution required to observe the corona, it is sensitive to the relative position and velocity errors when in alignment for data collection. Positional errors and velocity errors can cause a loss in focus or blurring of the image data that is being collected. Due to this sensitivity, there are strict requirements on the margins of error for key degrees of freedom as shown in Fig. 4 when aligned for an observation. The longitudinal deviation of the OSC with respect to the DSC must be maintained to 15 mm in order to maintain focus during an observation while the OSC cannot shift laterally beyond 18 mm. To prevent image blur, the lateral drift cannot be greater than 200 μm/s. These requirements are more fully described in [7].

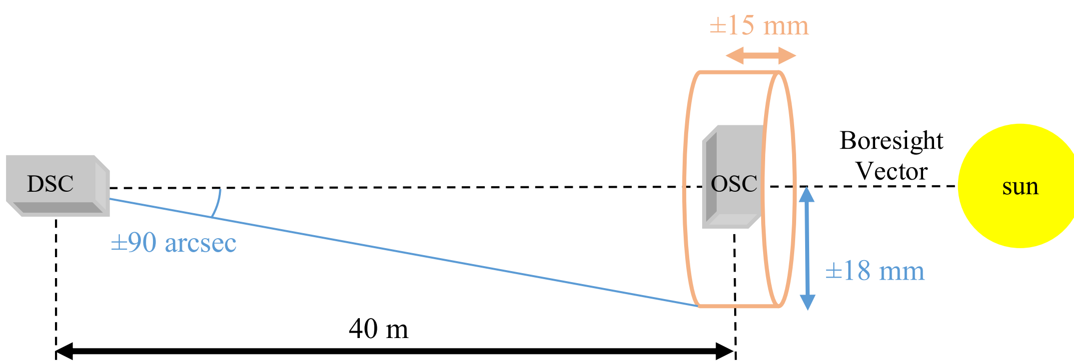

Fig. 4 Relative state requirements for the formation when aligned for collecting observations.

B. Collision Avoidance

The required performance margins of error pose a significant design challenge, and the close proximity operations of the spacecraft during observation poses a risk of collision. A collision would be catastrophic, mission ending, and pose a risk to other satellites in Low Earth Orbit. Due to the consequences of a collision, collision safety is a top-level mission requirement that flows down to active and passive safety considerations made at the subsystem level. Collision safety is considered in two ways: passive and active. Passive collision safety is provided by e/i vector separation as it provides minimum separation in the plane perpendicular to the flight direction. The passive safety degrades as the relative e/i vectors become perpendicular [6]. The rate of degradation is a function of the geometry of the relative orbits; stationkeeping maneuvers are required more frequently when spacecraft are closer together as the relative e/i vectors become aligned more quickly.

Active collision safety is also essential in the event passive collision safety is compromised or if the risk of collision is unacceptably high. This could occur if the spacecraft are exhibiting off-nominal behavior such as when a component has failed, or a spacecraft has lost the ability to perform stationkeeping maneuvers. The ability to perform an active collision safety maneuver requires the spacecraft to have access to accurate knowledge of the dynamic states of the formation, to accurately propagate the formation states, to compute a propulsive active collision maneuver, and to execute the computed propulsive maneuver.

C. Omnidirectional Inter-satellite Communication

The need for relative navigation and maneuver planning for both science observations as well as collision avoidance spurs the need for inter-satellite communication. Inter-satellite communication is essential for passing satellite state information between spacecraft. The ability for the spacecraft to communicate also allows the capability for them to have accurate attitude knowledge to ensure the spacecraft are aligned along the DSC's boresight vector when performing a science observation. Furthermore, it is essential in the event a spacecraft is behaving off-nominally, to allow the other spacecraft to perform an active collision avoidance maneuver in the event formation safety is compromised. Due to the nature of the relative orbits, the spacecraft do not maintain a consistent relative attitude with one another. To ensure mission safety at all times, a spacecraft must be capable of communicating irrespective of its relative attitude with the other spacecraft; this imposes the requirement for omnidirectional inter-satellite communication. Omnidirectional communication also prevents coupling with the pointing requirements other sensors impose on the spacecraft. For example, the GNSS antenna for position and velocity estimation must face zenith, so it has access to the GNSS constellation. Omnidirectional communication prevents a scenario where the pointing requirements of multiple components interfere with one another; a situation such as this may cause loss of satellite communication, increasing collision risk. Moreover, in the event that a spacecraft loses the ability to control its attitude, the omnidirectional requirement ensures communication between satellites is not lost.

D. 3 Degree-of-Freedom Propulsion

Both spacecraft require an onboard propulsion system for relative maneuvers to be executed. This capability is needed to transfer between relative orbits, to meet the margin of error during science operations, to perform relative orbit station keeping maneuvers, and to perform active collision maneuvers in the event of a collision risk. The requirement for 3 degree-of-freedom (3DOF) propulsion arises from the complexity of the mission operations and need for frequent translational maneuvers. As with the inter-satellite communication, having the capability to perform a propulsive maneuver without needing to change spacecraft attitude decouples the use of the propulsion system from the pointing requirements of other sensors. Furthermore, it is disadvantageous for the spacecraft to constantly slew when propulsive maneuvers are desired as time needs to be allotted for the spacecraft to complete its slew, settle to its new attitude, and reacquire accurate knowledge of its attitude using onboard attitude sensors. During quick propulsive maneuvers prior to a science observation, a lack of a 3DOF propulsion system may lead to a loss in relative state accuracies in part due to rapid slew maneuvers and potential pointing requirements not being met such as the GNSS antenna not being zenith pointed. In addition, a 3DOF system requirement ensures the spacecraft can perform an active collision maneuver quickly regardless of orientation; this capability assists in improving collision safety.

III. Enabling Technologies

Key enabling technologies are essential in meeting the requirements discussed and truly make the mission possible. A discussion is warranted on these technologies to provide the foundational context required to discuss the development of the overall spacecraft payload design.

A. Spacecraft Bus

The spacecraft bus is responsible for the supplementary functions that each spacecraft must have to operate and is responsible for housing the payload technologies for this mission. The VISORS mission intends to use the Blue Canyon Technologies (BCT) 6U bus. One reason for this decision is the flight heritage and Technology Readiness Level-9 (TRL-9) of the BCT bus [7]. Furthermore, the spacecraft includes the BCT Avionics Module shown in purple in Fig. 5. The BCT Avionics Module contains several flight proven subsystems necessary for the mission: the XACT-

15 Attitude Determination and Control System (ADCS), the Electrical Power System (EPS), the Command & Data Handling (C&DH), and the UHF Ground Communication system (COM). The XACT-15 also meets the precise attitude estimation and control requirements needed for science observation as proven by the ASTERIA mission [8]. The EPS includes a six solar panel array shown in the schematic Fig. 5 where four of the solar panels are fully populated. The EPS also includes batteries, necessary power delivery equipment, and power monitoring checks. Additionally, the bus includes the Novatel OEM719 GPS receiver that is compliant with the GNC software needed for relative orbit maneuvering as well. The ground communication system incorporates the SpaceQuest TRX-U UHF radio and a monopole antenna. Due to the flight proven nature of the BCT bus and its included components, the VISORS mission team can focus on the development of subsystems unique to this mission while avoiding the increased risk of internally developing the subsystems that the BCT bus provides.

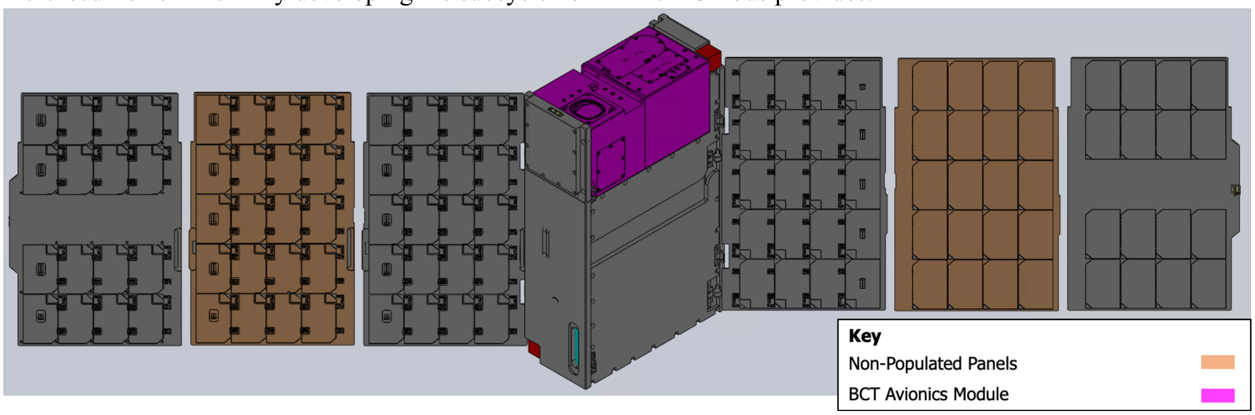

Fig. 5. BCT bus with deployed solar array.

To aid in the discussion of payload system design later in the paper, the body frame and key geometries of the bus are shown in Fig. 6. The orientation of the BCT Avionics Module in this picture shows the BCT Avionics Module on the -Z face on the face of the spacecraft. The GNSS antenna and star tracker are also located on the -Z face. The X faces of the spacecraft as well as the -Z face is composed of the spacecraft chassis. They each also contain a sun sensor array used for attitude estimation and coarse sun pointing; Fig. 6 shows the location of the sun sensor array on the -X face. Furthermore, the Y faces consist of panels that enclose the available 4U of available payload volume. The ground communication antenna will also be fastened to the Y panel, but the location of the antenna has not been determined by the BCT at the time of this report.

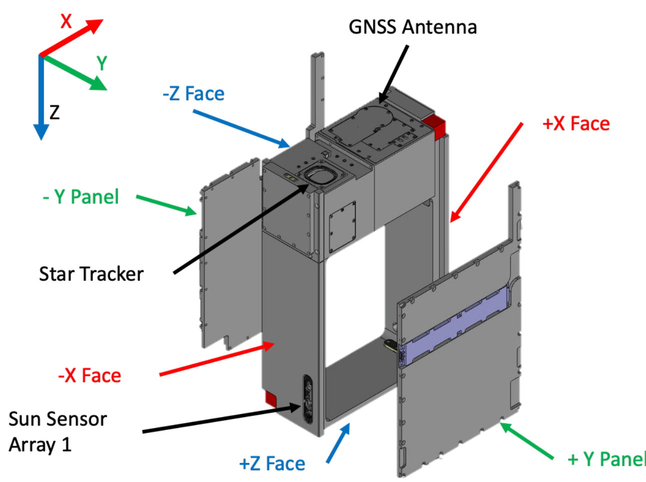

Fig. 6 BCT bus body frame definition, key geometries, and locations of bus sensors.

The payload components housed in each spacecraft's available payload volume share many technologies while the science instruments are unique to each. Unique to the OSC is a photon sieve based on the Fresnel zone plate [1]. The photo sieve is an aluminum coated silicon membrane 75 micrometers thick that contains a precisely cut conical pattern. NASA Goddard Space Flight Center (GSFC), a VISORS mission partner, is responsible for fabricating these components following the heritage of previous photon sieves. The detector system houses the backside illuminated

CMOS sensor that contains with a visible light filter to capture the EUV radiation focused by the photon sieve. The detector contains Compact Spectral Image Electronics (CSIE) system developed by VISORS team member University of Colorado Boulder Laboratory for Atmospheric and Space Physics (LASP) for processing science observation data; the design is built up on heritage systems used in Polar NOx Sounding Rocket Missions. Also, unique to the DSC is an additional BCT star tracker to reduce attitude uncertainties when performing an observation. The other components shown in Fig. 7 are common in both spacecraft, providing similar capabilities on both spacecraft and are discussed further below.

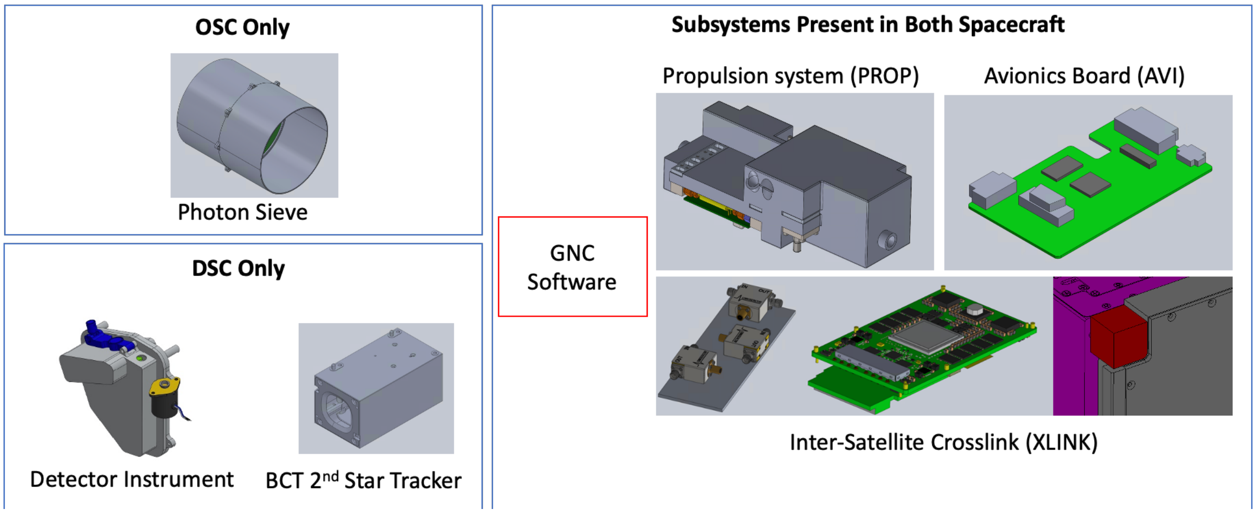

Fig. 7 Payload components of each VISORS spacecraft.

B. Guidance, Navigation, and Control (GNC)

To meet the requirement for relative navigation requirements for the mission, Stanford University, one of the partner institutions on the team, has proposed a solution to the relative navigation challenges through their Distributed Timing and Localization (DiGiTaL) system. The DiGiTaL algorithms use dual-frequency GPS pseudorange and carrier-phase observables provided by the onboard GNSS receiver to compute state estimates with <1 cm relative position accuracy and 1 m absolute position accuracy [9]. DiGiTaL utilizes inter-satellite communication to exchange GNSS measurements to compute the spacecraft relative states. Commonly pointed GPS antenna directions on each spacecraft are needed to maximize the number of commonly visible GNSS satellites.

GNC performs maneuver planning using two different schemes. When maximum accuracy is not required, closed form solutions are used with flight heritage from PRISMA and TanDEM-X [10] [11]. When maximum control accuracy is needed while performing science observations, a stochastic model predictive controller (SMPC) is applied [12]. The algorithm is designed to minimize propellant consumption by incorporating a fuel-optimal impulsive planning algorithm, which plans a maneuver consisting of 3-6 impulses in a user specified total ΔV cost [6]. The BCT flight computer allows allocation of computer resources to mission-specific software, reducing the need for additional processing elements in payload subsystems; the VISORS mission team intends to utilize this flexibility and host the GNC software on the BCT flight computer. The software is hosted identically in both spacecraft and takes on two distinct roles: deputy and chief. Because the GNC system operates in a deputy-chief architecture, the spacecraft in the deputy role is the only spacecraft performing propulsive maneuvers. The common payload systems and capabilities of the spacecraft allow the chief and deputy roles to be interchangeable during the mission, providing added flexibility on ΔV consumption. This interchangeability allows the total ΔV requirement for the mission to be split across the available ΔV of each spacecraft.

C. Inter-satellite Crosslink (XLINK)

The Inter-satellite Crosslink (XLINK) developed by Washington State University and the Ohio State University, partner institutions on the VISORS team, will accomplish the omnidirectional inter-satellite communication required for this mission. The crosslink system provides near full sky coverage with a communication range of 2 meters to 10 kilometers. The system works by having a set of patch antennae on opposite corners of the spacecraft. The antennae use a set of four radios, two of which are located on the FPGA board while the other two are found on the accompanying daughter board. Diplexers are used between the antennae and radios to combine the receive and transmit lines into one line leading to each antenna in the set. A gain amplifier is also used when transmitting to increase the bandwidth. The system transmits and receives using the 5.8 GHz frequency band. The FPGA in this system is responsible for the controller that activates one radio and one antenna at a time. It is also responsible for discovering which antennae can maintain the link as the spacecraft's attitude changes over time, during link establishment, and link maintenance. The XLINK system layout is shown in Fig 8.

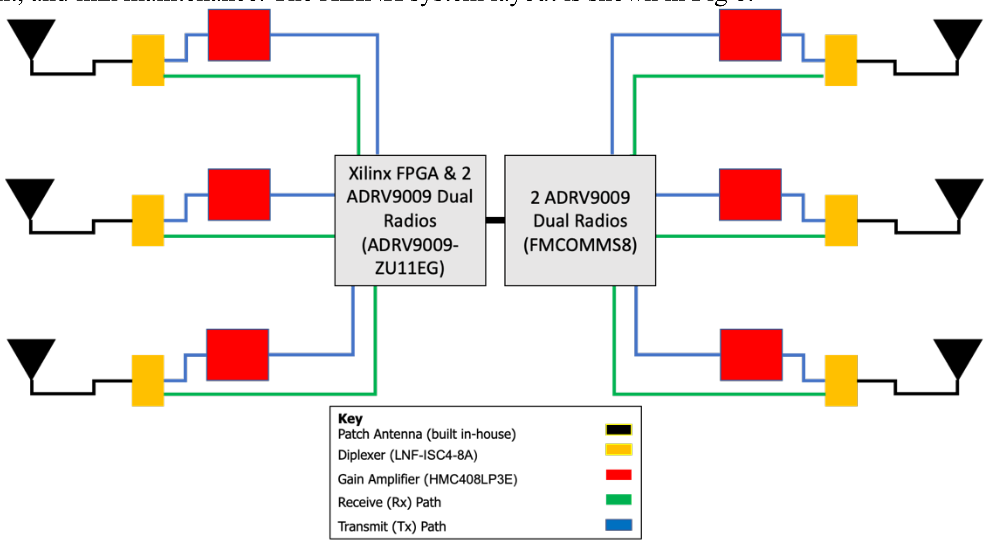

Fig. 8 The XLINK system layout.

D. Propulsion (PROP)

As discussed, a 3DOF propulsion system is a key technology requirement which is needed for mission success. The VISORS mission utilizes a 3D printed cold gas thruster propulsion system developed by team member Georgia Tech. The system currently has a TRL-6 as it leverages from previous systems designed for the BioSentinel, INSPIRE, and Ascent CubeSat missions [13]. In Fig. 9, the main dimensions and locations of components are shown. The system consists of six nozzles that achieve the 3DOF requirement. Due to the surface availability, a nozzle cannot solely point along the -Z direction. To ensure 3DOF is achieved, two nozzles are aligned along the spacecraft body X-axis while the remaining four nozzles are canted by 45° in the YZ plane. The canted nozzles can produce a thrust component along a ±Y and a ±Z direction. The system has a ΔV budget of 8 m/s, allowing the two spacecraft to have a total ΔV budget of 16 m/s.

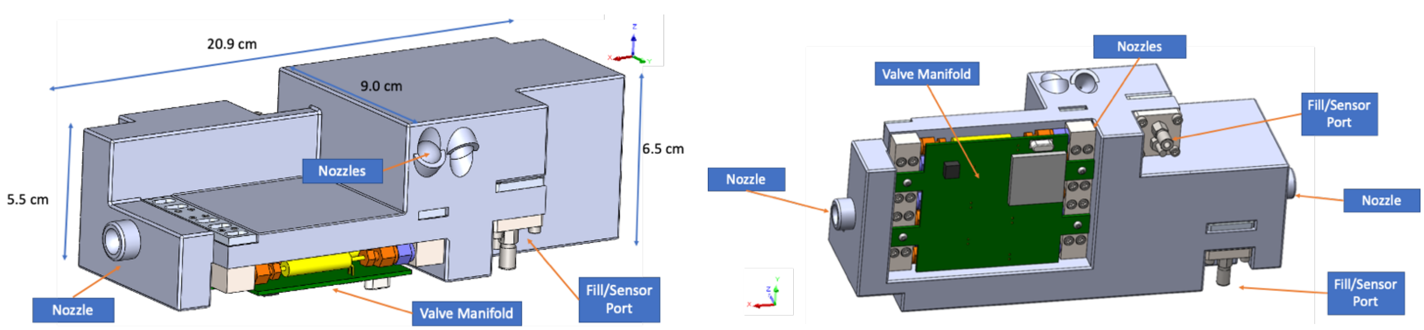

Fig. 9 Propulsion system CAD model and component layout.

E. Avionics Board (AVI)

The avionics board performs several functions that ensure the different payload subsystems work together and with the spacecraft bus. First, it manages data interfaces between the BCT bus and the different payload subsystems: propulsion, XLINK, and detector instrument. The board also provides nonvolatile storage for science observation data and telemetry data for downlinking. Finally, the board is also responsible for distributing and monitoring power from the bus to the payload subsystems. Fig. 10 shows the preliminary board layout for each spacecraft.

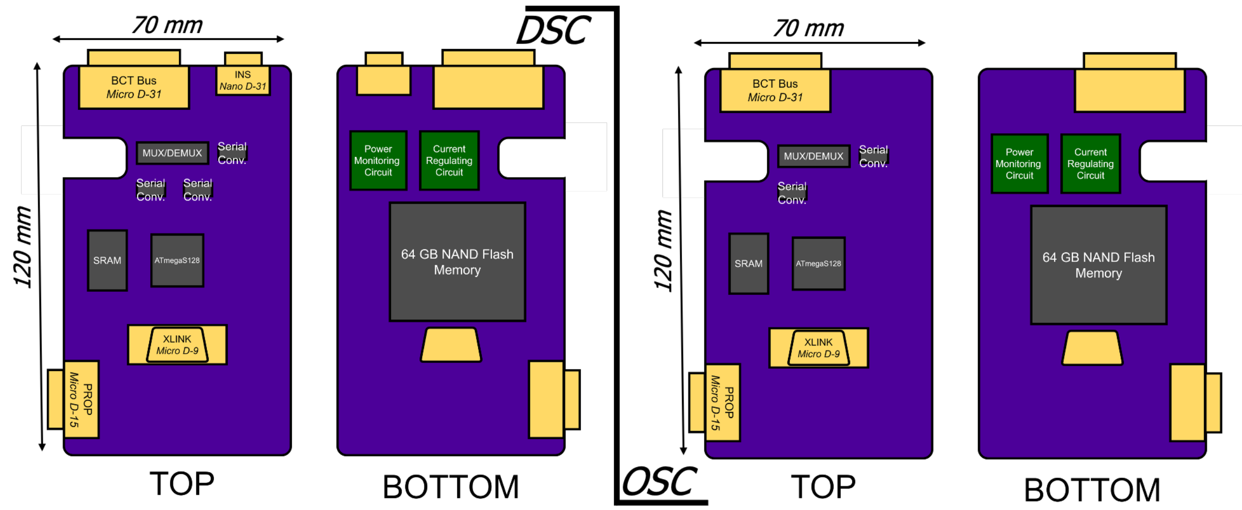

Fig. 10 DSC and OSC avionics board layout.

IV. Payload System Component Placement

With a general understanding of the major subsystems and enabling technologies, the discussion can now be focused on how all these technologies are integrated into the spacecraft bus and how they interface with one another. Incorporating iterative design techniques, the placement of payload components was considered as the design of technologies matured and the requirements for each component solidified. The preliminary design shown here in Fig.11 and Fig. 12 and discussed in the following sections is the result of several design iterations, negotiation within the team, and validation that all requirements are met.

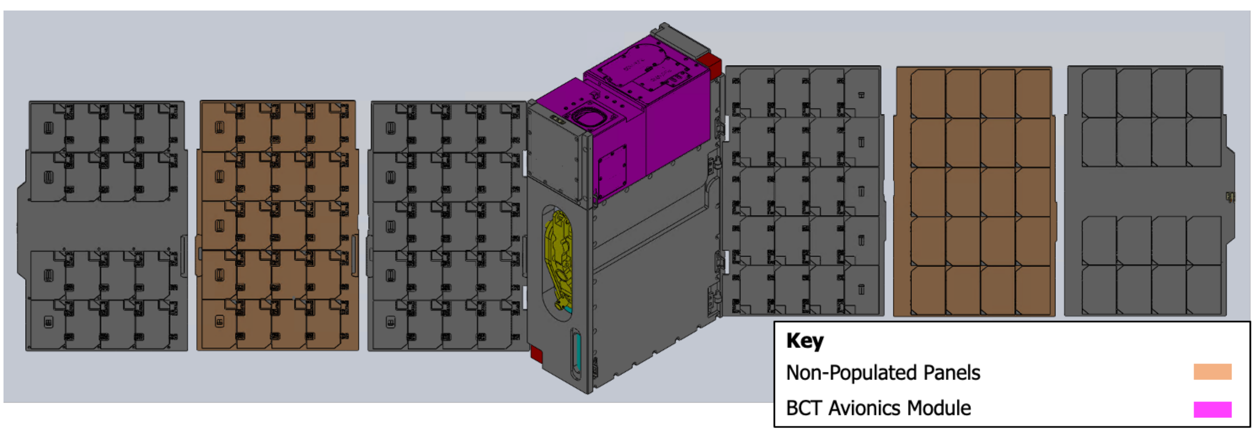

Fig. 11 DSC spacecraft with deployed solar panels.

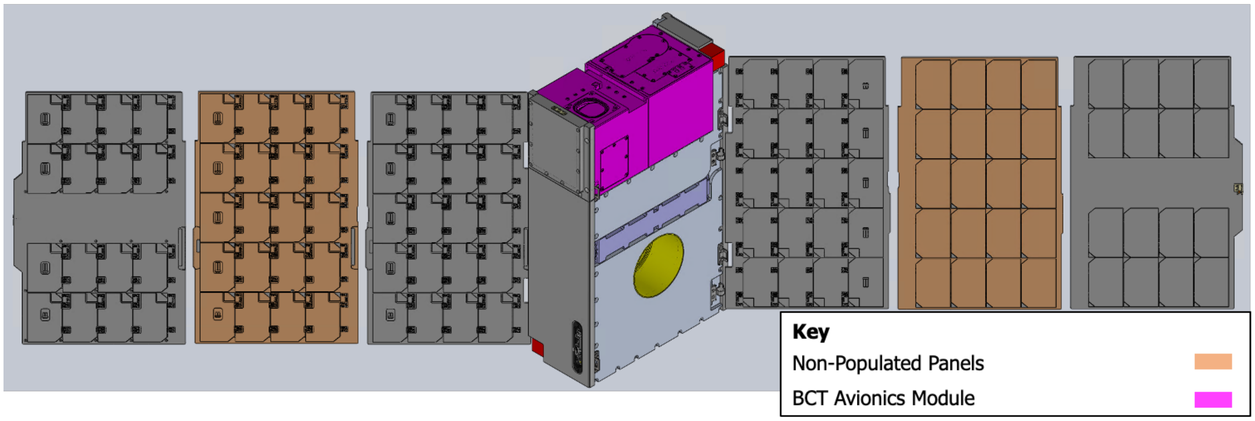

Fig. 12 OSC with deployed solar panels.

A. Component Placement Requirements

Before discussing the payload component placement for the spacecraft, it is important to highlight the key placement requirements that drove where subsystems were placed. The placement requirements can be categorized into hard and soft requirement categories. Focusing on the hard requirements, it is important that science instruments are placed properly to ensure the desired instrument performance is met for the mission. In the DSC, this manifests with the location of the mirror and the virtual image location as their locations to the center of mass effect the resulting image blur when taking observations. The science team concluded that the desired coronal imaging performance is met when the detector mirror is near the geometric center while the virtual image is 90 mm away from the spacecraft's geometric center; Fig. 13 visualizes this requirement. On the OSC, image blur is a function of the distance between the center of mass and center of the photon sieve pattern; the science team's analysis demonstrated the need for the optical center to be 9-15 mm away from the geometric center and for the photon sieve center to be 50 mm away from the optical center; a visualization of this requirement is shown in Fig. 14. Consequently, the detector boresight must be in line with the 2U x 3U face to ensure the mirror is near the geometric center. Furthermore, the DSC's second star tracker's boresight must be antiparallel to the detector boresight to reduce attitude uncertainty along the detector's boresight vector. Another hard requirement is the location of the XLINK patch antennae; a set of three patch antennae must be located on diagonally opposite sides of the spacecraft as this configuration allows for near full sky coverage for inter-satellite communication.

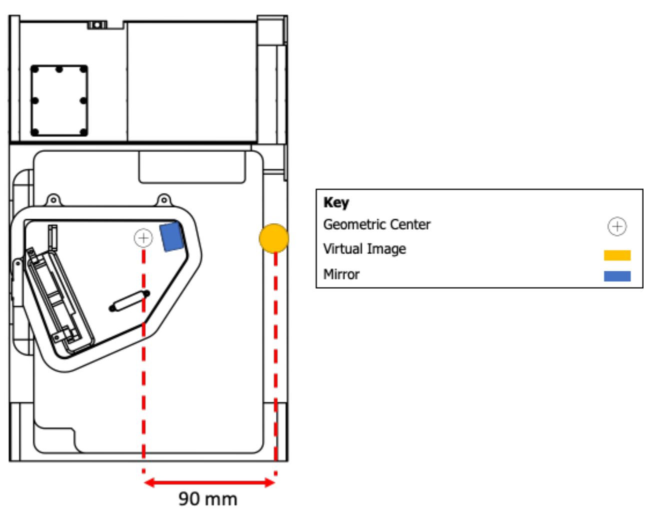

Fig. 13 Virtual image location requirement for detector instrument.

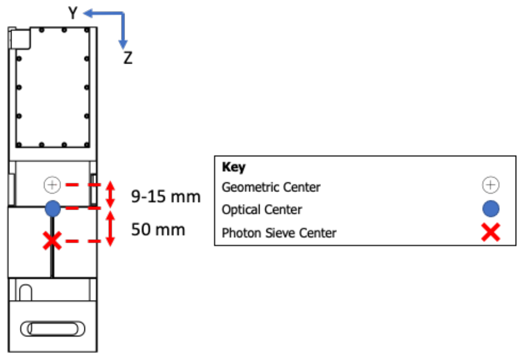

Fig. 14 Photon sieve and optical center location requirements in OSC.

Soft component placement requirements do not impact mission success but achieving these requirements reduces design complexity. For instance, the propulsion system must have access to as many sides of the spacecraft as possible to provide 3DOF. While the minimum number of sides is three to ensure 3DOF, this configuration complicates nozzle placement as well as internal design and couples thrust maneuvers with different actions, which increases complexity when performing propulsive maneuvers. The XLINK system becomes more complex as well if the XLINK FPGA and radio boards are not equidistant to the sets of antennae in the opposite corners of the spacecraft. Not only does this increase complexity for wiring, but it can also induce some line loss between each antenna and the radios. The avionics board must also be centrally located to ensure interfacing with all payload subsystems; a non-optimal location increases complexity of the board layout and wiring of components. Not all soft component placement requirements carry the same weight. To discern this, Table 1 lists the order of these requirements in the order of most importance. Even though soft component placement requirements do not hinder the key requirements from being met, they are considered to reduce design complexity.

Table 1: Hard and soft requirements for component placement.

Hard Requirements (no order)	
1	Virtual image of detector instrument must be 90 mm away from spacecraft geometric center
2	OSC optical center must be 9-15 mm away from the geometric center, and photon sieve center must be 50 mm below geometric center.
3	Detector boresight must be in line with the 2U x 3U face
4	Second star tracker boresight must be antiparallel to detector boresight
5	XLINK sets of patch antennae must be located on diagonally opposite sides of the spacecraft.
Soft Requirements (Ordered by highest priority)	
1	Propulsion system must have access to as many sides of the spacecraft as possible.
2	XLINK FPGA and radio boards must be equidistant to sets of patch antennae.
3	Avionics board must be centrally located in the spacecraft.

B. DSC Component Placement

Accounting for these considerations, the preliminary component placement for the DSC is shown in Fig. 15. Starting from the -Y panel, components are attached to a back plate that is mounted directly to the -Y panel. The plate is populated with the detector chamber, CSIE electronics, avionics board, and BCT's second star tracker. Based on the hard placement requirements, the detector boresight is in line with the 2U x 3U face of the spacecraft and Fig. 15 demonstrates that the mirror and virtual image are in the required locations. Furthermore, the ultrahigh frequency

(UHF) ground communication radio, 3DOF propulsion system, and each XLINK patch antenna set are mounted directly into the spacecraft chassis. The location of the 3DOF propulsion system provides access to five sides of the spacecraft, which is the maximum number of sides possible for a single module in the available payload volume. The 3D printed nature of the propulsion system is emphasized in component placement as it allows the system to utilize the available volume more efficiently than a conventional system. The system must adhere to a cutout as shown in the bottom left of Fig. 16 for the set of XLINK patch antennae and the location of the second star tracker as show in Fig. 16. Without this capability, achieving the placement requirements for the antennae and star tracker would limit the performance of the propulsion system through either a more complex design, reduced nozzle placement opportunities, or reduced propellant capacity. The other set of three XLINK patch antennae are best shown in Fig. 16 on the diagonally opposite side of the spacecraft near the BCT Avionics Module, providing the location necessary to achieve near full sky coverage by the XLINK system. The cutout panel near this set of patch antennae serves as a location to mount the diplexers necessary for the antennae. Finally, the remainder of components are mounted to the +Y panel; this includes the XLINK FPGA and radio boards along with the set of diplexers for the other set of three patch antennae underneath the propulsion system.

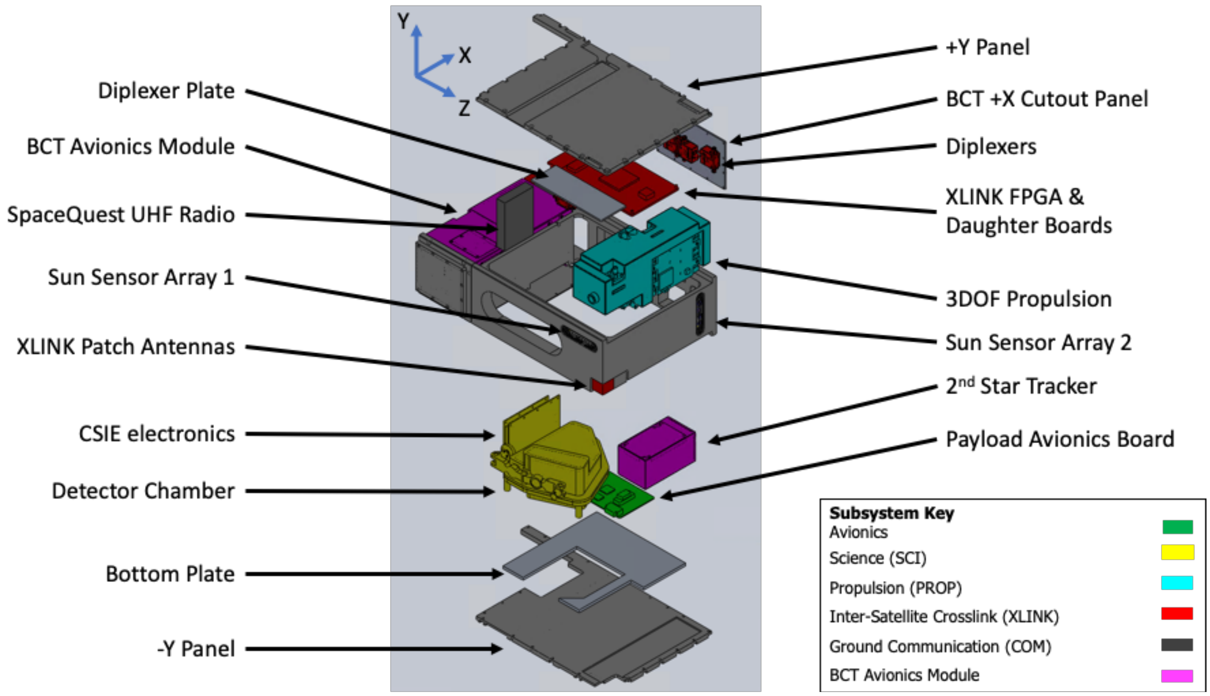

Fig. 15 Exploded view and payload composition of DSC.

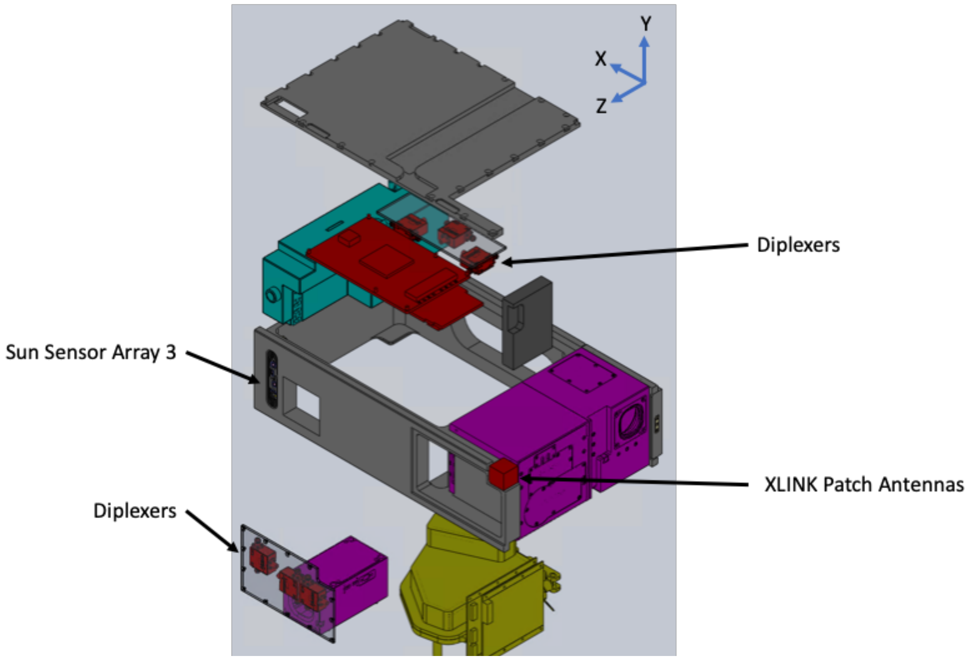

Fig. 16 Another exploded view of DSC payload components highlighting components not visible in Fig. 15.

C. OSC Component Placement

The placement of components in the OSC presents unique challenges and similarities to the DSC as shown in Fig. 17. Starting with the similarities, the XLINK antennae and 3DOF propulsion system are found in the same locations as on the DSC as these components are shared. However, the similarities end there due to the unique placement of the photon sieve along the centerline of the geometric center required for the coronal imaging performance desired. Consequently, the locations of the XLINK electronics and avionics board need to be modified from where they are found on the DSC. To meet these unique challenges, the avionics board is rotated clockwise by 90 degrees along the +Y body axis and moved down the Z axis in front of the photon sieve. To allocate volume for the XLINK electronics and diplexers for the patch antennae near the propulsion system, the components are vertically stacked off of the top plate as shown in Fig. 17. The other set of diplexers remain in the same location as in the DSC for the other set of XLINK patch antennae as shown in Fig. 18. While the diverging component placements of each spacecraft increases integration complexity, the utilization of the same components on both spacecraft greatly reduces development time for enabling technologies, a worthy tradeoff.

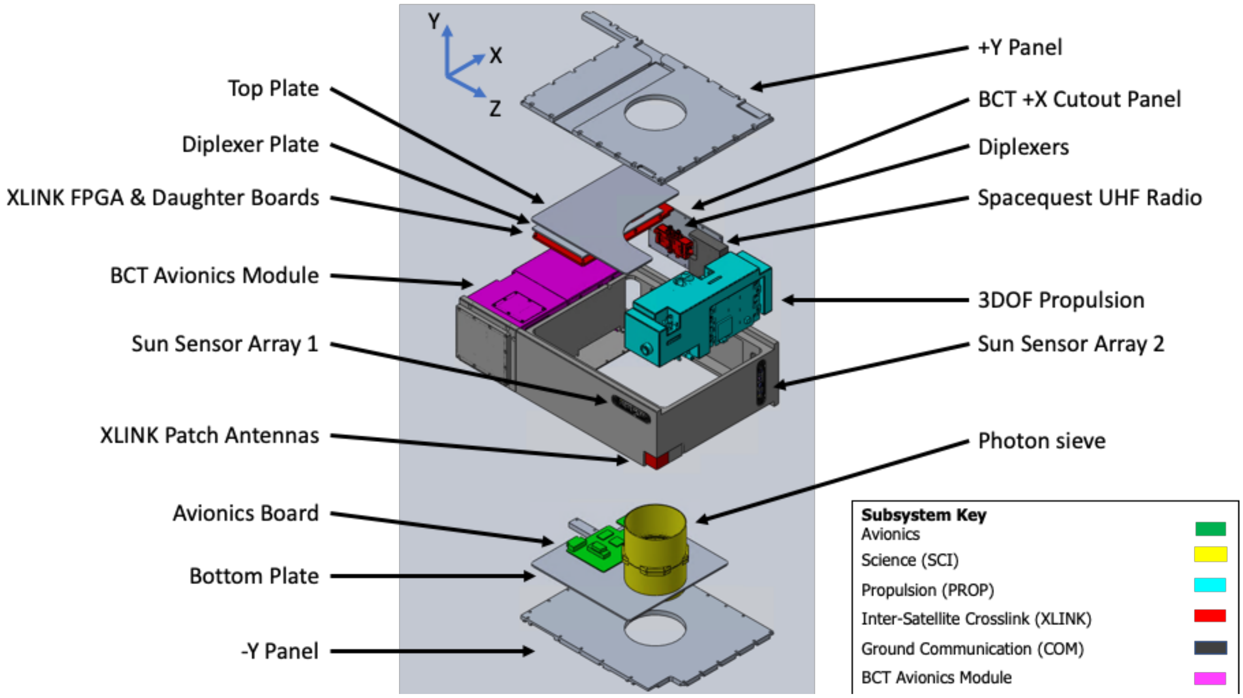

Fig. 17 Exploded view and payload composition of OSC.

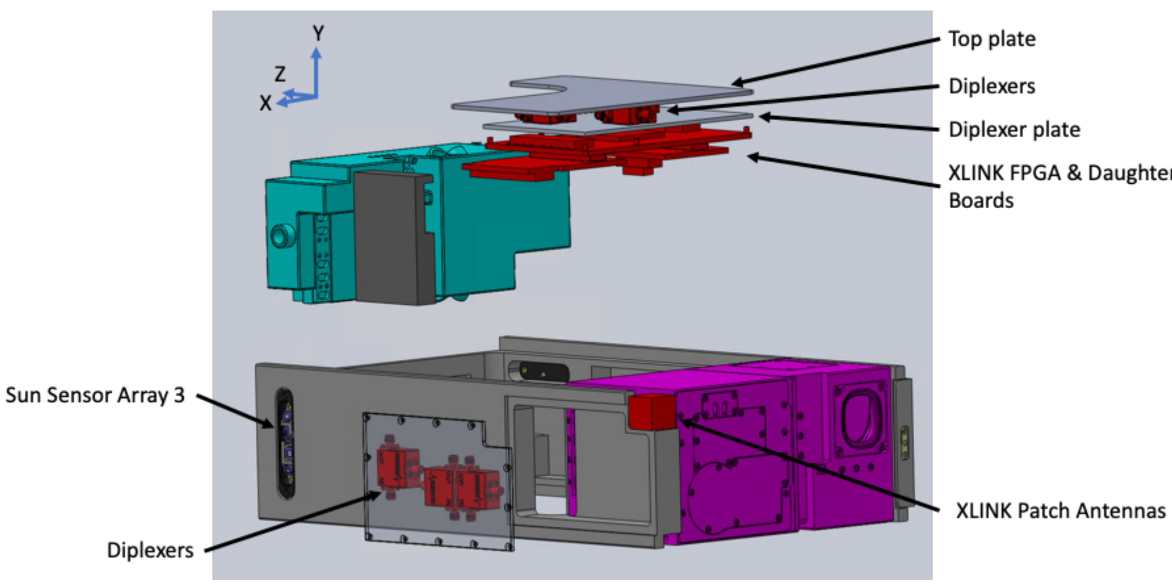

Fig. 18 Alternate exploded view of OSC highlighting components not visible in Fig. 17.

V. Electrical Interfacing

A. Electrical Interfacing Overview

The complex placement of components presents a unique challenge for electrical interfacing in both spacecraft. Not only must the electrical interfacing requirements align with the electrical constraints of the bus, but proper volume must be allocated for wiring and harnessing of these components. Provided in Fig. 19 is an overview of the electrical interfaces present in the spacecraft; this discussion will focus on the payload system components being developed by the VISORS mission. The specifics of BCT provided components and their electrical interface nuances are not discussed; however, volume considerations for wiring and harnessing are still made as they are located in the payload volume of the bus. As discussed earlier, one of the primary functions of the avionics board is to deliver power to the payload components. The specifics for each component and their respective wire harnesses are shown in Fig. 19. The consequence of three separate payload systems is the need to allocate space for three separate electrical interfaces. Furthermore, the XLINK system's complexity requires a multitude of interfaces between its many components. From the FPGA and radio boards, each of the six diplexers is wired with interfaces: one for receiving and one for transmitting signals. From the six diplexers, there is a single wired connection to the patch antenna. The specific harnesses for these connections are shown in Fig. 19. The complexity is reduced in the OSC as there are no science electrical interfaces; however, it is not trivial. The deviations in component placement in the OSC and DSC require unique electrical interface layouts.

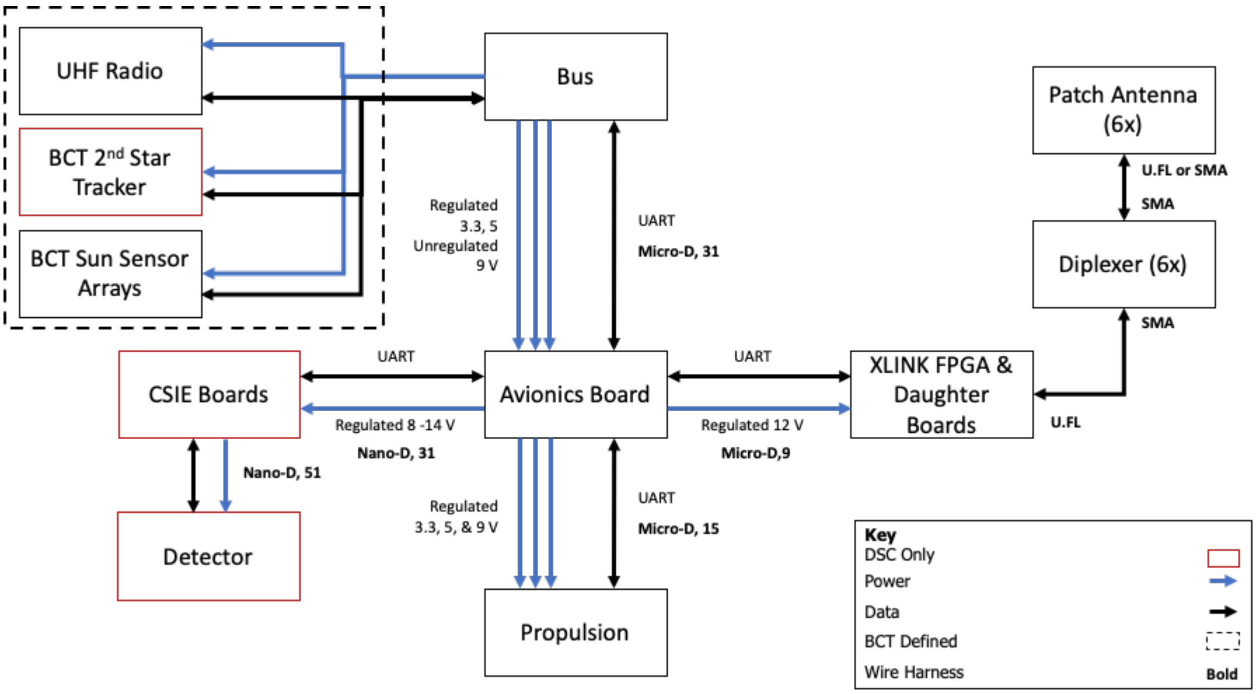

Fig. 19 Electrical interface diagram of components in the payload volume.

B. DSC Electrical Interfacing

The DSC requires the most complex wire routing due simply to the high number of components. The consequence will be a more complex full system integration but adhering to the volume constraints and placement requirements is necessary for mission success. Interfacing of payload subsystems is largely consolidated to the area around the AVI board as shown in the black border in Fig. 20. The wire routing is simplified for the detector instrument, bus, and XLINK due to their close proximity to the AVI board. The PROP system wiring gets more complex due to the location of the interface near the XLINK patch antennae as shown in Fig. 20 and means the interface is routed underneath the detector. The interface between the detector and CSIE boards are also routed underneath the detector as well, harnessing into the CSIE board above the bus harness on the AVI board. Volume considerations have been made for the UHF antenna which sits between the detector components and BCT avionics module; its location allows for interfacing with the BCT Avionics Module for power and data. The space in-between the XLINK and AVI boards is also used to route the second star tracker's electrical interface to the BCT bus.

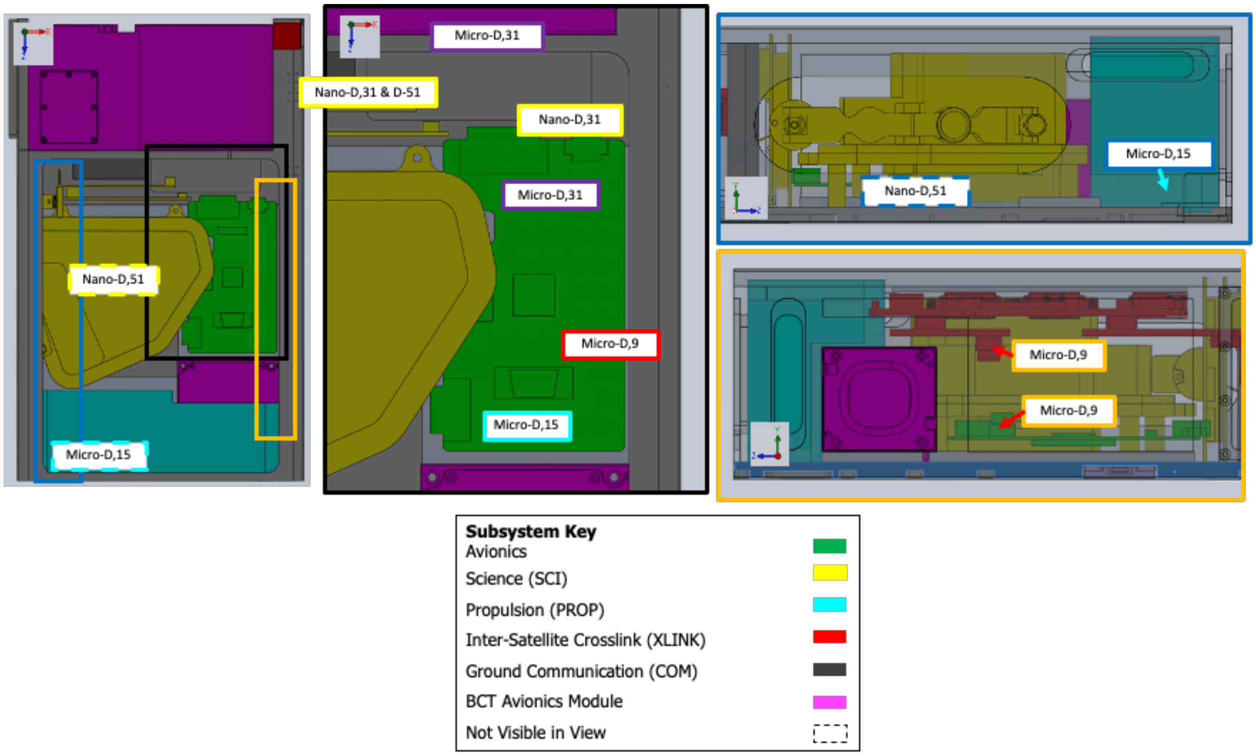

Fig. 20 Locations of key wire harnesses in DSC.

Due to the number of components and wired connections, electrical interfacing for the XLINK system is quite complex and the overall layout is shown in Fig. 21. Focusing on the set of antennae underneath the PROP system, the wires are routed similarly to the PROP to AVI electrical interface. However, due to the location of the diplexers on the +Y panel, the three patch antennae interfaces must vertically ascend from the -Y panel to the diplexers and travel along the +Y panel to their respective SubMiniature version A (SMA) harnesses. From there, the six interfaces from the diplexers are routed to the U.FL connectors on the FPGA board. U.FL connectors are capable of rotating, which reduces the issue of routing so many wires in such a small area. Traditional harnesses have reduced capability here as they are fixed at a certain orientation and would require the wires to perform a 90 degree turn from the FPGA to approach the diplexers. However, harnessing these connectors may still prove to be a challenge; there is flexibility in the orientation and placement of the diplexers, which may assist here if problems manifest. The other set of diplexers are located on the +X cutout panel; the connections from the FPGA daughter board are routed using the space between the main payload volume and volume accessible by the cutout panel. Space is limited in this area as well and wire routing to these patch antennae may be modified in the future.

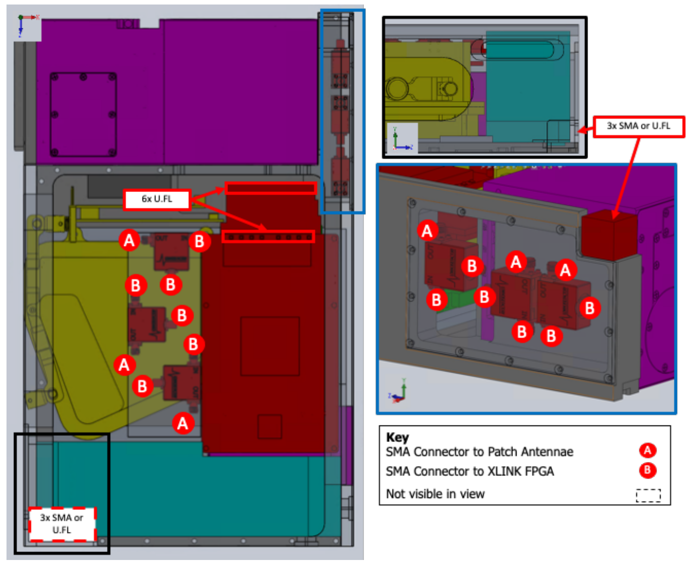

Fig. 21 XLINK component locations and wiring layout in DSC.

C. OSC Electrical Interfacing

In Fig. 22, it is clear that the available volume in the OSC allows for simplified interfacing between the different payload systems. All interfaces to the AVI board can be routed directly on the -Y panel of the spacecraft. The XLINK interface is the exception since the board is above the AVI board. Interfacing to the patch antennae near the BCT avionics module remains very similar despite the different orientation of the FPGA board and daughter boards with respect to the diplexers the +X cutout near the BCT Avionics Module. The wiring for the patch antennae near the PROP system does slightly vary due to diplexers being stacked on the FPGA board as shown in Fig. 23. From there, the cables to the patch antennae are route along the +Y panel to the PROP system. From there, they would need to be route along the Y-axis and below the PROP system just like with the DSC. Because the detector is not present, the number of 90 degree turns required for the wiring from the diplexers to the patch antennae is reduced.

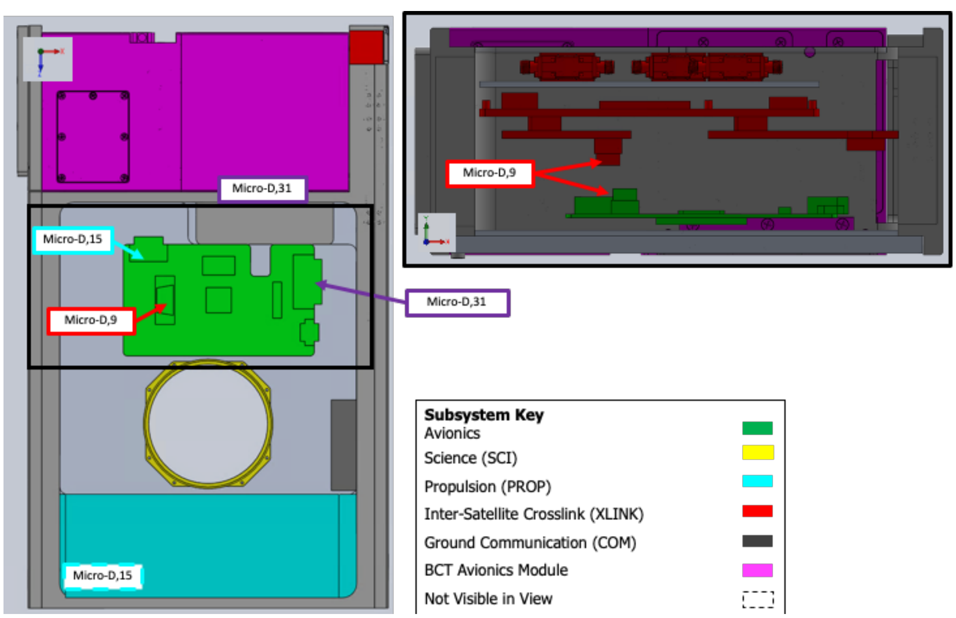

Fig. 22 OSC key wire harnesses.

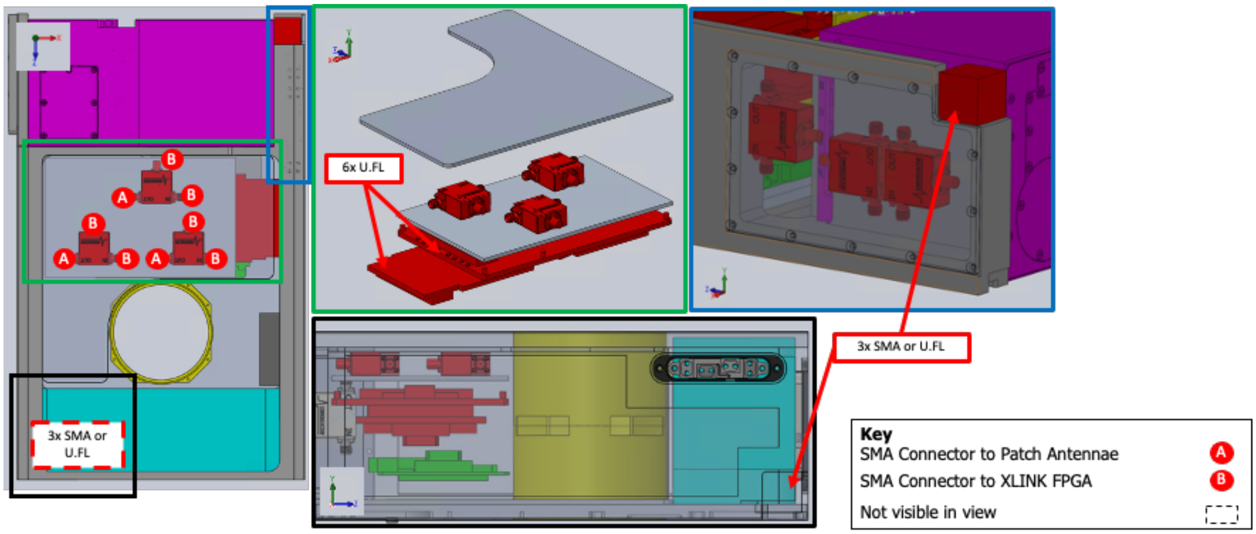

Fig. 23 XLINK component locations and wiring layout in OSC.

VI. Mechanical Interfacing & Spacecraft Integration

A. General Methodology

Following the pattern, the mechanical interfacing of components in the spacecraft will serve as a challenge for the integration team due to the complex wire routing and component placement. Accessibility into the bus for interfacing is challenging since the internals can only be accessed using the +Y and -Y panels. This may not be an issue initially but once the payload volume is fully populated, it may become difficult to reach interfaces. In order to assist integration, a mounting plate system has been developed to allow integration of components to occur in phases and outside of the confines of the spacecraft bus. The intention of the system is to allow mechanical interfacing of major components onto mounting plates outside the spacecraft. Once done, the plates with the attached components can slide into the spacecraft bus. Furthermore, this methodology reduces the mechanical interfacing required directly into the bus, simplifying the bus modification needed by the bus manufacturer. Not all components can be mounted on plates because of the geometries and locations of components. The specifics are unique to each spacecraft.

B. DSC Mechanical Interfacing

The DSC poses the greatest challenge for mechanical interfacing just like with electrical interfacing. The bottom plate serves as the primary mounting point for the detector, CSIE electronics, AVI board, and the second star tracker. Use of a mounting plate for these components allows integration engineers to fasten and wire these components without the constraints of the bus. Once done, it can be integrated as one big piece onto the +Y panel of the spacecraft. Mounting plates are limited to the dimensions of the Y panels; this poses a challenge for the PROP system as it is wider than the Y panel, extruding into the spacecraft chassis. Instead of using a mounting plate, PROP, the XLINK antennae, and the UHF radio are mounted directly into the spacecraft chassis. Because of this, these components would need to be interfaced with the spacecraft first. One advantage is the location of PROP and XLINK antennae are the same with both spacecraft, so bus modifications for mechanical interfacing will remain the same. The XLINK FPGA and daughter boards are mounted directly to the +Y panel. Ideally, a mounting plate would be used in place of direct mounting, but there is little clearance between the top of the second star tracker and the XLINK boards for an additional mounting plate. The diplexers serving the antennae under the PROP system use a mounting plate that mounts directly to the +Y panel aside the FPGA. One advantage is because the +Y panel is removable wiring of XLINK components is more feasible than if the wiring would need to be done more internally in the system. Finally, the +X cutout panel serves to fasten the diplexers that interface with the patch antennae near the BCT avionics module. Fig. 24 visualizes what components are mounted to what features of the spacecraft bus while Fig. 25 is an exploded view of the components interfaced onto the DSC bottom plate.

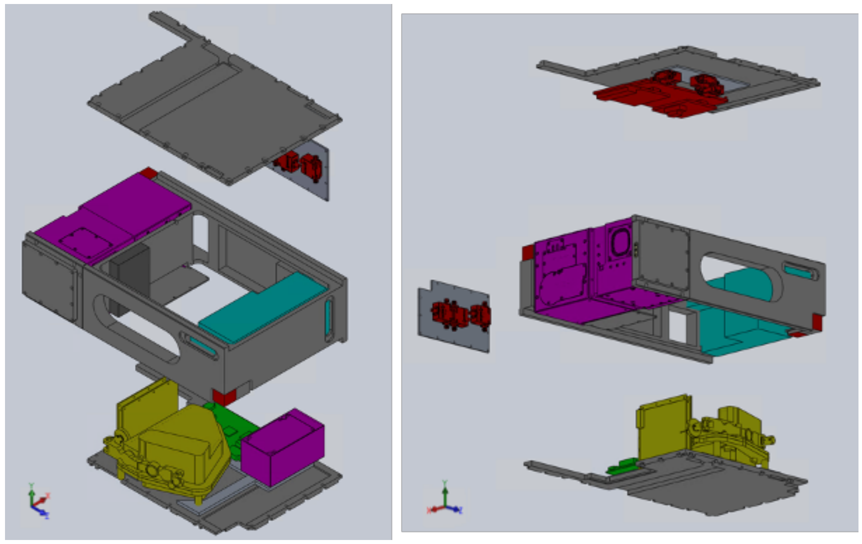

Fig. 24 Exploded views of the DSC showing primary bus locations used for mechanical interfacing.

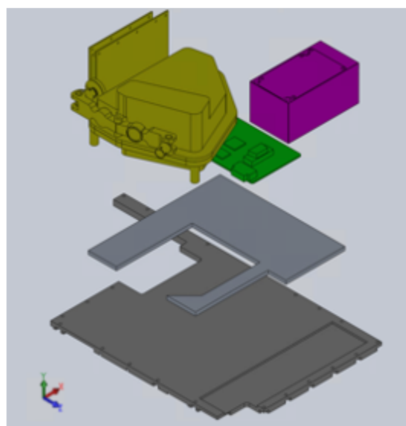

Fig. 25 Exploded view of the DSC bottom plate interfacing.

C. OSC Mechanical Interfacing

Following the same methodology as with the DSC, the OSC requires a simpler solution for mechanical interfacing and integration. The AVI board and photon sieve are proposed to interface with the bottom mounting plate as shown in Fig. 26. While the XLINK and set of diplexers are proposed to interface with the top mounting plate as shown in Fig. 26. Both mounting plates would serve as locations for mechanically interfacing wires as well. As mentioned with the DSC, the XLINK antennae, PROP system, and UHF antenna are mounted directly to the bus chassis. Furthermore, the +X cutout panel again serves as the mechanical interfacing location for the diplexers interfacing with the patch antennae adjacent to the BCT avionics. Fig. 27 offers views on how components are integrated into the spacecraft bus.

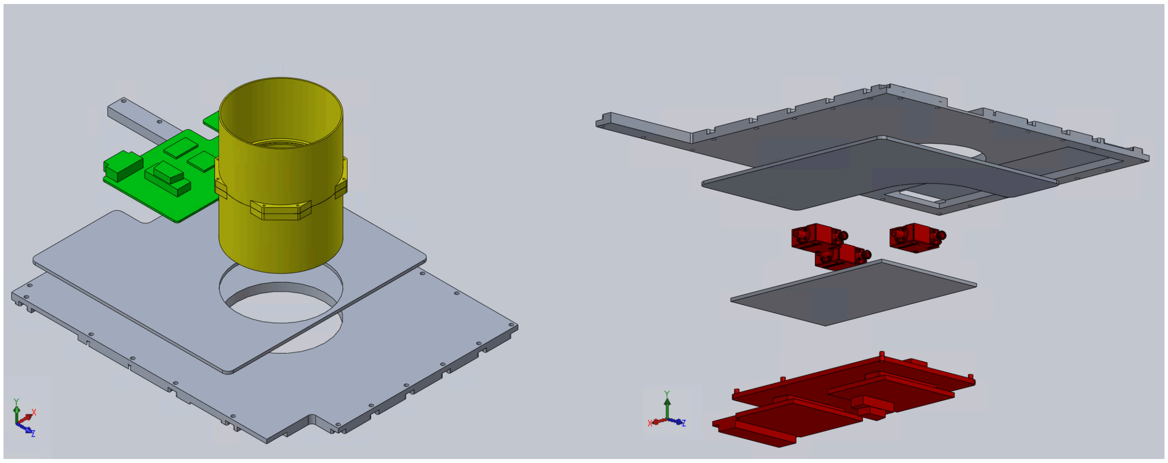

Fig. 26 Mechanical interfacing of components to bottom and top mounting plates in OSC.

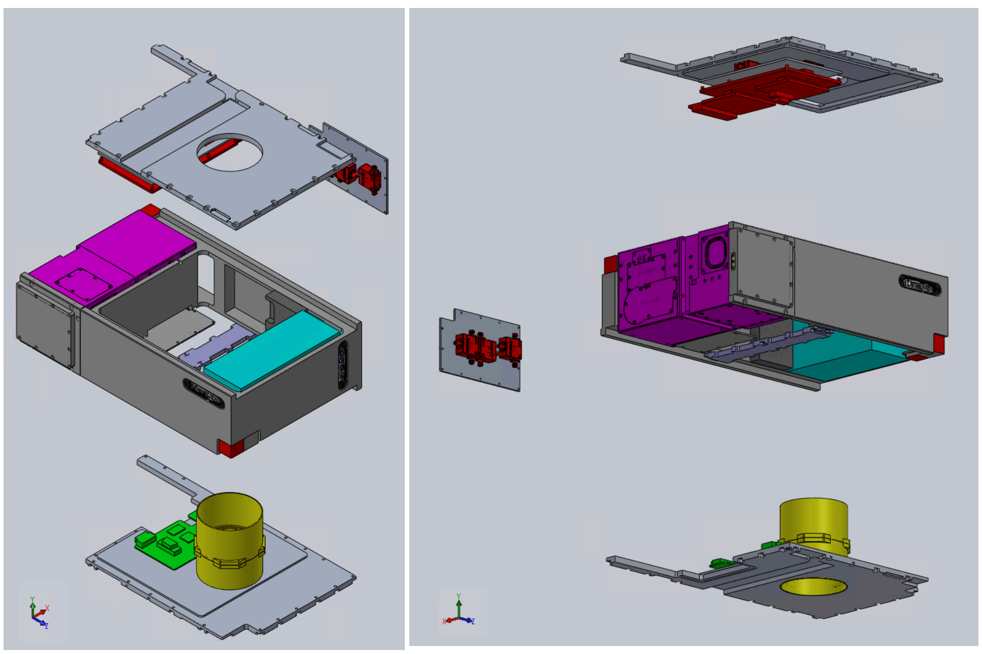

Fig. 27. Exploded views showcasing the key bus features used for mechanical interfacing.

VII. Design Considerations and Next Steps

What is shown here is a good start for payload system design; however, there are necessary steps to take as the component design matures to ensure mission success.

First, the mission team must work in conjunction with BCT to ensure the placement of components meet the bus's specifications and requirements. Communication and negotiation must begin between the bus partner and the mission team to ensure all requirements are met both for the bus and for the payload components. Currently, assumptions have been made on what material can be cut out of the spacecraft chassis to fit components in without consideration on the impact to structural integrity of the panels and chassis. For example, the Y panels are cut out in the OSC to allow for the photon sieve to function and focus EUV radiation; these holes may impact the structural integrity of the panels. As mentioned earlier, the location of the UHF antenna is yet to be determined; the team must work with BCT to finalize the location which has implications on the placement of payload systems. Furthermore, BCT has confirmed that sun sensor locations can be changed from what is shown currently in the CAD model. They currently are known to interfere with the propulsion system; it is critical their locations are finalized so no interference occurs, or the PROP system is modified accordingly. Finally, components are tightly spaced in the DSC; it is important that as the design progresses that interference issues are deconflicted.

As the component design matures, wire routing and specifics of electrical interfacing must be determined. There is currently little knowledge on the harness and wiring for BCT provided components in the payload volume; an understanding must be gathered on these components, and then, a plan must be determined on how to electrically interface these components. In general, the wire harnesses, wires, and electrical components that make up the payload must be verified that they are space rated. In addition, the thermal requirements of components and the radiation tolerancing of components must be considered as the matures. For instance, the XLINK FPGA is not space rated and FPGAs are known to have problems with single event effects; the team must analyze and consider mitigation strategies to ensure the FPGA as well as other components can handle the space environment. The team must also consider the turning radii of wires and how to mechanically interface with the wires.

With regard to mechanical interfacing and mounting, the design of the mounting plates and considerations on how components are interfaced into the bus should be considered more in depth. The next steps are to consider the specific fasteners and locations of fastener points on components as well as the spacecraft. Proper interface control documents must be developed as the designs of components mature. Furthermore, a formal integration plan must be considered to ensure integration runs smoothly. The design of mounting plates must also consider the loads that the spacecraft may face during launch; the designs shown here are notional to demonstrate the general methodology behind mechanical interfacing and integration. Mounting plates may also have to be modified to take on loads that normally the spacecraft bus or panels take. They may also need to be modified to stay within the allowable mass margins for 6U CubeSats.

VIII. Conclusion

Achieving unprecedented high angular resolution observations of the solar corona require several key technologies to work in conjunction. The consolidation of these key technologies into the 6U form factor poses challenges with strict requirements. The VISORS team utilized a mechanical and system design technique, considering the placement requirements of components, the interfacing of components, and the implications of payload design on spacecraft integration. The completion of the preliminary payload mechanical and system design demonstrates that the VISORS mission can incorporate the necessary technologies to accomplish its unprecedented mission and meet its strict requirements. With the completion of the mission's Preliminary Design Review (PDR) in the fourth quarter of 2020, the team will continue to mature components and payload design. As discussed in this paper, the VISORS team aims to develop, test, and integrate innovative technologies in the areas of propulsion, close-range communications, formation flying, relative navigation, and solar imaging. The developments from this mission will aid the creation of the next generation space missions that will utilize the capabilities and performance demonstrated by the VISORS satellite formation.

IX.Acknowledgments

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Table 2: VISORS mission team.

Participant	Institution
Farzad Kamalabadi (PI), Endowed Professor	**Univ. of Illinois**, Electrical and Computer Eng.
Alina Alexeenko, Professor	**Purdue Univ.**, Aeronautics & Astronautics
Philip Chamberlin, Research Scientist	**Univ. of Colorado**, LASP
Simone D'Amico, Assistant Professor	**Stanford Univ.**, Aeronautics & Astronautics
Adrian Daw, Research Astrophysicist	**NASA GSFC**, Heliophysics Science Division
Kevin Denis, Electrical Engineer	**NASA GSFC**, Instrument Systems and Tech. Division
Eylem Ekici, Professor	**Ohio State Univ.**, Electrical & Computer Eng.
Subhanshu Gupta, Assistant Professor	**Washington State Univ.**, Electrical Eng.
John Hwang, Assistant Professor	**Univ. of Calif. San Diego**, Mechanical & Aerosp. Eng.
James Klimchuk, Research Astrophysicist	**NASA GSFC**, Heliophysics Science Division
Glenn Lightsey, Professor	**Georgia Tech**, Aerospace Eng.
Hyeongjun Park, Assistant Professor	**New Mexico State Univ.**, Mechanical & Aerosp. Eng.
Douglas Rabin, Research Astrophysicist	**NASA GSFC**, Heliophysics Science Division
John Sample, Assistant Professor	**Montana State Univ.**, Physics
Thomas Woods, Associate Director	**Univ. of Colorado**, LASP

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