

# **Design and Systems Engineering of AFRL's Demonstration and Science Experiments**

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## **ABSTRACT**

The Air Force Research Laboratory (AFRL) Space Vehicles Directorate has developed the Demonstration and Science Experiments (DSX) mission to research technologies needed to significantly advance Department of Defense (DoD) capability to operate spacecraft in the harsh radiation environment of medium-earth orbits (MEO). The ability to operate effectively in the MEO environment significantly increases the DoD's capability to field space systems that provide persistent global targeting-grade space surveillance, high-speed satellite-based communication, lower-cost GPS navigation, and protection from space weather on a responsive satellite platform. The three DSX experiments areas are:

1. Wave Particle Interaction Experiment (WPIx): Researching the physics of very-low-frequency (VLF) transmissions in the magnetosphere and characterizing the feasibility of natural and man-made VLF waves to reduce space radiation;
2. Space Weather Experiment (SWx): Characterizing and modeling the space radiation environment in MEO, an orbital regime attractive for future DoD and commercial missions;
3. Space Environmental Effects (SFx): Researching and characterizing the space weather effects on spacecraft electronics and materials.

DSX uses a modular design that allows for launch either as a primary satellite on a conventional launcher, such as a Minotaur, or as a secondary payload on a larger rocket, such as the Evolved Expendable Launch Vehicle (EELV).

An overview of the DSX spacecraft design, requirements, systems engineering approach, bus subsystems, payload designs, and experiments will be described.

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\*Contractors

## 1.0 Introduction

The Demonstration and Science Experiments (DSX), is AFRL's fourth space science technology experiment. It was originally conceived by AFRL researchers in 2003 to conduct physics based experiments, and was selected as an AFRL mission in 2004. With primary funding from AFRL, DSX enjoys additional support from DARPA and NASA, and is comprised of elements provided by AFRL, NASA, academia, and many contractors.

The DSX spaceflight experiment comprises three major research areas, which together will pave the way for new DoD capabilities in space surveillance, microsatellites with significant operational capabilities, and protection of space assets from natural and enhanced radiation environments. This mission will advance the warfighter's capabilities in communication, surveillance, and navigation. The DSX experiments include research in three major experiment categories.

The physics of Very Low Frequency (VLF, 3-50 kHz) electromagnetic wave injection from space and ground-based transmitters, propagation, and wave-particle interactions in the magnetosphere comprises the first category, *Wave Particle Interaction Experiments (WPIx)*. Equipment on DSX will transmit and receive VLF waves and quantify their effect on the trapped electron populations in the magnetosphere. Detailed measurement and mapping of high and low energy particle and plasma distributions; radiation dose rates, local magnetic fields and pitch angle distributions in the poorly characterized MEO environment and slot region make up the DSX *Space Weather Experiments (SWx)*. The third major experiment category, *Space Environmental Effects (SFx)*, involves characterization of the effects of space weather and the space environment on materials and electronics. SFx consists of NASA's Space Environment Testbed, as well as AFRL developed photometers and radiometers.

Coupling these experiments into a single platform provides a lower-cost opportunity for AFRL due to their common requirements and goals. All three experiments need a 3-axis stabilized spacecraft bus, a suite of radiation sensors, and extended duration in a MEO orbit.

## 2.0 Flight Experiment Concept

The functional baseline configuration for the DSX flight experiment is shown in Figure 1. The core is the Evolved Expanded Launch Vehicle (EELV) Secondary Payload Adapter (ESPA) Ring [1,2], which is used to maximize launch opportunities on both Space Test Program (STP) and operational DoD launchers. ESPA was developed and designed by CSA Engineering. Every EELV launch is a potential ride for an ESPA ring, and thus also for DSX. The ESPA ring comprises the primary structure for DSX, and is upgraded to provide host spacecraft functions (avionics, TT&C, ADCS, C&DH, and power management and distribution) by the addition of components packaged on an *avionics module (AM)*. The DSX payloads (including deployable booms) are mounted on an identical structure, the *payload module (PM)*, attached to the ESPA ring opposite the avionics module. The AM and PM together comprise the DSX Host Spacecraft Bus (HSB). The entire assembly is designed to be stowed within a 4-m diameter EELV fairing.

The design intent for the ESPA was for it to remain attached to the launch vehicle upper stage to facilitate deployment of up to six microsatellites. However, unlike the traditional ESPA approach, the DSX avionics and payload modules do not separate; they remain attached. After the primary satellite is deployed, the DSX ESPA separates from the EELV 2nd stage booster to become a free-flyer spacecraft. The ESPA ring serves as the DSX spacecraft's primary structure, with an upper separation interface to the primary payload adapter and a lower separation interface to the EELV upper stage adapter. Figure 2 shows DSX and identifies its major components in a stowed configuration. The experiment is baselined for flight in the radiation belts with a nominal orbit of 6,000 km x 12,000 km elliptical, mid-inclination, and 1-year of mission operations.

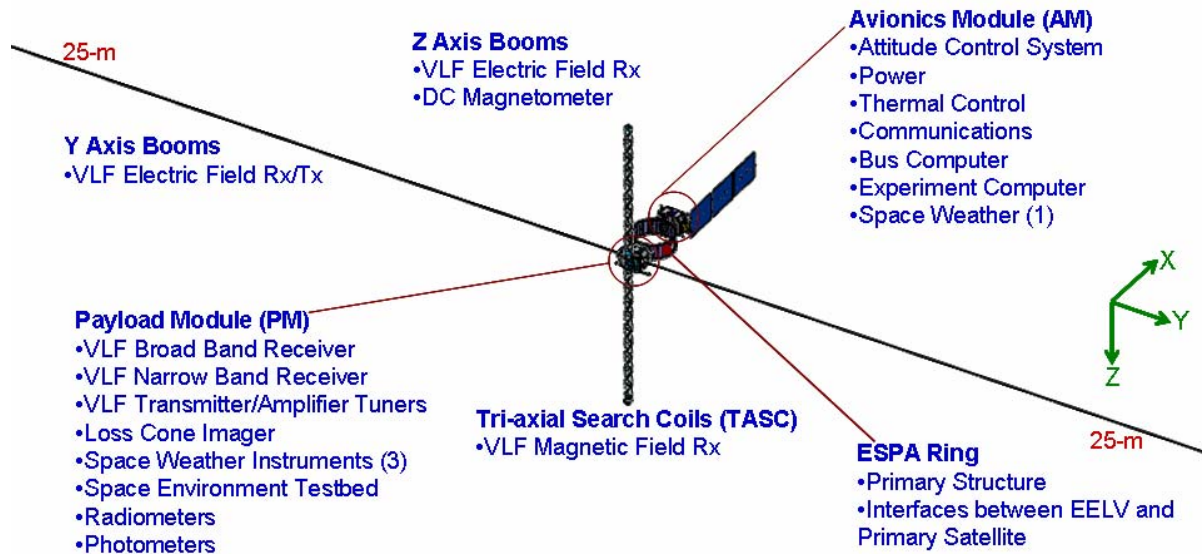


Figure 1. DSX Flight Experiment Configuration (deployed)

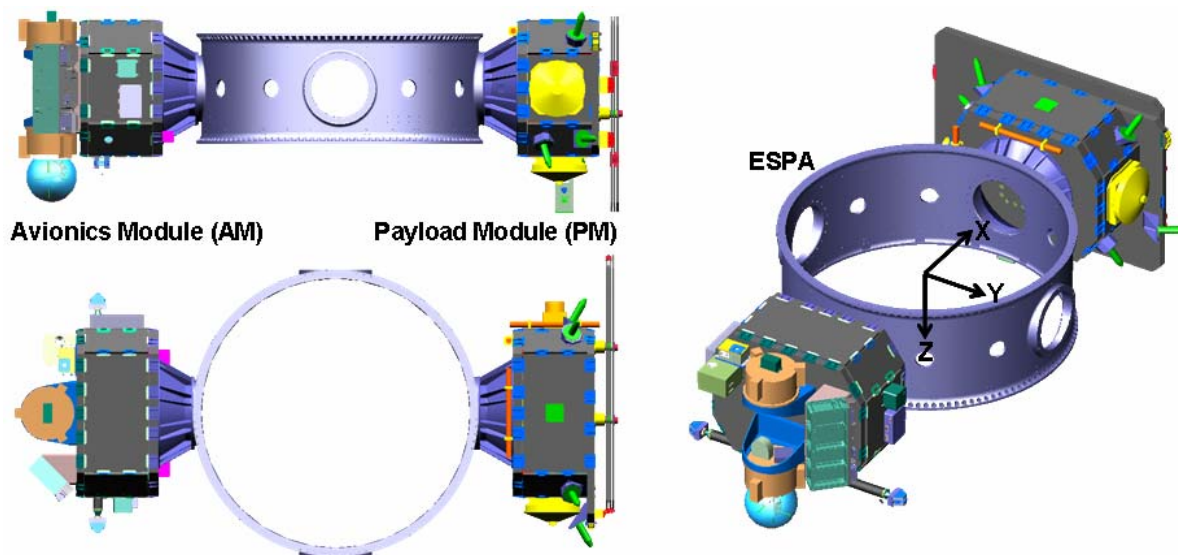
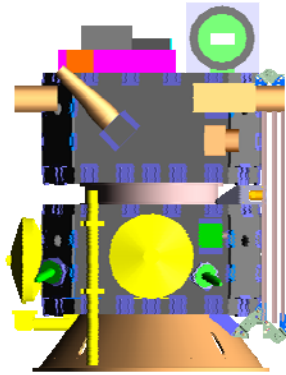
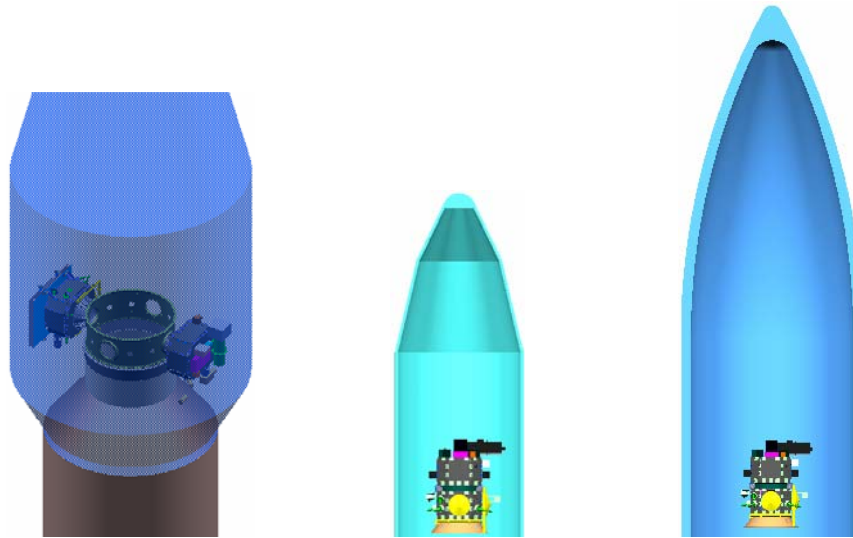


Figure 2. DSX Baseline Design, Stowed (Separation Systems not Shown)

From the earliest DSX conceptual planning, it was understood that the key to a successful program execution was maximizing DSX's compatibility with numerous launch vehicles. While the ESPA-based approach makes every EELV a potential ride, not all co-manifest opportunities possess sufficient excess launch vehicle performance needed to inject DSX into its uncommon MEO orbit. Therefore, flexibility in the design will be maintained for as long as possible to allow repackaging DSX for flight on dedicated launchers. Inherent in the modular approach shown in Figure 2 is the ability to separate the two modules from the ESPA ring, and directly stack them in a vertical assembly for a dedicated launcher. The stacked HSB configuration is shown in Figure 3, and fairing encapsulated views of DSX are shown for EELV, Minotaur IV/V, and Falcon V in Figure 4. This modular architecture not only makes DSX reconfigurable for various launch vehicles, but it also greatly simplifies payload integration.



**Figure 3. DSX Stacked Configuration for Dedicated Launchers**

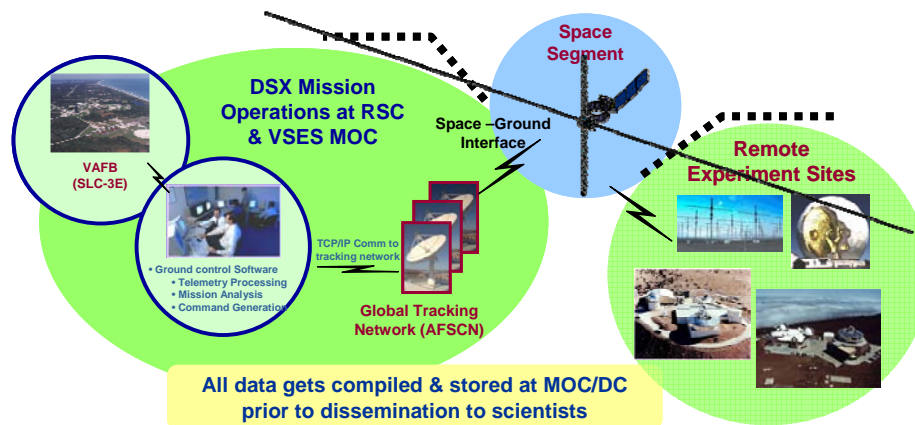


**Figure 4. DSX Stowed, Encapsulated**

Left to right: Atlas V EELV (baseline ESPAsat), Minotaur IV/V, and Falcon V

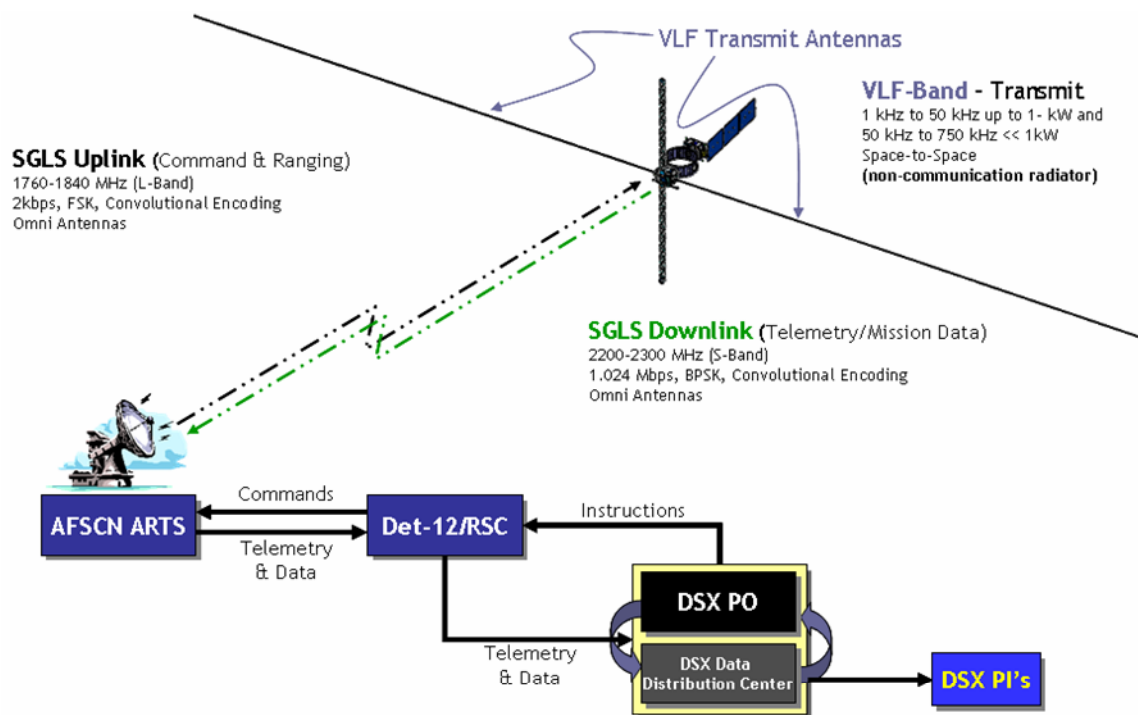
## 2.1. Ground Segment

The DSX ground segment consists of three major elements: mission control, global telemetry tracking & control (TT&C) network, and the remote experiment sites, as shown conceptually in Figure 5.



**Figure 5. DSX Ground Segment Overview**

The mission will be operated out of both the Space and Missile Systems Center's Research, Development, Test and Evaluation Support Complex (Det-12/RSC) and AFRL's Missions Operations and Data Center (MOC/DC) facilities located at Kirtland AFB, NM. Det-12/RSC will provide the real-time contacts support, schedule the Air Force Satellite Control Network (AFSCN) TT&C, and acquire both spacecraft and payload data from the space vehicle. Commands will initially be developed from AFRL's MOC/DC facility where the effects will be simulated to validate proper operation. These commands will be sent to the RSC for uplink via Space-Ground Link System (SGLS) communication protocol to the DSX satellite. Downlinked information from the space vehicle state-of-health will be validated from the real-time operations system located at the RSC. Data from remote experiment sites i.e., optical and radar ranges and VLF ground installations, are coordinated at the data center. All uplinked commands and downlinked telemetry and science data for DSX will be encrypted. In addition to SGLS communications bands, the DSX spectrum allocation application also includes the VLF frequencies used for the wave particle interaction experiment. The RF emissions flow for DSX is shown in Figure 6.



**Figure 6. DSX RF Emissions Flow Diagram**

All DSX data products, including real-time telemetry, stored state-of-health, stored and real-time experiment data, and ranging information, will be sent to the MOC/DC for further analysis and data archival. The DC will correct time tags, parse data files, collect data from remote sites (optical and radar ranges and VLF ground installations), and permanently archive the data. Experiment data and required spacecraft by-products will be distributed to each experiment team. DSX has applied for the necessary spectrum allocations.

Preliminary orbit analysis performed for DSX has shown that AFSCN provides fairly continuous coverage over the orbit. Based on the need for AFSCN to prioritize operational assets, an average of 85 minutes contact time per orbit, for a 5.3 hour orbit period (6,000 km x, 12,000 km) is assumed. Figure 7 shows the DSX orbit, along with the AFSCN, optical, and radar ground stations along its track.





of VLF from a space-based platform and the current levels of natural and man-made VLF waves in the inner magnetosphere. Models of ground-based VLF injection and global magnetospheric propagation will also be validated in a controlled manner. An electron loss cone detector on DSX will provide the capability to directly correlate changes in energetic particle distributions with injected wave power. Accurate models of the VLF injection, propagation, and wave particle interaction processes are critical to assess the viability of an operational space-based VLF transmission system. Measurements of the VLF wave field will also accelerate the development of the natural radiation belt forecast models needed for space situational awareness and mission planning. The WPIx payloads consist of four major components, further described below.

### **3.1.1 Wave-Induced Precipitation of Electron Radiation (WIPER) Broad Band Receiver**

The WIPER broadband receiver (BBR) is a passive receiver designed to measure the natural and man-made ELF, VLF and LF signals in the inner magnetosphere in the range 3-50 kHz. The BBR is electrically connected to all four deployed booms shown in Figure 1, with two channels of the BBR measuring electric fields via the y and z-axis VLF antennas, and three more channels measuring magnetic field about three axes of the Triaxial Search Coil Antennas (TASC). TASC is mounted on the tip of the +Z boom and the BBR is contained within the payload module. BBR measurements are also coordinated with high power transmissions resulting from the WIPER transmitter, transmissions from ground based facilities, as well as naturally occurring sources of magnetospheric VLF (i.e., lightning). The WIPER BBR is being developed by Stanford University.

### **3.1.2 WIPER Transmitter and Narrow Band Receiver Electronics**

The WIPER transmitter electronics portion consists of the transmitter controller unit (TCU) with an integral narrow-band receiver (NBR) as well as two transmitter amplifier tuner units (TATU). Each TATU electrically connects to a pole of a center-fed dipole antenna supported by the  $\pm Y$  booms shown in Figure 1. The transmitter electronics include dynamic tuning circuits, embedded within the TCU. The VLF transmission system will broadcast up to the kilowatt level in the range 1-50 kHz and at much smaller power levels from 50 kHz - 750 kHz by charging conductive antenna elements that run along the Y-axis booms to levels up to 10 kV. The TCU/NBR and both TATUs are housed within the payload module. The WIPER transmitter electronics are being developed by the University of Massachusetts-Lowell, benefiting from significant heritage resulting for a similar payload flown on NASA's IMAGE.

### **3.1.3 Loss Cone Imager (LCI)**

The Loss Cone Imager (LCI) is an electron loss-cone particle detector that will provide a measurement of three dimensional energetic particle distributions with emphasis on the measurement of the fluxes of energetic electrons along the direction parallel and anti-parallel to the local geomagnetic field vector. The LCI consists of two scanning heads, with  $\pm 180^\circ$  motorized articulation capability, and each projected a  $180^\circ$  field of view fan, with each scan head position, so that together the full  $4\pi$  unit sphere may be imaged. LCI depends on data from the DC Vector Magnetometer (VMAG) for its motor pointing control loop, and for later ground data correlation.

A separate Solid State Detector (SSD) telescope called the High Sensitivity Telescope (HST) will be mounted on the DSX in order to obtain fluxes of energetic electrons along the geomagnetic field vector direction. This telescope is designed to have a geometric factor of  $0.1 \text{ cm}^2 \text{ steradian}$  with sufficient shielding to permit the detection of  $100 \text{ particles/cm}^2\text{-sec-steradian}$  in the loss cone. The LCI is under development by Boston University, and also benefits from heritage obtained in the development and flight of similar instruments such as the Imaging Electron Spectrometer (IES) on Cluster.

### 3.1.4 DC Vector Magnetometer (VMAG)

A DC fluxgate vector magnetometer (VMAG) will be used to determine the direction of the magnetic field to better than one degree at all points in the orbit. This performance will allow the mapping of locally measured particle distributions to global distributions. VMAG sensors will measure the DC B-field over a 100-10000 nT range with  $\pm 0.1$  nT accuracy at 20 Hz. VMAG data is also used in real-time by the LCI, as previously discussed. VMAG will operate continuously to provide DC magnetic field and ULF wave environment data required by the SWx MEO space particle modeling experiment, thus it supports both WPIx and SWx experiment objectives.

The VMAG electronics will be mounted within the payload module and the fluxgate sensor will be deployed on the tip of the -Z boom, opposite TASC. VMAG is under development for DSX by UCLA, and is similar to other fluxgate magnetometers (shown in Figure 9) developed by UCLA and flown on several missions.



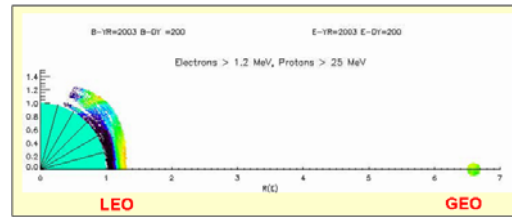
**Figure 9. Fluxgate Sensor (left) and Electronics (right)**

## 3.2. Space Weather Experiments (SWx)

With an orbit between 6,000-12,000 km, DSX will explore a large swath of the inner magnetosphere, in particular the outer region of the inner proton belt the “slot region”, and inner regions of the outer electron belt. This domain has remained largely unexplored due to the understandable tendency of military, commercial, and scientific satellite systems to be located outside the intense regions of radiation, most often at LEO or GEO. However, increasing demands for uninterrupted coverage of the Earth from space are driving planners to consider putting satellite constellations in orbits spending significant time in the inner magnetosphere. Current standard space particle models (e.g. AE-8 and AP-8) can be off by as much as 50 times or more in estimating the time taken to reach hazardous dose levels in the MEO regime. It is imperative that measurements be made of the plasma and energetic particles so that adequate climatological, situational awareness, and forecast models can be developed to enable the successful design and operations of systems in these new and desirable orbit regimes. Accurate environment determination is important for DSX so that quantitative correlation with the performance of the spacecraft and its payloads over the course of the mission may be performed.

Deficiencies of current standard radiation belt models in the inner magnetosphere include the lack of (a) spectrally resolved, uncontaminated measurements of high energy protons (10-400 MeV) and electrons (1-30 MeV) and (b) accurate mid to low energy ( $< 1000$  keV) measurements of the energetic particle and plasma environment (see Figure 10). The Space Weather instruments on DSX will address these deficiencies. Included will be electron and proton detectors measuring both the spectral content and angle of arrival of both species over broad energy ranges. An on-board magnetometer (VMAG, see above) will allow for the transformation of angle-of-arrival measurements into estimates of the flux distribution with respect to the local pitch-angle (i.e. the angle between the particle velocity and magnetic field). Local pitch-angle distributions can then be used to estimate global particle distributions by mapping techniques using the well-known equations of motion and magnetic field models tagged to the local measurements. The nominal space weather sensor suite capabilities are described in detail in the following sub-sections.





**Figure 10. Space Weather Data from TSX and DSP Shows Lack of Data in MEO**

### 3.2.1 Compact Environmental Anomaly Sensor (CEASE)

Composed of two dosimeters, two particle telescopes and a Single Event Effect detector, CEASE has the capability to monitor a broad range of space hazards from surface damage and charging (keV electrons) to Single Event Effects resulting from  $>100$  MeV cosmic rays and solar protons [5,6]. The angular field-of-view for CEASE is relatively large and will not allow for pitch angle resolved measurements. CEASE will be mounted on an exterior panel of the payload module. One change for DSX is that CEASE will capture and downlink the full dose spectra from each dosimeter, whereas prior versions only captured six reduced data points (two for low LET data and four for high LET data). The CEASE instrument was developed by Amptek, Inc. and has already flown on several spacecraft. CEASE and the other SWx instruments are shown in Figure 11.

### 3.2.2 Low Energy Electrostatic Analyzer (LEESA)

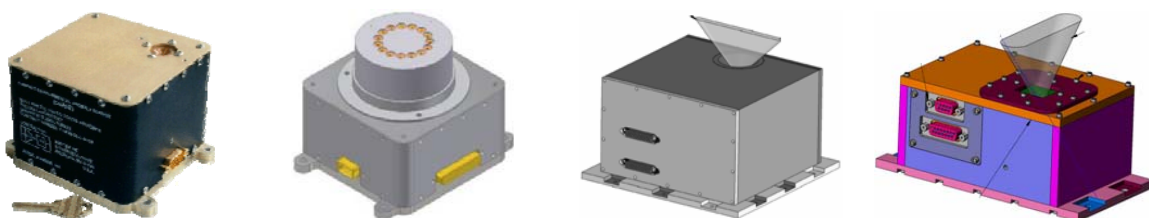
LEESA will measure energy fluxes and energy spectra for low energy electrons and protons (100 eV to 50 keV). These low energy particles are responsible for surface electric charging and damage to thin films such as thin-film photovoltaics, conventional solar cell cover glasses, and coatings. LEESA will be mounted on an exterior panel of the avionics module. It is under development for DSX by Amptek.

### 3.2.3 High Energy Imaging Particle Spectrometer (HIPS)

HIPS will measure electrons with energies between 1-10 MeV and protons with energies between 30 and 300 MeV. These high energy particles are responsible for microelectronics damage, displacement and total dose damage, SEEs, and deep dielectric charging. HIPS will be mounted on an exterior panel of the payload module. It is under development for DSX by Physical Sciences, Inc (PSI).

### 3.2.4 Low Energy Imaging Particle Spectrometer (LIPS)

LIPS is designed to measure the ring current particles that are important in the energy flow processes in the magnetosphere. This particle population plays an important role in spacecraft charging and surface damage due to sputtering. The instrument, built by Physical Sciences Inc, uses specially designed combinations of scintillator materials to detect electrons and protons with energies between 20 keV and 1 MeV. Eight 10 degree by 8 degree field-of-view windows will provide pitch angle resolution. LIPS will be mounted on an exterior panel of the payload module, and is also being developed by PSI.



**Figure 11. Space Weather Instruments (CEASE, LEESA, HIPS, and LIPS)**

### 3.3. Space Environmental Effects Experiment (SFx)

#### 3.3.1 Space Environment Testbed (SET)

The SET carrier, under development by NASA GSFC provides standard mechanical, electrical, & thermal interfaces for a collection of small flight investigations. The general areas of investigation of the instruments carried on SET fall in the following categories:

1. Characterization of the space environment;
2. Performance improvement for microelectronics used in space;
3. Accommodation and/or Mitigation of Space Environment Effects for detectors & sensors
4. Accommodation and/or mitigation of charging/discharging effects on spacecraft & spacecraft components;
5. Definition of mechanisms for materials' degradation and performance characterization of materials designed for shielding from ionizing radiation.

The first SET mission, SET-1, was recently added to the DSX experiment suite. SET-1 consists of the following specific investigations (also refer to Figure 12):

**Credance.** Cosmic Ray Environment Dosimetry and Charging (Credance) measures the radiation environment (total dose), and the charging environment for normal background and during space weather events. Credance specific measurements include:

1. Proton flux > 40 MeV per unit solid angle
2. Charge deposition in large silicon diodes arranged in telescopes. Pulse height analysis is used to obtain ion linear energy transfer (LET) spectra of heavy ions in the 100 MeV cm<sup>2</sup>/g to 25000 MeV cm<sup>2</sup>/g range
3. Threshold voltage shift as a function of time to measure total ionizing dose in silicon at 2 different shielding depths
4. Charging current at 3 different shielding depths which provides energetic electron flux measurements at 3 energies

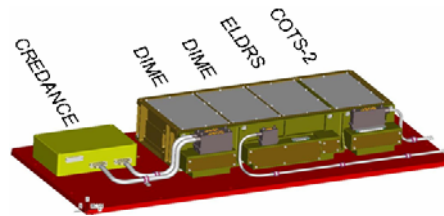
Its measurements overlap in part with the CEASE space weather instrument, and will provide for mutual validation and data correlation of both instruments. Credance has been developed by QinetiQ/UK.

**DIME.** Dosimetry Intercomparison and Miniaturization (DIME) will measure space radiation environments which are detrimental to space system reliability using novel dosimetry techniques. DIME occupies two 3U card slots on the SET-1 carrier, and is being developed by Clemson University. Specific DIME measurements include:

1. Radiation-Sensing Field-Effect Transistor (RADFET) - threshold voltage shift as a function of time; converted to total ionizing dose.
2. Erasable Programmable Read-only Memory (EPROM) - threshold voltage shift as a function of time; converted to total ionizing dose. Also number of single event upsets as a function of time to measure rates as a function of radiation level.
3. Static Random Access Memory (SRAM) - hold devices at different voltages and measure single event upsets. Change in voltage at which an error occurs is converted to dose. Also number of single event upsets as a function of time to measure rates as a function of radiation level.
4. Linear Energy Transfer (LET) Spectrometer - pulse height spectra as function of time; converted to LET to measure ions including protons.
5. Optically Stimulated Luminescent (OSL) Films - visible emission spectrum as a function of time to measure micro-dose.

**ELDRS.** Focused on characterization of Proton Effects and Enhanced Low Dose Rate Sensitivity (ELDRS) in Bipolar Junction Transistors (BJT), ELDRS measures space radiation induced total ionizing dose and displacement damage on linear devices sensitive to enhanced space low dose rate effects. ELDRS will measure base and collector currents for 24 COTS BJTs as functions of the emitter and gate voltages and time. ELDRS is being developed by Arizona State University.

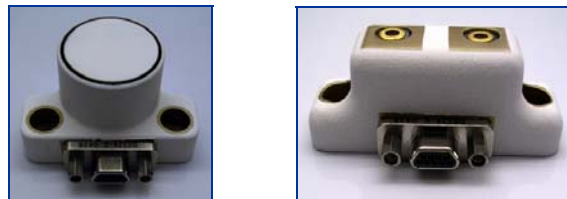
**COTS-2.** COTS-2 measures particle induced single event effects on a single event effects mitigation platform in normal backgrounds and during solar storm events. COTS-2 will measure a single event effect (SEE) on COTS FPGAs, classify the event by event type, and determine if mitigation of the effect occurred without watchdog intervention. COTS-2 is being co-developed by NASA and CNES.



**Figure 12. SET-1 (red plate is none-flight)**

### 3.3.2 Photometers and Radiometers

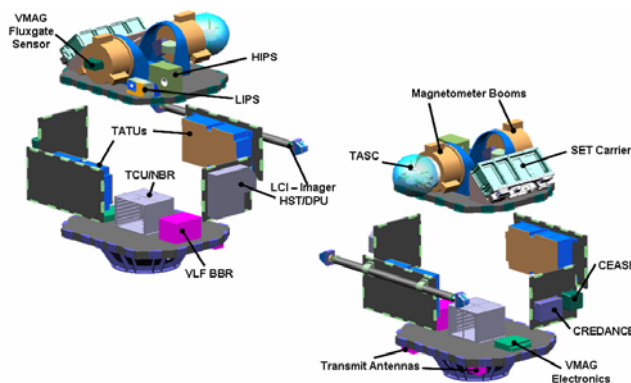
DSX will fly several photometers and radiometers (see Figure 13), developed by AFRL's Propulsion Directorate (Edwards, AFB). Together these will directly measure changes in optical transmission, thermal absorption and emission due to the MEO radiation environment. Coupon level testing of specific developmental coatings intended for thin-film photovoltaics may also be performed using these sensors.



**Figure 13. Radiometers (left) and Photometers (right)**

### 3.3.3 Mechanical Packaging of Payloads

The design for both the avionics and payload modules, and mechanical arrangement and packaging of payloads within the payload module has been performed by MicroSat Systems, Inc (MSI). The resulting payload module equipment layout is shown below (Figure 14).

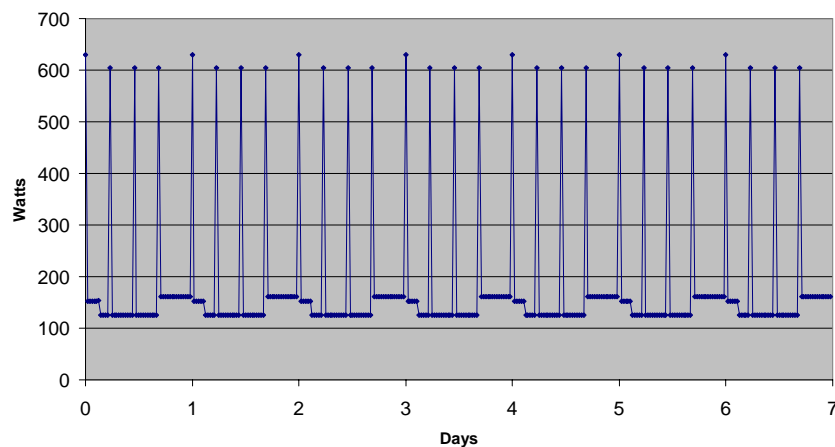


**Figure 14. Mechanical Packaging and Layout of Payloads**

## 4.0 Concept of Operations (CONOPS)

The DSX mission starts with launch as a secondary payload on an EELV. During launch DSX is in a minimally powered state, so that it may detect a separation signal. Following separation of the primary payload from atop DSX, it is likely that the EELV upper-stage would need to be re-started in order to inject DSX into the correct orbit. At this point DSX then separates from the upper stage. Upon detection of the separation signal, DSX will undergo a series of autonomous scripted commands to initialize the bus. After acquisition of signal from the ground, and verification of nominal operation, the bus solar arrays deploy from the avionics module, and the attitude control system is activated in a sun-safe mode in order to orient the spacecraft into a power-positive attitude. Ground controllers will then perform various checks to verify the spacecraft state of health before initiating any boom deployments. Booms will be deployed in pairs: y-axis pair and z-axis pair, to minimize inertial imbalances. During boom deployments the attitude control system is placed in a less responsive mode in consideration of the fact that during deployments the center of pressure and moments of inertia of the spacecraft will vary significantly. Upon completion of these deployments, additional in-orbit tests are performed before the spacecraft assumes an operational mode. The scheduling of the sequence of events from separation through entering the operational phase is being developed, but is expected to be accomplished in under one week.

Once the spacecraft is operational, routine commanding and data collection will be initiated as discussed in Section 2.1 (Ground Segment). Because spacecraft resources (namely power and data volume) are limited, and because certain experiments can only be operated at certain times, experimental payloads are duty cycled to fit within the capabilities of the spacecraft. In general, SWx instruments will operate continuously with only limited ground intervention, such as occasional commanding to adjust threshold and gain settings. They would only be turned off under low power conditions. The SET-1 payload will have two modes of operation, a low power mode where only Credance is powered and collecting data, and the normal mode where all its experiment investigations are operating. While in normal mode measurements take place on a set interval, with some measurements triggered by the on-board orbit propagator, which will indicate to SET-1 the timing of ascending and descending node crossings. The most demanding constraints are associated with the WPix measurements, since these consume the greatest power (during the high power Whistler transmission mode), and also generate the most data, as collected from the broadband receiver. For science reasons, the high power transmissions can be constrained to occur when the spacecraft is within  $\pm 20^\circ$  latitude of equatorial, and because of power limitations the transmission durations will generally be limited to no more than 30 minutes per orbit. Figure 15 is a plot of payload power demand over a period of five days, prepared by Planning Systems, Inc. A more detailed year-long mission timeline is being prepared to plan all DSX operations and coordinate ground resources.



**Figure 15. Payload Power Profile – A Week in the Life of DSX**

## 5.0 Technical Performance Measures (TPM)

Since its earliest stages of development, the DSX program has implemented a mass and power control process, and carefully tracks these metrics. A master equipment list has been developed to identify every equipment or structural element of the spacecraft, with mass and power consumption estimates, contingency estimates, and allocations. Summary forms of these budgets are presented in Tables 1-2, respectively. Note that the end-of-life (EOL) power generation reduction indicated in Table 2 is a result of photovoltaic degradation due to the severe MEO radiation environment.

The upper mass limit requirement for DSX, in the absence of a specific manifested launch vehicle with identified throw weight margins, was simply determined as the mass of a fully loaded ESPA ring, or 1205.6 kg. However, DSX's true mass margins are measured against launch vehicle performance. To that end, launch vehicle performance studies were conducted for EELV ESPA as well as dedicated launches on Minotaur IV/V and the SpaceX Falcon V (in development). The results of those studies are presented in Table 3.

**Table 1. Mass Budget Summary (EELV ESPA Case)**

Equipment	Basic Mass (kg)	Mass w/Contingency (kg)
<b>Total Vehicle, Separated Mass</b>	<b>466</b>	<b>536</b>
<b>Avionics Module (AM)</b>	<b>148</b>	<b>169</b>
Attitude Control (except IMU)	29	31
Avionics (C&DH and PMAD)	9	10
Electrical Power Subsystem	41	46
Thermal Control System (TCS)	4	5
Telecom	7	7
Experiment Computer (ECS)	7	9
SWx (LEESA)	8	10
Harnesses	9	10
AM Structure	35	40
<b>Payload Module (PM)</b>	<b>139</b>	<b>170</b>
WPIx (WIPER, VMAG, LCI)	41	52
Y and Z booms	16	18
SWx (CEASE, LIPS, HIPS)	12	16
SET-1	13	17
Harnesses	12	16
IMU	1	1
TCS	4	5
PM Structure	40	46
<b>Primary Structure</b>	<b>179</b>	<b>197</b>
ESPA	122	128
Separation Systems	28	32
Harness	13	18
Misc.	16	19
<b>Maximum per ESPA Port</b>		<b>181</b>
<b>Max. ESPA Populated</b>		<b>1206</b>

**Table 2. Power Metrics and Budget Summary**

<b>Array Power Generation (W)*</b>	1320 BOL, 850 EOL
<b>Battery Capacity*</b>	60 Amp-hr
<b>Bus Steady-State Avg. Power (SSAP, W)*</b>	~300
<b>SSAP Available for Payloads (W)*</b>	~350
<b>Composite WPIx Exp. SSAP (W)</b>	115 (normal), 720 (HP transmit mode)
<b>Composite SWx Exp. SSAP (W)</b>	23
<b>Composite SFx Exp. SSAP (W)</b>	55 (all experiments), 26 (Credance only)
<b>Total Payload Heater Power (W)*</b>	100 (cold case)

\*Data provided by MSI

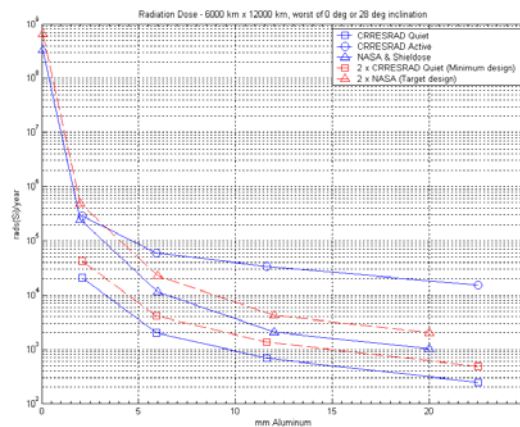
**Table 3. DSX Launch Studies Summary**

Launch Option	S/C Separated Mass (kg)	LV Performance, Mass to Orbit (kg)	Reference Orbit	Mass Margin (kg)	Study Performed by
EELV, Atlas V-401, DMSP Co-manifest	466 (basic) 536 (max.)	865	5,000x11,000 km @117° inclination	329 (min.) 399 (max.)	USAF Det. 12 (STP), Lockheed Martin
Minotaur IV, dedicated	343 (basic) 395 (max.)	500	6,000x6,000 km @28.5° inc.	N/A	RSLP, Orbital
Minotaur V, dedicated	343 (basic) 395 (max.)	313	6,000x21,000 km @28.5° inc.	-82 (min.) - 30 (max.)	RSLP, Orbital
Falcon V, dedicated	343 (basic) 395 (max.)	1000	6,000x12,000 km @9° inc.	605 (min.) 657 (max.)	SpaceX

The current DSX contingency mass estimate of 536 kg allows 670 kg of unallocated mass margin to the above described ESPA limit, but only 329 kg against the launch vehicle performance margin for a specific case of an EELV launch: co-manifest of DSX with a DMSP primary satellite on an Atlas V (401) launch vehicle. The stacked configuration of DSX is significantly lighter at 395 kg with contingency, due primarily to the mass associated with the ESPA ring no longer being carried. However Table 3 shows that that of the dedicated launch options, only Falcon V shows sufficient performance margin. Minotaur V has a negative margin of at least 30 kg; as such a more refined performance analysis may yield sufficient performance if in conjunction a severe mass reduction program on DSX were implemented.

## 6.0 Spacecraft Driving Requirements

The science objectives of DSX impose certain mission specific driving requirements on the design of the spacecraft bus and payloads. These are identified here. The nominal MEO orbit (6000 x 12000 km, mid-inclination), results in a fairly severe radiation environment, because of prolonged residence time in the upper Van Allen belts. The total ionizing dose (TID) for the mission is approximately equivalent to 10 krad/year behind 8 mm of Aluminum absorber shielding. The results of radiation dose analyses performed for DSX are shown in Figure 16. The implication of this is that all DSX equipment must be designed to survive the radiation environment, through the use of shielding, selection of radiation and SEU tolerant or hardened components, or a combination of these approaches.

**Figure 16. Total Ionizing Dose for DSX Mission**

With respect to flight orientation, the attitude control system must provide 3-axis stabilization, maintain a power positive attitude throughout the mission, and orient the spacecraft such that the VLF antennas are aligned normal to the local magnetic field lines, for the duration the spacecraft is within  $\pm 20^\circ$  of equatorial (i.e., when VLF propagation experiments and measurements will be performed). Pointing requirements are fairly relaxed with a  $2^\circ$  control and a  $1^\circ$  knowledge requirement deemed to be sufficient.

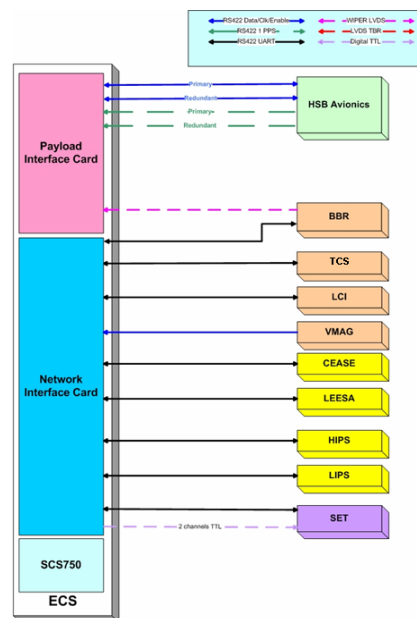


## 7.0 Responsive Enabling Technologies

Historically, spacecraft payloads have been produced via end-to-end development of the entire payload system as well as significant development of key satellite bus components with interfaces to the payload such as the flight computer and power management system. As part of this effort, a considerable fraction of the engineering development required to produce a working system is devoted to specifying, developing, and testing the components designed to perform payload functions that, in general, fall under the category of payload interface electronics for data acquisition and control. This approach is not conducive to rapid response missions because it depends on monolithic integration methods. Monolithic integration of a satellite involves assembly of many components simultaneously. This can present an integration and test nightmare, where debugging problems found in system level testing and routine maintenance of the vehicle can be very complex and costly to perform.

Modular integration on the other hand allows major subsystems and payload modules to be fully integrated and functionally tested separately. System level testing can be performed in a “FlatSat” configuration, pre-integration where the subsystems are wired together using test harnesses to verify connectivity as well as performing closed-loop performance tests. This allows for much faster isolation of problems and shorter integration spans.

DSX maximizes the benefits of modular integration via an architecture that requires that all devices adhere to a standard interface for control and communications (electrical, power, command & communications protocol). This is enabled by the use of dual-redundant network interface cards (NIC) in the Experiment Computer System chassis that provide for command and data interface between the flight control computer and all payloads. Rather than a single on-board computer responsible for handling all spacecraft functions, DSX unburdens the main spacecraft computer with the payload related operations, by providing a dedicated payload interface computer, or Experiment Computer System (ECS), under development by Planning Systems, Inc. for DSX. This architecture decouples the requirements of the flight computer with the design of the payloads, allowing for a completely standardized spacecraft flight computer, almost independent of payload requirements. The highly capable ECS design is also generalized in nature, so that it can handle a wide range of payload types and classes simultaneously, with low data latency. An ECS block diagram is provided in Figure 17., below.



**Figure 17. Experiment Computer System (ECS)**

These modular avionics technologies complement the modular reconfigurable mechanical design, and together represent a path finder for future rapid-response spacecraft. With this approach, the DSX avionics module can be designed, integrated, and delivered before all the payloads have completed their development cycle, or even been selected. The next evolutionary step for this technology involves adding networking capabilities and enhancing the flight control computer so they are pre-programmed to self-configure spacecraft subsystems, components, devices, and payloads. This allows for rapid configuration and integration of off-the-shelf, modular spacecraft consisting of pre-defined building blocks “on the fly”.

## 8.0 Summary

AFRL’s DSX mission is well underway. The Spacecraft Systems Requirements (SRR) was completed in the fall of 2004. Avionics, spacecraft bus, and Experiment Computer System Preliminary Design Reviews (PDR’s) were completed in 2005, and most payloads will have reached the PDR milestone by the first quarter of 2006. With the fully integrated avionics module, depopulated payload module, and ESPA ring scheduled for delivery to AFRL in the first quarter of 2007, and all flight payloads delivered for AI&T by April 2008, DSX is currently on track to be launch ready by late FY08.

The DSX mission will provide an important database from which future designers of space missions intended to operate in the MEO environment can rely. Beyond the inherent value of the scientific experiments, DSX is also path-finding a new, responsive approach for integrating spacecraft. The ESPA platform ensures affordable and frequent opportunities for launch. In addition, the combination of a dedicated payload/experiment computer and modular integration approach, allows a “standard” host spacecraft bus to be used, limiting the development risk to the experiments alone.

## 9.0 Acknowledgements

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Specific content provided by others includes many of the images in this paper. Specifically all stowed and deployed views of the spacecraft (Figures 1, 2, 4, and 14) were provided by MSI, Figures 3, 15, and 17 were provided by Planning Systems. Images of VMAG components (Figure 9) were provided by UCLA. In Figure 11, images of CEASE and LEESA were provided by Amptek, and those of HIPS and LIPS were provided by Physical Sciences. The SET-1 image (Figure 12) was provided by NASA GSFC.

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