THE DESIGN AND DEVELOPMENT OF A SEPARATION SYSTEM FOR A LOW-COST SPHERICAL NANOSATELLITE

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1. ABSTRACT

The Drag and Atmospheric Neutral Density Explorer (DANDE) is a spherical, 50kg satellite designed and built by students at the University of Colorado, Boulder. The goals of the DANDE mission are to fabricate a low-cost proof-of-concept spacecraft that improves our understanding of spatial and temporal variability of atmospheric neutral density and winds in the thermosphere. This project recently (January 2009) won the University Nanosat 5 competition sponsored by the Air Force Research Lab (AFRL) and as a result will be launched in early 2011. To accurately measure atmospheric drag requires a spherically shaped satellite to allow for a known cross sectional area and coefficient of drag creating several unique challenges for the student engineers. These challenges include creating a separation system while attaching the spacecraft to the launch vehicle and maintaining the spherical form factor. A solution has been achieved by creating an adapter plate that uses off-the-shelf mechanisms and a student designed four-point kinematic mount system to connect the 50 kilogram spacecraft to the launch vehicle. The student team has performed a series of analyses and testing over during the design phase of the project to ensure that the system meets the requirements set forth by the AFRL. These include finite element and solid mechanics analysis, vibration tests, microgravity releases and a future planned thermal vacuum test. This paper describes the student effort led by Bruce Davis to design, analyze and test this separation system over the course of two years. Finally, this paper closes with the lessons learned from testing and its influence on the flight build.

2. INTRODUCTION

The thermosphere, where low earth orbiting (LEO) satellites operate, is a dynamic region with large variability in winds and density. This is caused by the occurrence of solar and geomagnetic storms which cause the atmosphere to expand as it absorbs this solar energy. As a result of such storms, the density within this region have been known to vary by 300% to 800%[(1)] however the processes responsible for this variability are not understood. For LEO spacecraft, this creates an unpredictable external drag force that is exacerbated during increased levels of solar activity. As an example, this can be seen by reviewing the orbital altitude of the International Space Station displayed in Figure 1 below. This plot shows the regular decay and reboost cycles applied to the station to maintain the orbit. Note however that in July of 2000, an observed solar flare can be coupled with an unexplained altitude drop of 10km over a relatively short time period. Had this occurred a few months earlier when the space station was at 330km it could have created operational problems. A better understanding of this solar-terrestrial relationship needs to be understood in order to protect assets placed into low earth orbit.

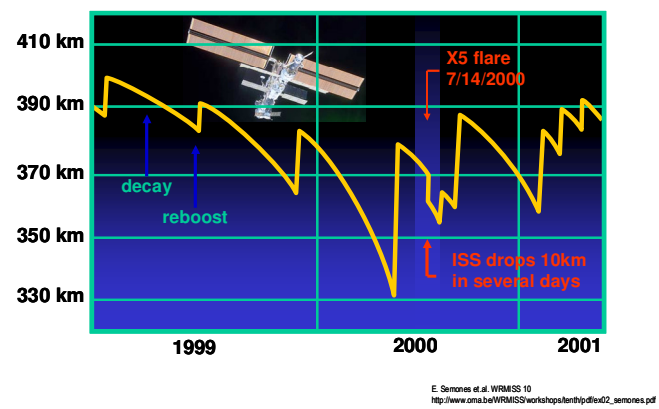

Figure 1: Altitude of the International Space Station over the course of 2.5 years[(2)].

2.1. The DANDE Mission Concept

DANDE aims to determine the atmospheric variability by actively measuring its own drag force over a period of 120 days. The concept takes advantage of the drag equation (1) where the force of drag is a function of density, coefficient of drag, cross-sectional area and relative velocity. In addition, the force of drag can be further compared as a function of mass and deceleration.

$$\vec{F}_D = \frac{1}{2}\rho \cdot C_D \cdot A \cdot \vec{v}^2 = m \cdot \vec{a} \qquad (1)$$

The DANDE mission controls each of the drag equation variables through design constraints and measurements:

- C_D, A: The spherical shape allows for a known coefficient of drag and constant cross-sectional area.
- $\vec{v}$: Wind velocity vector is measured with a Neutral Mass Spectrometer. This instrument is based off of a similar concept used by NASA Goddard. Relative orbital velocity is measured from Air Force Space Command Tracking.
- $\vec{a}$: Deceleration is measured with a unique accelerometer instrument utilizing six navigational grade accelerometers and a sophisticated 4th order band pass filter.
- m: Mass is empirically measured during the integration phase
- ρ: Density is solved

2.2. The DANDE Spacecraft/Background

The DANDE mission contains a single 18-inch sphere that houses the internal electronics and instruments. On the outside are three patch antennas and hundreds of solar cells which conform to the spherical shape. The spacecraft is spin stabilized about the orbit normal allowing the neutral mass spectrometer instrument to sample the incoming neutral gas once per revolution while also allowing each of the six accelerometers to sample the deceleration as it sweeps into and out of the ram-vector. To ensure the accuracy of the accelerometer instrument, the spacecraft center of gravity has to be near the geometrical center. This had large implications on the spacecraft architecture and structural design.

The spacecraft is required to interface to the launch vehicle via the ESPA ring containing a circular 24 bolt pattern. At the desired time following launch, the ESPA ring will eject (via an independent ESPA separation system) the DANDE system from the launch vehicle allowing the initiation of the mission. To connect the spherical DANDE spacecraft to the ESPA ring, an interface bracket was created (see Figure 2). Upon ESPA separation the DANDE system (containing both the interface bracket and sphere) are ejected. At a later time, the spacecraft will be commanded to independently separate from the interface bracket enabling the start of the science portion of the mission. There were discussions to determine if the ESPA separation system was required within this architecture as the DANDE sphere / interface bracket is sufficient to release the spacecraft from the launch vehicle. However it was considered cost and risk adverse as this would require additional coordination with the launch vehicle providers and would require a custom, untested electrical interface.

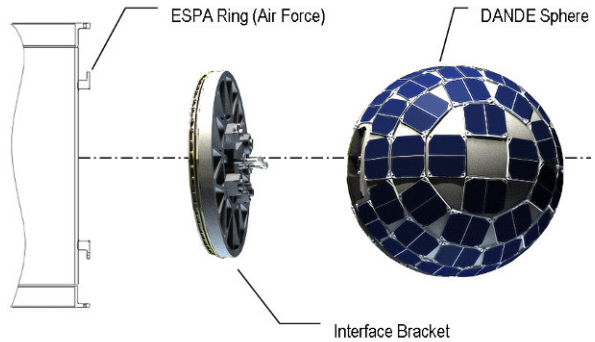

Figure 2: Baseline separation system architecture. The DANDE system contains both the interface bracket and the DANDE sphere.

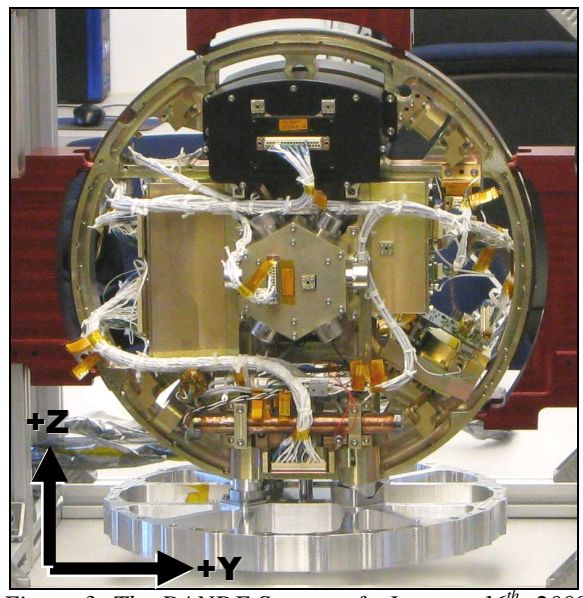

Figure 3: The DANDE Spacecraft, January 16th, 2009 Note: The DANDE coordinate system

The internal components of the DANDE spacecraft can be seen in Figure 3 above. In the center the hexagonal shape of the six-accelerometer instrument can be seen. The spacecraft spin axis is placed in the center of this instrument and is in/out of the page. At the 12-o-clock position are the battery boxes, the 3-o-clock position is the spacecraft computer, the 4-o-clock position is the neutral mass spectrometer instrument and the 9-o-clock position is the power regulation box. On the opposite side of the accelerometer instrument is the communication system.

The primary structure consists of a solid circular 'equatorial' plate that is 1.5 inches in thickness which interfaces to the hemispheres, mechanism bracket and

most of the internal components on the spacecraft. The mechanism bracket located at the 6-o-clock position on the spacecraft holds the release mechanisms and attaches to the interface bracket through a 4-point kinematic mount system.

3. DANDE ARCHITECTURAL TRADE STUDY

Many spacecraft architectures were envisioned during the pre-PDR design phase which each uniquely satisfied the design constraints. A thorough trade study was performed to balance the needs of the ESPA ring, preserving the spherical shape, instrument field of views, and available resources. The final system architecture decision had major implications on all subsystems and resources. As a result, feedback and team awareness of the trade study played a central role in the process.

Over the course of two months, along the same period of time that the secondary-level requirements were being solidified, architectures were narrowed down into six unique concepts. The concepts generally focused on the orientation of the 'equatorial plate' which is the backbone of the sphere structure, the physical shape of the interface bracket and the number of release mechanisms used. Figure 4 below shows four of the six final concepts.

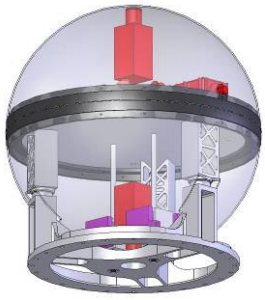

Figure 4.1: Three Legs, 2 Mechanisms 0º

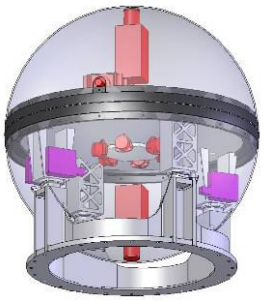

Figure 4.2: Four Legs, 2 Mechanisms 0º

Figure 4.3: Four Legs, 2 Mechanisms 90º

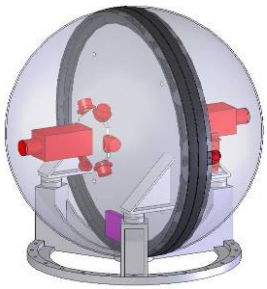

Figure 4.4: Three Legs, 1 Mechanisms 90º

In order to choose the "best" configuration a set of 12 factors were formulated and can be reviewed in Table 1. Cost and Mass were two factors initially considered however later eliminated from the study. This was justified since the cost to mechanism supplier is reflected in the total *Number of Mechanism* category while the cost of machining is reflected under *Ease of Manufacturing*. Mass was not considered since a thorough analysis had not been completed on all of the various architectures and may have improperly skewed the results. The % weight (or factor importance) was determined by the use of a 'Factor Weighting Matrix' where four students were asked to compare the importance of one factor to the other with a "yes" or "no" response. The results were tallied and a weighted average was calculated. In addition, each configuration was voted on a scale of 1-5 in how likely the design would accommodate each factor. The results were compiled and can be seen in Table 2 below which show that the winning configuration was where the equatorial plate is at 90º with respect to the interface bracket with two mechanisms and 4 short legs used to rigidly hold the components together (Figure 4.3). Upon comparing the winning architecture to the completed satellite 19 months later in Figure 3 above, the impact of this decision can be clearly seen.

Table 1: Factors ranked as a function of importance

Rank	Factor	Weight
1	Release Certainty	16%
2	Abiding by AFRL Requirements	14%
3	Sphere Center-of-Gravity	11%
4	Ease of Bracket Installation	9%
5	Tip-Off Speed	9%
6	Number of Mechanisms	8%
7	Accessibility to Bolts / Actuators	8%
8	Obstructed Solar Array Area	8%
9	"Feel Good Factor" (subjective)	7%
10	Ease of Manufacturing	5%
11	Allowable size of sphere	3%
12	Loading Symmetry	2%

Table 2: Trade Study Calculated Results

Rank	Configuration	Score
1	Four Legs, 2 Mechanisms 90º	4.21
2	Three Legs, 2 Mechanisms 0º	4.01
3	Three Legs, 1 Mechanism 90º	3.41
4	Three Legs, 3 Mechanisms 0º	3.26
5	Four Legs, 2 Mechanisms 0º	3.22
6	Three Legs, 1 Mechanism 20º	3.19

4. DESIGN FINALIZATION

Once the architectural configuration was finalized, several other important decisions followed which completed the top-level design of the system. The spherical diameter was set to 18 inches with a geometric center displaced 10.5 inches from the launch vehicle interface plane. A four-point kinematic mount system would be required to restrain the sphere to the four legs on the interface plate as the release mechanisms used were not designed for shear loading.

4.1. Industry Relevance

SpaceDev INC, a spacecraft mechanisms company and subsidiary of Sierra Nevada Corporation is located within a short drive of the CU-Boulder campus. SpaceDev was contacted about the DANDE mission and presented with the separation needs. Their engineers introduced the team to a mechanism under development called the Low Shock Release Mechanism (LSRM). The unique device contains a cam shaft which holds a release bolt under a desired load. Upon actuation with a Starsys™ High Output Paraffin Actuator (HOP), a pin is released which allows a small momentum wheel -geared to the cam shaft- to spin. After a short time period (<1 second), the cam shaft has rotated enough to separate the release bolt using an external spring and the preload strain energy to pull against the momentum wheel. The mechanism is called a "Low Shock System" because the inertia of the momentum wheel absorbs the initial energy allowing for a smooth, controlled release. The two mechanism units to be flown on the DANDE mission were engineering prototypes used during development and are certified for up to 5000 pounds of preload. Each mechanism is mechanically redundant, requires two paraffin actuators (although only one is needed to initiate a release). For simplicity we placed the mechanisms to allow for a single HOP actuator to trigger both mechanisms simultaneously. Since the DANDE and SpaceDev teams first met in February 2007, there has been a consistent and fruitful student / industry mentor relationship that has had tremendous benefits to both parties. This is detailed in a previous publication [3]

4.2. Separation System Design Details

With the SpaceDev LSRM set as the baseline, further geometric properties were determined: Since the mechanisms were not designed for shear loading, a series of kinematic mount would need to be used to ensure a rigid connection between the sphere and interface plate. In addition, the loaded and analytical ratings of the mechanisms allowed the team to determine a minimum offset distance between the bracket center and each of the four restraining legs to be three inches (the legs evolved to be more stub than its initial truss structure from the trade study), see Figure 5. This was a critical decision which had impacting effects on the structural analysis, natural frequency, required stiffness of the interface bracket, implied load on the kinematic mounts and impact on the spherical shape.

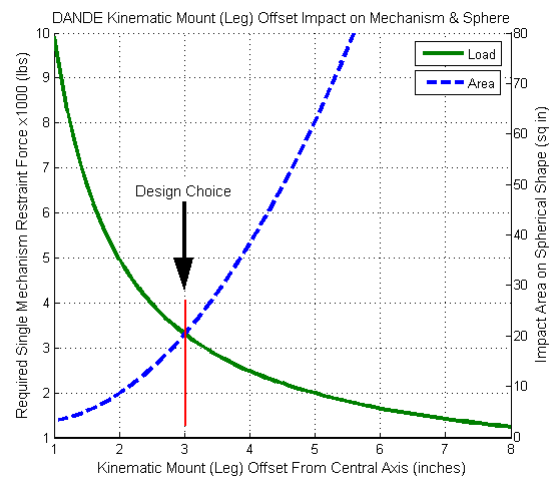

Figure 5: Kinematic Mount Offset Impact Study

A series of kinematic mounts were designed for each of the four legs to restrict motion between the interface plate and sphere. A kinematic mount is a geometrically constrained interface that allows motion in one direction but not in others. This is done to create a well seated connection given attainable manufacturing tolerances. This also ensures that a known stress is applied to the system. Kinematic mounts are often a spherical-plane interface which enables a point contact to ride along a smooth contact. This is an example of a 1-D mount which restricts translational motion in one direction while allowing motion along the plane in two directions and rotation in all three axes. Expanding on the idea a sphere/cone (ie cup/cone) 3-D mount interface which restricts motion in all three directions while still allowing rotational motion in three axes. Finally a sphere/trench (ie canoe/trench) 2-D mount geometrically restricts motion in 2 directions while allowing rotational motion in all three directions.

The DANDE separation system uses a combination of these three mount types to restrain the components in all six degrees of freedom as seen in Figure 6 below. The cup/cone 3D mount is located in the +B direction and locks the components in all three translational directions (A-dir, B-dir, C-dir). The canoe/tench 2D mount located in the –B direction restrains the two components in the two rotational axes when coupled with the 3D mount; (A-rotational

axis, C-rotational axis). Finally, the sphere/trench 1D mount located at the ±A directions in conjunction restrains the system in the final axis (B-rotational axis).

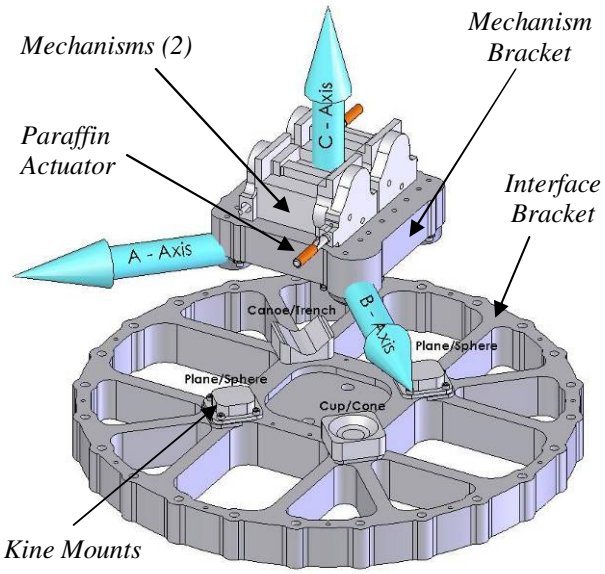

Figure 6: DANDE Separation system containing 4 sets of kinematic mounts.

Within this design the female mounts (flat surfaces) are set to be a harder material than the male counterparts. Therefore, when the point loading causes material deformation, dents in the flat surfaces will not form preventing motion (small dents in the male ie: spherical surfaces are expected and can be disregarded). Since this is a 4-set kinematic mount system, shims are used under one of the four female mounts to prevent a "wobbly 4-legged table." Finally male and female materials are different metals and surface coated to prevent cold welding upon flight.

5. ANALYSIS

The structural analysis of the DANDE system can be separated into three main categories. The first is the static and dynamic analysis of the primary structure and its implications on the separation system. Second are the kinematic mounts which are under intense compressive point loading and third the required preload of the mechanisms that will prevent gapping of the kinematic mounts and not exceed the rated mechanism loads. To avoid confusion the results of the analysis are displayed in terms of Margin of Safety (MOS) which is related to Factor of Safety (FOS) by equation 2 below. A MOS ≥ 0 meets the structural requirement.

$$MOS = \frac{allowable\ stress}{(actual\ stress)(FOS)} - 1 \quad (2)$$

5.1. Interface Bracket Structural Analysis (Static & Dynamic)

The AFRL University Nanosat Program requires the DANDE spacecraft to within stand a ±20g static load in all directions with a FOS of 2.0. In addition the spacecraft must have a minimum natural frequency of 100 Hz[(4)]. The COSMOS 2007 Finite Element Analysis (FEA) software was applied to the structure supporting the separation system. In addition, closed form solutions were used where possible as an estimate to increase the level of confidence in the FEA results. It was found that the stiffness of the interface bracket had a large impact on the structural integrity of the entire system resulting in its large, bulky appearance. The interface bracket was determined to have a static loading MOS of 0.68 which is experienced when the system is loaded in the Y coordinate direction (as seen in Figure 3). The MOS was inflated due to the dynamic analysis (discussed below). Figure 7 below shows the resulting static stress concentrations of the interface bracket under the maximum loading conditions. The maximum stresses were calculated to be 1.194e4 lb/in^2.

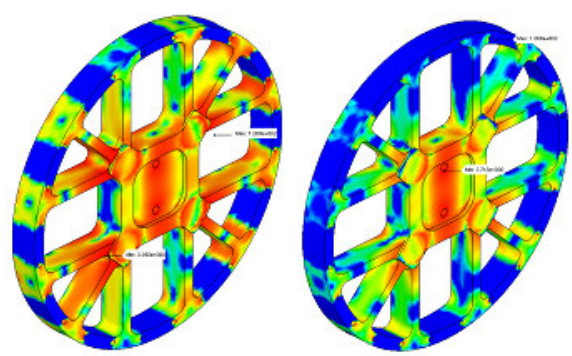

Figure 7: Interface Bracket Finite Element Results showing a lateral load (Y direction) and a normal load (Z direction).

The dynamic structural analysis proved to be a challenging problem because the rigidity of the kinematic were not known at the time. From early finite element models with a coarse mesh, the lowest natural frequency of the entire spacecraft was determined to be a rocking mode of the sphere on the interface bracket. This makes intuitive sense because a plate secured at the perimeter does not effectively restrain rocking motion in the center. The only way to restrain this mode is to make the interface plate's spokes stiff to restrict this motion. The predicted spacecraft natural frequency was found to be 110 Hz.

This was determined by assuming the entire sphere structure to be a stiff, solid piece sitting on top of the interface plate. Intricate details of the equatorial plate were not considered at this time as it was computationally expensive and the full design was not completed. Figure 8 below shows the fundamental mode shapes for the first two natural frequencies.

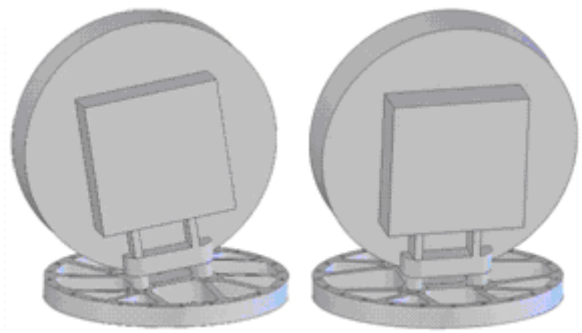

Figure 8: Fundamental Frequencies & Mode shapes, 110Hz (left side), 114Hz (right side).

5.2. Kinematic Mounts

The kinematic mounts are a unique analysis problem in which a spherical surface is compressed on a flat plane. This creates a point load or an infinite stress concentration. As a result the spherical surfaces deform and flatten out over time. This by design is accepted since the spherical surfaces are a softer material which still allows the opportunity for sliding. To perform analysis on these mounts a series of closed form solutions based on Hertizian stress theory[(5),(6)] were derived for the DANDE system. First however a series of assumptions were established:

- The maximum normal loading per kinematic mount is 5333 pounds. This comes from a combination of the mechanism maximum preload (distributed equally over all four mounts) and external forces (distributed equally over 2 mounts).
- All shear (lateral) loading was taken through the cup/cone mount.
- There are no induced strains from miss-tolerances between components due to kinematic mount design. The same is true for thermal expansion.
- The male/female material selection is steel / and hard aluminum (respectfully). *(Please contact us for further details regarding the material choice and properties trade study).*
- The required factor of safety of 2 was integrated into this stress analysis by doubling the applied exterior loads. This conservative method was done under the direction of an industry advisor since the correlations between load and stress are highly non-linear.
- The material yield stress was multiplied by a factor of 1.6 since they are experiencing a compressive bearing load which is acceptable according to a discussion in Roark's tables[(6)].

From these assumptions the analysis was performed on the 3 unique types of kinematic mounts designed for the DANDE separation system. The radii of the kinematic spheres and the angle of the flat planes were optimized to have a positive margin and satisfy the complex geometric constraints. This was a rigorous process lasting several weeks to ensure all constraints were met. As an example, the methodology for the simplest mount, the 1-D sphere / plane is discussed. Equations 3-6 are the fundamental results from the Hertizian stress theory derivation:

C_E	=	Effective Material Modulus
E_1, E_2	=	Material Modulus (Male/Female)
υ_2, υ_2	=	Possion's Ratio (Male/Female)
P	=	Sphere / Plane Interface Load
d	=	Sphere Diameter
σ_c	=	Compressive Stress (at max loc.)
τ_c	=	Shear Stress (at max location)
$A_{contact}$	=	Deformed Area on Sphere (contact)

$$C_E = \frac{1-v_1^2}{E_1} + \frac{1-v_2^2}{E_2} \qquad (3)$$

$$(\sigma_c)_{\max\ sphere} = 0.918\sqrt[3]{\frac{P}{d^2 C_E^2}} \qquad (4)$$

$$A_{contact\ sphere} = 0.520\pi(PdC_E)^{\frac{2}{3}} \qquad (5)$$

$$(\tau_c)_{\max} = \frac{1}{3}(\sigma_c)_{\max} \qquad (6)$$

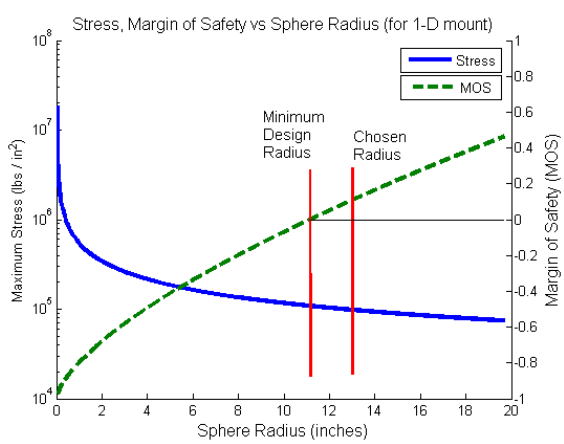

Figure 9: Sphere/Plane Kinematic Mount Analysis

Figure 9 above shows the stress concentration plots as a function of sphere radius. The solid line represents the stress concentration while the dotted line represents the corresponding margin of safety (MOS) based on the material properties. The vertical solid line at a sphere radius of ~11 inches is the minimum required radius to meet a perfect (zero) margin of safety. The solid line at a sphere radius of 13 inches represents the chosen sphere radius with a corresponding margin of 0.11. The 3-D mount was the most complex as there is not a single point load, but a line load around the edge of the cup/cone interface. The Hertizian stress equations were modified to represent a cylinder on a flat plane where a new variable is introduced: L is the length of the cylinder.

$$(\sigma_c)_{\max\ cylindar} = 0.789\sqrt{\frac{P}{d\ C_E L}} \qquad (7)$$

$$A_{contact\ cylindar} = 0.509\pi\sqrt{P\ L\ d\ C_E} \qquad (8)$$

The stress analysis results for the male components of the three different kinematic mount types are shown in Table 3 below. The contact area is the predicted circle of flattened area where the kinematic mount will observe the predicted yielding. Figure 10 below shows the 2-D "canoe" kinematic mount which contains the designed 10 inch radius.

Table 3: Analysis results for the three kinematic mount types.

Mount	Design Sphere Radius	Contact Area	Margin of Safety
1-D	13 inches	0.173 in2	0.11
2-D	10 inches	0.109 in2	0.07
3-D	0.95 inches	0.180 in2	0.04

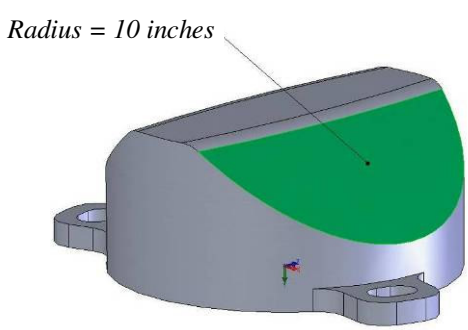

Figure 10: A CAD drawing of the 2-D male "canoe" kinematic mount with a spherical radius of 10 inches shown in green.

5.3. Mechanism Preload

During the design phase, the interface bracket and the mechanism bracket were set to be greater than 1.25 inches in thickness to support the assumption that the plates are rigid and do not significantly deflect when loaded. This greatly simplifies the analysis at the cost of adding additional mass. This assumption creates the following boundary conditions: 1) while loaded, the mechanism bracket and adapter bracket remain parallel, 2) gapping (see next section) does not occur unless the load applied on the mechanisms exceeds the set preload, 3) The restoring force of the deformed adapter or mechanism bracket is infinitesimal and does not contribute to gapping or loss of mechanism preload. Table 4 below shows the margin of safety for the SpaceDev LSRM mechanisms when loaded within the DANDE system. The certified rating margin displays the loading limits the mechanisms can experience while the analysis rating is what the mechanisms were designed to withstand. Note that the mechanisms will not exceed this certified rating at the maximum testing load of 24gs. Additionally, note that there is a positive margin at the 40g mark which satisfies the AFRL 2.0 factor of safety requirement.

Table 4: Mechanism Preload Margin of Safety (MOS) under various loads

Load Scenario	MOS (Certified Rating)	MOS (Analysis Rating)
20g Design Load:	0.26	1.01
24g Design Test:	0.05	0.68
40g Design Yield:	-0.37	0.01

Gapping

A second analysis was performed without the rigid interface assumptions. This was necessary as the flexing of the interface surfaces can cause a loss of preload. This causes potential for one set of kinematic mounts raise off the surface and no longer touch which is referred to as "gapping." When gapping occurs, it puts additional loads onto the remaining kinematic mounts and mechanism thus invalidating the previous analysis. The gapping calculations performed for DANDE treats the separation system as a single mechanism / restraint interface and determines if the preload relative to the force exerted is high enough to account for variations such as thermal expansion, interface stiffness etc. The analysis takes into account: mechanism preload, all material stiffnesses (bolts, interface bracket, mechanisms), thermal expansion, temperature extremes, temperature at the time of mechanism preload and preload uncertainties. The material stiffnesses (dependent on

geometry) of the interface plates were calculated by equation (9) below. Where P is load, L is length, E is material modulus, I is the geometrical inertia and δ is deflection.

$$\delta = \frac{PL^3}{3EI} \rightarrow \frac{3EI}{L^3} = \frac{P}{\delta} \qquad (9)$$

The results indicate that under normal loading (Z direction as seen in Figure 3) the flexibility of the plate subtracts 497 pounds of loading to the kinematic mounts. Under lateral loading (X, Y directions) the flexibility of the interface plate creates a loss of 1023 pounds to two adjacent sets of kinematic mount. Since the preload of the mechanisms is greater than the loss caused by flexural deformation, it can be concluded that gapping will not occur.

6. TESTING

It was determined that an engineering design unit (EDU) of the full DANDE primary structure would be built prior to the Critical Design Review (CDR) as a learning experience for the structures team. This effort was successful and the EDU was used over the following 12 months for a full level vibration test, separation tests (some in microgravity) and general spacecraft tests such as electrical grounding, antenna testing etc. The tests have identified areas of success and areas for improvement which has forged a flight unit with a high level of confidence and workmanship.

6.1. Vibration Testing

A vibration test was performed on the DANDE engineering unit in March of 2008 at the Ball Aerospace Corporation facility in Boulder, CO. The engineering unit primary structure was complete and installed with 10 aluminium, steel and brass blocks placed around the spacecraft to represent the internal components. Figure 11 shows how these mass simulators were placed within the plate. The mass of the entire EDU system was within 0.5 kg of the maximum contingent mass of the spacecraft and the center of gravity was within 0.25 inches of the required location. Six tri-axis accelerometers were placed throughout the spacecraft to identify mode shapes and the health of the spacecraft during the test. Particular interest was focused on the spokes of the interface bracket ring as that was where the most characterizing deflection was predicted when the fundamental mode shape was excited. By understanding this location, the finite element model could be verified with increased knowledge about the locations of the strain energy concentrations.

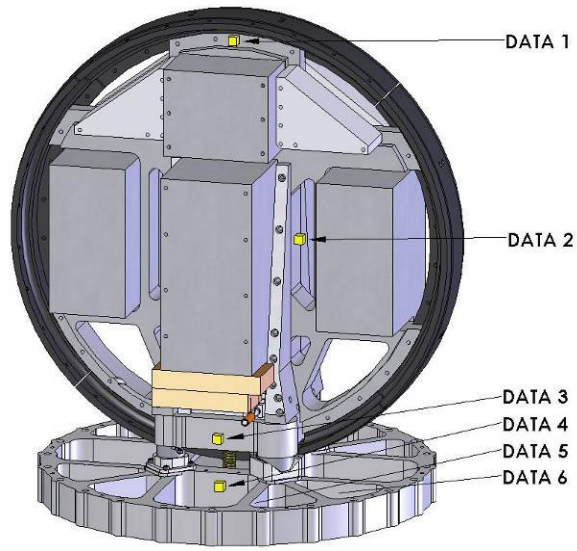

Figure 11: The internal components of the DANDE EDU vibration structure and location of accelerometer data sensors.

Test Profile

To simulate the true launch vehicle hole pattern, a layer of washers were placed between the interface bracket and vibration table. This ensured that the primary mode shape would not be affected by the center of the interface bracket impacting the vibration table base. The test profile consisted of a sine burst test and a random test to be performed in each axis. The sine burst contained a ramp-up, 6 full cycles at 24g's and then a ramp-down, all performed in less than a second. These sine bursts were conducted with great caution and in incrementing powers as the team did not want to over-test beyond the 24g load. The random profile spanned from 20 to 2000 Hz starting at 0.01 g^2/Hz (power spectral density, PSD) and reaching 0.05 g^2/Hz from 50-1500 Hz lasting for 120 seconds[(4)]. The test was incremented for 30 seconds at -12dB, -9dB, -6dB, -3dB powers each time carefully reviewing the result and calculating if the spacecraft responses were linear. Equation 10 below shows the method derived from the Miles Equation[(7),(8)] to calculate the load imparted on the structure (in terms of acceleration root-mean-square).

$$(\ddot{x}_{rms})_{exerted} = 3\sigma\sqrt{\Delta f_{1/3}(PSD)} \qquad (10)$$

The equation determines the g-load of the random response by measuring the span in the natural frequency peak (Δf) at 1/3rd of the amplitude, the power spectral density (PSD) at that corresponding frequency and a statistical Gaussian value of 3σ. If it was determined (from the random testing power) that

the structure was going to be loaded beyond 24g's then notching was induced.

Although some would argue that the practice of notching (reducing power at the natural frequencies) compromises a test, in reality it creates a more realistic environment of the launch vehicle / spacecraft system and prevents over-testing. With the use of a vibration table there is an infinite amount of available energy. When the satellite is excited at a natural frequency, energy is absorbed by the spacecraft to excite these modes. The vibration table feedback loop notices the loss of energy and compensates until the specified energy is implied. On the contrary, a launch vehicle has a finite amount of energy; and the spacecraft like before will dampen out the extreme values at the natural frequencies. Thus notching is an important practice to prevent over-testing.

Finally, between each major test a ¼-G sine sweep was performed and overlaid with the initial responses to determine shifts and signs of failures.

Figure 12: DANDE spacecraft on the vibration table.

Test Results

The vibration test was successful in that the engineering unit experienced full flight-like loading conditions without showing signs of major problems. The sine-sweep performed identified the primary natural frequency of the system to be at 85 Hz and the accelerometer data confirms that this fundamental mode shape is the rocking of the interface bracket as predicted. The next primary mode was >500Hz as predicted. The sine burst tests were completed in all three directions with a maximum of a 24g load applied at the base. During the lateral testing (configuration seen in the Figure 12 above) the sine burst applied created a 31g load at the top of the sphere which is a signature of the system stiffness. The random test was performed in the vertical (Z-direction, X-direction per Figure 3). For all of the random tests, notching was found to be necessary to ensure that the spacecraft never experienced above a 24g load at the base of the setup.

The final Y-direction was not performed due to time constraints at the facility. One area of interest is that the first random test in the vertical (Z-direction) experienced a 56% growth within the first fundamental frequency which was outside of our passing criteria. This is attributed to a "seating" of the kinematic mounts as this was the first random test to be performed on the system. After a mechanism preload inspection, the test was re-run and passed. Figure 13 below shows a typical plot that the team received showing the natural frequencies of the system before and after the testing. The similar plots show that the spacecraft stiffness signature remains stable during the tests indicating passing criteria.

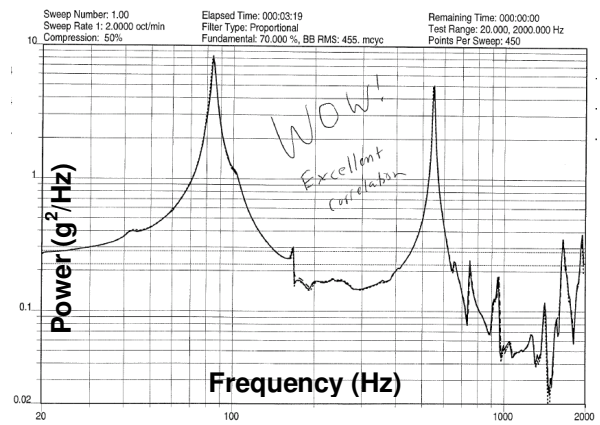

Figure 13: Three overlaid sine-sweeps plots of the DANDE spacecraft in the configuration shown in Figure 12. The plots were taken between the major tests and show the structural signature remains unchanged. The author wrote a note on this plot in '08 to capture the mood of the team upon seeing this result.

Test Discussion

The vibration test identified several areas of improvement which the team has designed into the flight structure. Of particular importance the spacecraft fundamental frequency was 30% off of the previous analysis (discussed in section 5.1). After an in-depth investigation, it was determined that the assumption of having a rigid sphere component was inaccurate and prevented the strain energy of the equatorial plate from adding to the primary mode. As a result more computational demanding higher-fidelity finite element analyses were completed with the entire structure and found the natural frequency result to be within 15% of the tested value. It is unknown at this time how the kinematic mount / mechanism interface plays a role in this discrepancy. Several design changes have been proposed to further stiffen the

interface and equatorial plate however after discussions with the Air Force, a waiver for the 100 Hz minimum frequency is likely the best option to avoiding new risk and need for further testing when significantly modifying the structure.

The kinematic mounts experienced the flattening patterns that were expected from the analysis. In addition, specific interfaces experienced a galling where some of the metal at the surface had a rough, braided pattern. This mainly occurred during the initial seating of the mounts and is attributed to the residual friction within the system when initially preloaded. This will be mitigated by applying a surface coating to the male mounts and by using Molybdenum Disulfide Powder on the flight unit to lubricate these joints.

6.2. Microgravity Separation Testing

In December of 2007, DANDE was awarded an opportunity to test the separation system aboard the NASA C-9 Microgravity Aircraft. The opportunity was tremendous as it allowed the team to test the separation system in a relevant environment. The goals of the test were identified to demonstrate that:

- The spherical sector and interface bracket separate at a rate of 1 meter/second
- The two separating parts do not undesirably contact during separation
- The adapter bracket separates from the spacecraft and clears the sphere by at least one foot.
- Ambient factors such as the wall of the aircraft, or human hands do not interfere with the separation system systems during the test.

Test Results

During the testing on the C-9 aircraft the team was capable of conducting 5 successful separations. The test required resetting the mechanisms after each test and was difficult as the team experienced cyclic microgravity and 2g environments every 90 seconds. Each of the five tests went well and provided valuable information regarding the system. Data was captured on video footage taken during the flight (Figure 14) showing a clean separation of the two components from one another. Clear field of view of the interface bracket motion after separation is visible in four of the five separations. Each of the four trials shows minimal rotational motion of the bracket relative to the ejection velocity and adds a high level of confidence that the bracket will not contact the spherical portion of spacecraft after separation has occurred.

The separation velocity of the equatorial plate was measured at 0.242 m/s which is within 3% of the predicted velocity of 1 m/s relative separation.

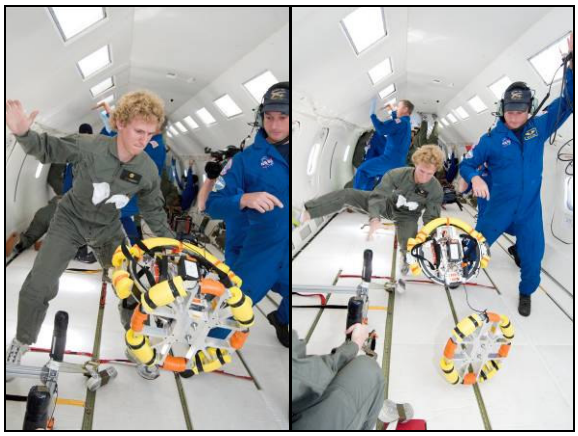

Figure 14: DANDE separation system just moments before and after separation.

Test Discussion

The LSRMs performed well during the test. The mechanisms released when activated by two servo devices simulating the paraffin actuator system which is the baseline for the flight. This change was necessary since the microgravity separation window is ~15 seconds and is much smaller than the 70±15 seconds nominal release time of the paraffin actuators.

The LSRMs by design are intended to be used only a small number of times (<25 without being rebuilt) especially after completing multiple full cycles of being set, loaded, and released. Prior to using the LSRMs for the test they were reset over 500 times during practice. These repetitions were not loaded cycles but were intended for the team to familiarize themselves with the mechanisms in order to provide the highest probability of success during the testing period. Throughout the 500 repetitions, the LSRMs performed as expected and did not noticeably degrade. During the test, up to 30 complete reset cycles (with various preloads) were performed. During these cycles the personality of the mechanism began to change. They took more time to reset and required more finesse movements to preload the bolts. The mechanisms were tested beyond their intended lifetime (without servicing) and performed exceptionally for the test showing clean releases and performing as expected.

The mechanisms upon returning to SpaceDev were inspected and disassembled for post-analysis. The results show that the mechanisms performed as expected, were clean and did not have significant ware on the chem-film coated actuator shafts. There was however a damaged shear pin on one of the reset driver axles which is only required during reset. Additionally it was found that the actuating shafts had an increased amount of sliding friction preventing the mechanisms from being reset without finesse. From

this post-inspection, there was no visible evidence that the mechanisms were overstrained during the test or that their ability to maintain preload was compromised despite the 500 non-loaded resets and the 30 full loaded cycles.

Figure 15: DANDE Separation system with microgravity support structure. The experiment PI (author) is pictured with setup.

6.3. Flight Qualification Separation Testing

Multiple separation tests have been performed to quantify the system behaviour (outside of microgravity). The first formal testing took place before and after the vibration test on the engineering design unit. The test was the only opportunity the team would have to perform a post-vibe separation test. Video of the separation showed a smooth even release without evidence of interface bracket tip-off. The test used gravity to separate the bracket instead of the flight springs which were not available at the time of this test. In addition, the mechanisms were actuated by hand instead of using the flight HOP actuators for simplicity.

A second series of tests were performed in the spring of 2009 on the flight unit. It was determined that a series of 8 successful bracket releases in the flight configuration would be sufficient to qualify the system. The test setup contained a higher fidelity system with flight springs, flight structure, paraffin actuators and spacecraft electronics. This test series was the first which incorporated the flight paraffin actuators into the system. This improved fidelity but caused new personalities of the system to emerge. The actuation was a slower push compared to the fast action microgravity and vibration separation tests. The slow actuation allowed the mechanism release timing to be slightly off (even though each actuator is coupled to both mechanisms). This uncertainty creates a new system level risk in that there is a potential mode where the differential in mechanism release timing allows the release bolts to become pinched along the side, preventing a clean release. Several solutions have been conceptualized and discussed with the team and engineers at SpaceDev. The least invasive solution was to remove a small non-structural piece of the mechanism material originally aimed at guiding the release bolt.

6.4. Thermal Vacuum Qualification Separation Testing (In Planning Stage)

The DANDE team is currently planning to perform a thermal vacuum test as the final qualification of the separation system. The primary goal of this test is to show that the mechanisms allow the release bolts to separate and that the kinematic mounts do not bind in a vacuum at the temperature extremes. Since extensive testing has been performed on full system separations (speed, tip-off rates, etc), this test does not include measures to perform a full separation where the interface bracket is completely removed from the sphere. This simplifies the test and prevents risk to the vacuum chamber facilities.

The test will be performed in a vacuum greater than 1e-5 torr and at three temperatures: Nominal 15°C, hot 60°C and cold 0°C. This test will contain all flight parts with the SEP system in its flight configuration. All components that the SEP system are dependent on such as the flight computer, power regulation boards, wiring harness and batteries will be included in this test. Figure 16 below shows a conceptual view of the setup in a vacuum chamber. The mechanisms will be supported by a test fixture which allows the interface bracket to deploy in the direction of gravity. Upon release, the interface bracket will move (approximately 0.5 inches) and land on an elastic material to absorb the release energy.

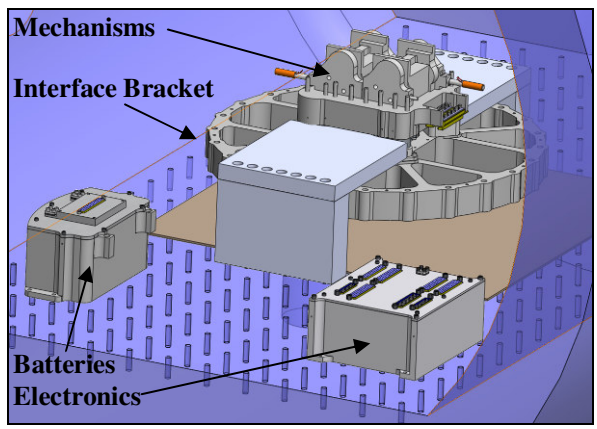

Figure 16: Conceptual Design for Separation Testing in a Thermal Vacuum Chamber.

7. FLIGHT BUILD / LESSONS LEARNED

The flight build separation system was completed in the fall of 2008. A few "nice-to-have" modifications were made to the flight model as additional mass became available including: 1) Modifying the mounting pattern of the female kinematic mounts to allow for ease of integration, 2) Increasing the thickness of the interface bracket spokes as the spacecraft mass margin is positive and this will help mitigate the low natural frequency experienced during the vibe testing (although this not required). In addition several new practices were introduced which further improved the compatibility of the SEP system with the rest of the spacecraft:

- Thermal coatings – exposed aluminium on the interface and mechanism bracket were coated anodized black, all metal-on metal interfaces were conversion coated with gold alodined. The male kinematic mounts were anodized black as a further method to prevent galling or cold fusion.
- An electrical interface connector was added within the system. The space certified connector (provided by Planetary Systems) contains a force-less release. The connector interface contains Viton rubber and oversized holes to allow for miss-alignments within the kinematic mount system.
- Strain gages were added to the mechanism release bolts to further ensure that the proper preload of 5000 pounds was applied (originally determined by a torque wrench). These gages were used by the ground support equipment only and are cut prior to launch.

All of these noted changes were performed prior to the flight qualification of the system and are clearly documented. The separations team currently has more than 50 signed drawings, analyses, and procedural documents which will continue to grow until the system successfully operates on orbit to ensure its success and for use by future student missions.

8. CONCLUSIONS

The separation system for the DANDE spacecraft was a student-led development process spanning two years. The system was mentored by industry personnel and testing facilities were provided to allow the student team to develop a high fidelity, robust system for flight on an Air Force launch vehicle. The design process followed the University Nanosat Competition guidelines and the system was tested in various forms to prove that these requirements were met. This DANDE separation system is a unique design containing two off-the-shelf mechanisms, and a four-point kinematic mount system. Extensive analysis was performed in addition to vibration and separation testing with a planned thermal-vacuum test to qualify design. The team has a high level of confidence with this system and looks forward to witnessing its success on orbit.

9. ACKNOWLDEGEMENTS

The DANDE team consists of over 40 students who have worked incredibly hard to create a high fidelity, award winning spacecraft that will soon be launched to measure atmospheric drag using a conceptually unique approach. While I have discussed my role of being the driving designer & test conductor for the DANDE separation and structural subsystems, it would not have been possible without help and support of the entire team of which I am grateful. I would especially like to thank the DANDE advisors, collaborators (especially at SpaceDev) and support staff who challenged us to go above and beyond when building a real, world-class spacecraft.

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