DESIGN & INTEGRATION OF THE AEHF PAYLOAD WING DEPLOYMENT ASSEMBLY

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ABSTRACT

The Advanced Extremely High Frequency (AEHF) system of space vehicles is a follow on to the existing Milstar system, which provides secure communications for the United States military and its international partners. Each AEHF space vehicle (Fig. 1) is equipped with two wings which carry the bulk of the satellite's payload – the antennas. These antenna-laden Payload Wings are securely locked in a stowed configuration during launch and, upon reaching orbit, are deployed to a precise, repeatable, stable position. Deployment of these wings is carried out by two Payload Wing Deployment Assemblies, one per wing.

This paper discusses the design of the hinged deployment system, the Payload Wing Deployment Assembly (PWDA). The payload wing is held with launch locks and once released, deploys via hinges and a 1.5-meter, damped, graphite epoxy hinged strut. The extensive test program for the mechanism is discussed in [1]. The first four wings have deployed successfully on orbit.

Figure 1: AEHF Satellite

1. PWDA DESCRIPTION

Each Payload Wing Deployment Assembly consists of two Hinge Assemblies, two Launch Restraint Assemblies (LRA), one hinged Strut Assembly, and one Strut Guide. The LRAs restrain the Payload Wing during launch while the Strut Guide secures the Strut Assembly. Once the LRAs are fired, a kick-off force is supplied by springs located within the LRAs. Deployment is then driven by laminar springs on the Hinge Assemblies and on the Strut Assembly. Two large wire harness assemblies cross the joint and provide substantial resistance to motion. The deployment speed is controlled by an eddy current damper located at the hinged joint of the Strut Assembly. Total deployment time is approximately 20 seconds.

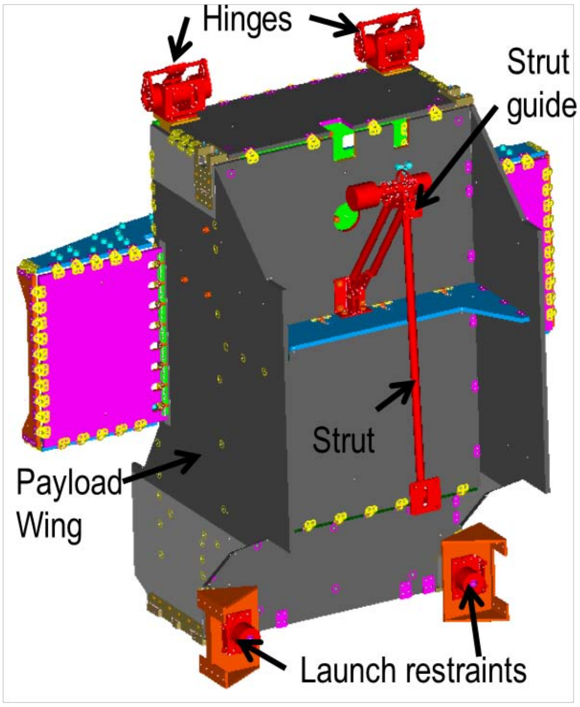

Figure 2: Stowed PWDA Looking Outward from Spacecraft (wire harness not shown)

Together, the PWDA Strut Assembly, the Payload Wing and the Spacecraft Bus create a 4-bar linkage. As the Payload Wing deploys about the two PWDA Hinges, the Strut Assembly unfolds from its stowed configuration and eventually locks out in a fully deployed configuration. This fully deployed configuration determines the deployed angle of the Payload Wing. Once deployed, the locked out Strut maintains the pointing requirement of the Payload Wing and provides stiffness to the system, increasing the deployed frequency of the 300-kg Payload Wing.

2. COMPONENT DESIGN

The PWDA was sold as a low risk heritage design that differed little from previous missions and required only minimal engineering effort. As the design progressed from preliminary design reviews to critical design reviews and beyond, several key design changes had to be incorporated to ensure the PWDA met the requirements of the AEHF Space Vehicle. This minimal effort expanded into several discoveries and much more work along the way.

2.1. Strut Assembly

The Strut contains an inboard graphite composite tube, a center hinge assembly, and twin graphite composite outboard tubes. The tubes are attached to clevises via spherical bearings (monoballs). The inboard clevis attaches to the vehicle bus structure and the outboard clevis attaches to the under-side of the payload wing. The center hinge assembly contains a laminar spring stack of nine spring laminates with a total torque output of 25 N-m (220 in-lb) over 135^{o} rotation. In the stowed configuration (Fig. 3), the strut is folded underneath the payload wing and is cradled in the strut guide. The strut unfolds during deployment, exerting torque about the payload wing hinges while damping the motion as well. Upon reaching the fully deployed position, the Strut locks into place (Fig. 4) with an over-center latch. This latch prevents the Strut from over-deploying and ensures that the Strut cannot be back-driven once it is locked out.

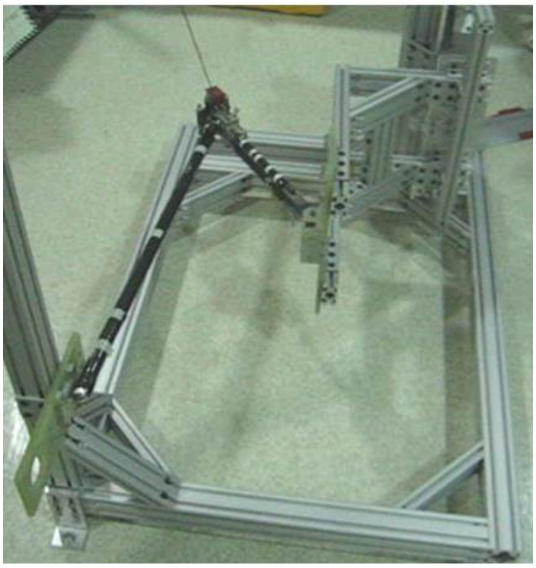

Figure 3: PWDA Strut installed on Torque Test Equipment in Stowed Configuration

Figure 4: PWDA Strut installed on Torque Test Equipment in Deployed Configuration

2.1.1. Strut Material

The Strut Assembly was originally intended to be a near replica of the orbital-arc-welded titanium deployment strut used on Milstar. The plan for AEHF was to modify the length of individual struts and re-use this design. The Milstar titanium strut worked great kinematically, and a development unit of the size required for AEHF was built, and the critical design review was completed. However, somewhere buried in a spacecraft-level pointing analysis was the assumption by the analyst that the strut had zero thermal expansion. Obviously a titanium strut does not have a zero coefficient of thermal expansion and using the correct value had the assembly violating the AEHF payload pointing requirements.

A re-design ensued to use graphite strut tubes. It was determined that if we had the struts completely made of graphite with a near-zero expansion, and kept the center fitting metallic and the end clevises metallic, the pointing requirement would be met. The composite strut supplier was able to produce a unique design with all-composite end fittings as shown in Fig. 5.

Figure 5: Composite Strut End Fitting

2.1.2. Strut Guide

Like the Milstar strut, the PWDA Strut Assembly has monoballs on each end to ensure that binding does not occur during deployment. This extra degree of freedom has the effect of allowing the strut assembly to “flop” to one side or the other when not fully deployed. During launch, this unrestrained movement would be unacceptable, so the Strut Guide (Fig. 6) was designed to stabilize the Strut Assembly. The Strut Guide cradles the Strut Assembly at the center hinge and prevents it from traveling back and forth during both ground operations and launch. The Strut Guide, which interfaces to the Strut via flexible closed cell foam, also provides some damping and serves to help isolate the Strut Assembly from the Spacecraft Bus during launch.

Figure 6: Strut Guide

2.1.3. Eddy Current Damper

The supplier of the Milstar damper was not in business anymore by the time the AEHF program began. However, another Lockheed Martin program eddy current damper could be re-used with a minor modification of an additional pass in the gear train. The AEHF thermal environment was predicted to be much colder though, and the grease lubrication used previously would not work at the -95°C temperature. The supplier had a solution though.

The lubrication method was identified to be the highest risk at the damper critical design review. The solution proposed by the supplier was to omit the grease normally used (which some remember as the supplier as saying they had done before as well). The heritage approach consisted of using grease in conjunction with advanced anti-friction coatings known as diamond-like-coatings or DLCs. Based on the technical information available at the time, use of the DLC coatings alone was considered to be a viable approach. Technical data indicated that the DLC provides extremely low friction in vacuum. As the specific capacity and endurance limits for the DLCs were unknown, a lube life test was added to the development program to mitigate the risk. The lube life test was performed in vacuum at temperature but at a nominal load. This test was successful. The peak load case (which resulted in failure in qualification) was omitted from the test. The peak load occurs momentarily during strut deployment and was not believed to be significant in assessing the life of the lubricant. This would later prove to be a critical omission as the peak load in vacuum exceeded the wear capacity of the DLC coating inducing rapid failure (Fig. 7). A second lube life test was performed that did include the predicted peak loading, but was conducted at ambient pressure and temperature. This test was also successful.

The qualification life test was the true “test like you fly” test with the loads, vacuum and temperatures accurate. The damping rate on the second test cycle was measured at 2000 N-m/s (18000 in-lb/s) versus a requirement of 560 N-m/s (5000 in-lb/s) maximum. Supplier concluded the readings do not make sense, however, once we disassembled the unit, we found significant wear on mating surfaces. Fig. 10 shows the wear on a gear pin. The test failed as the lubrication method has high wear with the conditions of loads and vacuum that the eddy current damper experiences. The temperature requirements were re-examined and with more maturity in the program, the revised values were much warmer (-25°C). The qualification life test was re-run with grease with the coating and passed.

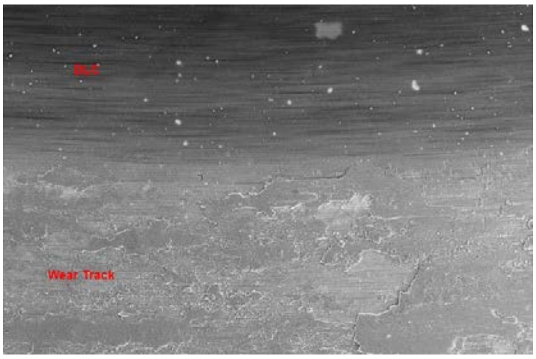

Figure 7: Wear Track on Axle of Damper Gear

2.2. Hinge Assembly

A hinge trade was done during the preliminary design stage of the PWDA to determine the best heritage hinge to use. The Milstar hinge was oversized, very heavy and large for the AEHF application. Typical hinges for interpanel use on solar arrays were too small, not having the required strength or stiffness. A hinge used on the Gravity Probe B satellite we built for NASA seemed to be just right – high strength, high stiffness, right size.

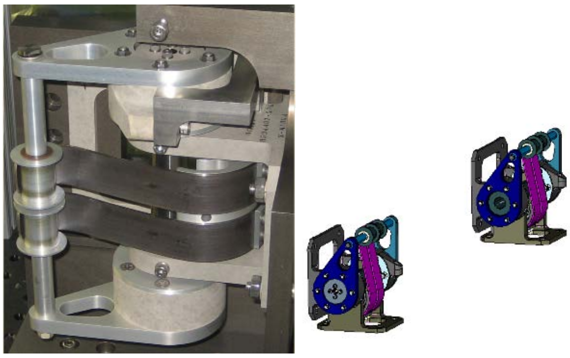

Figure 8: Hinge Assembly

The hinge assembly (Fig. 8) contains two laminar spring stacks, each with six laminates, exerting a total of 17 N-m (150 in-lb) over 90^{o} rotation. The hinge shaft is supported by drawn-cup needle bearings and thrust bearings on both ends. However, the bearings were made from common steel and thus could corrode. Few, if any, needle bearings are made from stainless steel in this size and certainly none that could be obtained in a timely manner. A test was devised with bearings lubricated with Bray 601 and others unlubricated as control samples to determine if the grease would be effective at preventing corrosion. The bearings were tested for weeks at 75% humidity at 32°C, and then again at 95% humidity at 60°C. The control samples corroded as expected. No corrosion was evident on the greased bearings and no increase in torque was evident due to the exposure.

2.2.1. Shear Inserts

Shear Inserts, or “shearserts”, were used in the PWDA mechanisms to hold them in position on the spacecraft. Shearserts are commonly used to connect panel edges or other structural members where position variability is needed, but typically many are used to share the load. However, only four were used at each hinge interface resulting in a highly loaded state that they were not designed for. This led to failures during hinge static load testing. An image of the shearsert with a failed bondline is shown in Fig. 9. The bonding process was revised and additional epoxy was added for sufficient strength in the joint. Later, the shearserts were redesigned for additional margins and for installation simplification.

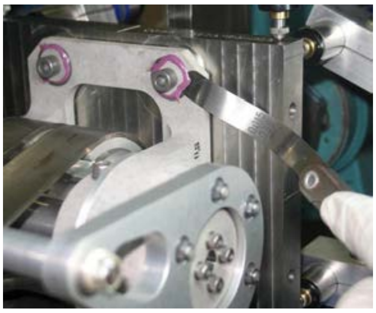

Figure 9: Hinge Shearsert with Failed Epoxy Bondline

2.3. Launch Restraint Assembly (LRA)

The LRA (Fig. 10) is modified from a design used on a previous Lockheed Martin program. It contains a rod preloaded into a Split Spool Release Device (SSRD), a high-preload low-shock release mechanism, and a bolt-catcher assembly to catch and hold the preload rod after it is released by the SSRD. A cup/cone interface kinematically constrains the payload wing in the shear directions. A kick-off spring provides an initial force to start the wing deployment and overcome any stiction in the system.

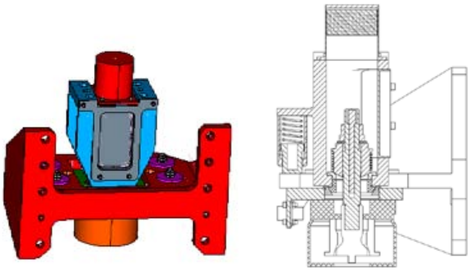

Figure 10: Launch Restraint Assembly

2.4. Torque Margin

The design of the Hinge’s and Strut’s laminar spring was bounded on both sides. On the lower end, it was bounded by the requirement that the Payload Wing deploy with at least 100% torque margin. On the upper end, it was bounded by the need to limit the torque on the eddy current damper to a level that the damper could withstand. Due the fact that the damper was operating near the upper limit of its allowable input torque and the high resistant torques of the large harness assemblies, this turned out to be a fairly tight design space.

To calculate torque margin, the quasi-static torque input from each component had to be determined and then combined as in Eq. 1 about the main hingeline where MA is the mechanical advantage of the Strut at a given deploy angle, relative to the hinge line, and FR is the ratio of friction torque divided by available torque.

$$TM = \frac{T_{available}}{T_{resisting}} - 1 = \qquad \text{Eq 1}$$

$$\frac{T_{hinge\ line} + MA\, T_{strut}}{FR_{hinge\ line} T_{hinge\ line} + T_{wire\_harnesses} + MA\,(FR_{strut} T_{strut} + T_{damper\ fric} + T_{latch\ fric})} - 1$$

The kick-off torque from the LRAs and the latch lock-up torque, which aid deployment at the beginning and end of deployment, respectively, were omitted to simplify the calculation of the minimum torque margin.

The wire harnesses were designed, fabricated, and owned by an outside company and, therefore, Lockheed Martin had little control over them. Luckily, we were able to obtain two flight-like development harnesses. In addition to testing the harnesses’ sensitivity to installation parameters and routing, a battery of torque tests were performed, at temperature and after subjecting the harnesses to multiple thermal cycles, to determine the aiding and/or parasitic torque caused by each harness. The resistive torque of the latch, which is held against the center fitting by two torsion springs, which provide approximately 89 N (20 lb) of force, was a simple friction calculation with the coefficient of friction assumed to be 0.3. Once we understood how the harnesses and latch performed, and given the torque output of the PWDA Hinges and the geometry of the four-bar, we were able to finalize the minimum torque output necessary from the strut's laminar spring. Maximum torque was simply governed by the maximum capability of the eddy current damper.

2.4.1. Spring Relaxation

Lessons learned from previous programs taught us that laminar springs tend to relax after being subjected to repeated thermal cycles while being held in a stressed configuration. A more extensive discussion of this issue with laminate springs is in [2]. Therefore, prior to specifying the number of laminates for the springs, a thermal relaxation test was conducted to determine the number of thermal cycles necessary to fully relax the laminates and to measure the subsequent torque that the springs provided post relaxation. Thermal cycles and torque tests were continued until the torque drop from one torque test to the next was less than 3%.

Fig. 11 shows the results of a spring relaxation test. The spring's initial torque hysteresis curve has higher torque than the curves for subsequent cycles up to about 9 cycles. The friction does not show any change. The laminar springs are thermally pre-conditioned for the flight units for the number of cycles determined for each configuration prior to installation. The reduced torque output post conditioning is used in the torque margin calculation.

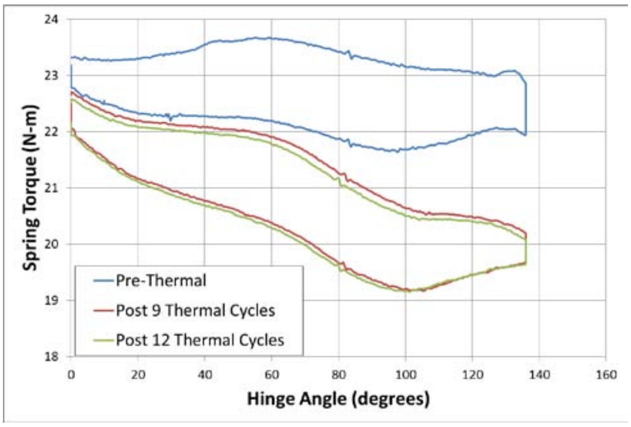

Figure 11: Strut Laminar Spring relaxation with exposure to repeated thermal cycles. Average drop in torque between cycle 9 and 12 was approximately 0.5%

3. INTEGRATION AND SYSTEM TEST

The PWDA mechanisms form the connection of the payload wing to the satellite, deploy it to a precise position and hold that position throughout the mission. The near-exact wing alignment and pointing requirements resulted in a complex integration and system test process. The process flow for integrating the PWDA components onto the spacecraft is illustrated in Fig. 12.

The hinges are first attached to the spacecraft and shimmed to set the hingeline position. The hinges are then removed from the spacecraft and attached to the wing, shimmed and aligned to each other. The wing is mated to the spacecraft and the strut is installed. The wing is then stowed to align the LRAs, and deployed again to tighten them down. The wing is then stowed and locked down for vehicle testing. Before and after acoustic testing, the wing is partially deployed for first motion tests, and the hinge friction is characterized. After thermal vacuum test the hingeline is characterized again, then the wing is stowed and LRAs preloaded for flight.

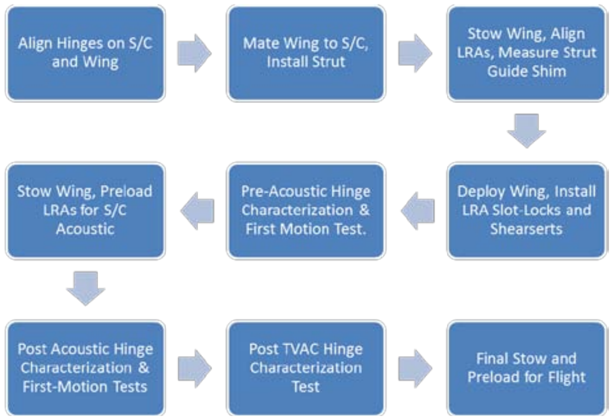

Figure 12: Vehicle Integration and Wing Deployment Testing

3.1. Hinge Installation and Alignment

Re-use of the Gravity Probe B hinge assembly provided a high strength and stiffness single-hinge system. However, the size of the AEHF payload meant putting two hinges together on a hingeline. This resulted in an over-constrained condition with two needle bearing hinges, requiring very tight alignment to each other and a complicated alignment process on the vehicle. The hinge integration and alignment process on the vehicle also needed to ensure the hinges were in position to satisfy the payload pointing requirements.

The hinges were first attached to the vehicle bridge panel and shimmed to set the rotation axis of the payload with respect to the spacecraft centerline. A laser tracker measured the hinge rotation and calculated the rotation axis direction and orientation of each hinge. The hinge positions were adjusted and tracked again until the rotation vectors of each hinge were aligned to each other, and together were aligned to the vehicle coordinate frame to within very tight translational and angular requirements. This process was quite tedious and took several iterations to complete.

The hinges were then removed from the vehicle and attached to the payload wing. The alignment process with the laser tracker was repeated, but this time the hinges were shimmed to set the hingeline rotation axis with respect to datum features on the payload wing. Once the hinge positions were established, the shearserts were removed one at a time and bonded in place for the flight installation. The payload wing, now with hinges installed, was lifted by crane onto support equipment and positioned to place the hinges at the locations shimmed previously. The wing was then mated to the vehicle and hinge shearserts bonded in place.

3.2. Strut Integration

The strut was installed next while the wing was deployed. The strut also used shims and shearserts to set the exact position, though the loads were not as critical as on the hinge. The strut and strut guide attachment point positions were measured by photogrammetry and were used to calculate the shim thickness under the inboard strut clevis. The strut was then installed in place (Fig. 13) and the required thickness of the outboard shim was measured directly.

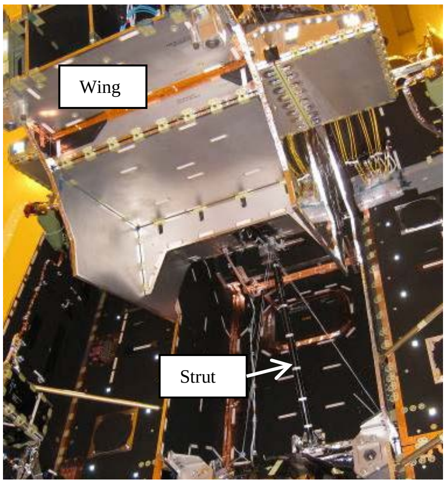

Figure 13: Underside of the Payload Wing, Partially Stowed with Strut Installed

3.3. Wing Offload and Stowing

The system to offload the payload wing was designed such that the strut was not overloaded at any time during the stow/deploy motion or when locked out in the fully deployed position. The ADAMS model of the offload system is illustrated in Fig. 14. Special crane equipment was used for precision control in the vertical direction. To stow the wing, a force was manually applied to the center of the strut to unlock it, or “break the knee”. The crane, set at its slowest speed, then lowered the wing a few degrees while maintaining the force on the strut hinge. This had to be done very carefully and deliberately because the mechanical advantage of the strut at this position is high enough to deploy the wing, dropping the offload to zero and crushing the locked-out strut under the weight of the wing. Once it was safe to remove the force on the strut, the crane slowly lowered the wing the rest of the way.

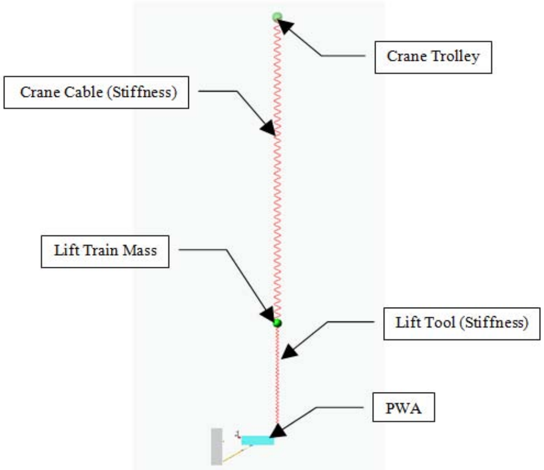

Figure 14: Model of the Payload Wing Offload System

The inboard and outboard strut clevises contain spherical bearings to ensure that binding does not occur during deployment. Unfortunately, they also give the system an extra degree of freedom. This extra degree of freedom has the effect of allowing the Strut Assembly to “flop” to one side or the other when not fully deployed. The amount of travel (or “flop”) is limited by the outboard strut clevis. When completely stowed, the Strut Assembly can travel approximately 3.8 cm (1.5 in) in either direction. This was a minor annoyance during component test of the Strut and PWDA, but was much more of a problem at vehicle level where there is no access or visual line of sight to the strut as it stows into the Strut Guide.

The solution was to have technicians on each side of the payload holding strings that were looped through the PWDA center fitting to guide the strut into the strut guide. A makeshift tool (Fig. 15) was quickly constructed that attached a camera to the end of a rod to be able to see the strut as it mated with the strut guide (Fig. 16).

Figure 15: Camera used to View the Stowed Strut Underneath the Payload Wing

Figure 16: Strut Fully Stowed in the Strut Guide as seen by the Camera

3.4. Launch Restraint Assembly Integration

Each LRA contains a cup/cone interface to react the shear loads during launch. With the wing in the stowed position, the cone (on the vehicle) was aligned to the cup (on the wing). The cone was then locked in place by transfer-punching and drilling removable plugs called “slot locks”.

The preload rods were installed into the SSRDs on the bench and then together were attached to the rear of the bracket on the spacecraft. Note that this work is done with the wing in the deployed configuration, where access was not a problem. However, preloading the LRAs with the payload wing stowed was quite a different story. While re-using the LRA from a previous program provided a proven release mechanism and a method for retracting the preloaded rod, access to the launch restraint was very poor, resulting in a preloading process that was risky, time consuming and required a high amount of skilled hands-on effort. Very few technicians were able to perform the preloading operation well, adding to the risk of human error. The technician had to lay on a “diving board” style of human lift, ~5 m (15 ft) off the ground, while it was driven up to the vehicle in close proximity to the payload components (Fig. 17). Once in position, he mostly could only use one hand to install and tighten all of the preload components. This tiring process took several hours for each LRA.

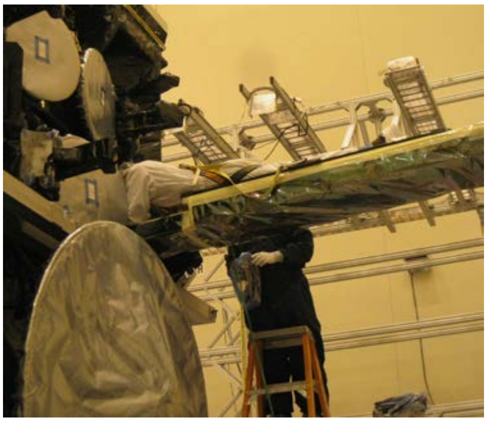

Figure 17: Technician Preloading an LRA

3.5. Hinge Characterization Test

A test method was desired to verify the PWDA functionality at the vehicle level. However, it was not possible to perform a full wing deployment torque in a 1G environment. Instead, the hinge characterization test was developed to measure the system hysteresis about the hingeline over the first few degrees of motion from the stowed position. This test ensured that there was no additional friction due to installation in the system, giving confidence that the wing would deploy properly.

A tool was designed to apply a lateral force to the wing at a specific location, pulling it from fully stowed to a few degrees and pushing it back to 0° (Fig. 18). The resulting force curve formed a hysteresis loop from which deployment torque and friction can be determined.

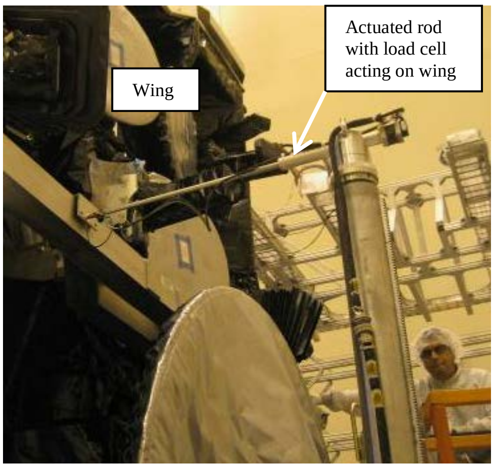

Figure 18: Hinge Characterization Test Setup

On one occasion, large hysteresis was observed after acoustic testing. The resulting investigation found an accelerometer wire attached to the side of the payload wing interfering with the motion of the wing. The test was repeated after removing the accelerometer and the friction “bumps” disappeared (Fig. 19).

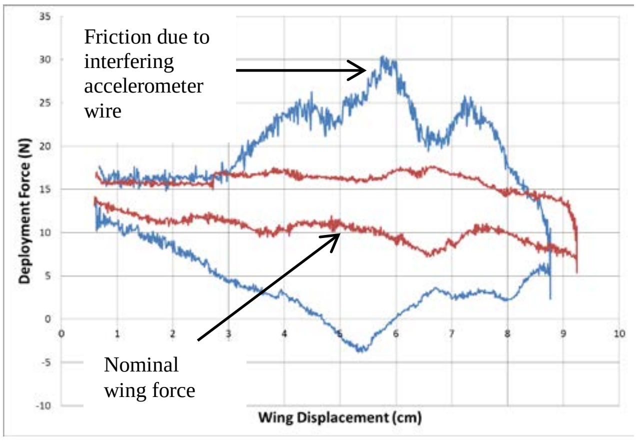

Figure 19: Measured Wing Force Hysteresis used to verify Hinge Torque and Friction

3.6. First Motion Deployment Test

The first motion test verifies that the LRAs separate and identifies gross losses in torque or changes to the deployment system due to acoustic testing. This is performed with the wing offloaded but the crane is motionless, so the wing will only deploy outwards less than 10°. The wing position was tracked and measured using a theodolite optical tracking system. The maximum transient angle and the final steady-state angle are measured and compared before and after acoustic and to the family of data for all other wing tests. Pre/Post acoustic first motion test results for one wing are shown in Fig. 20. One or two degree change between tests was not considered significant.

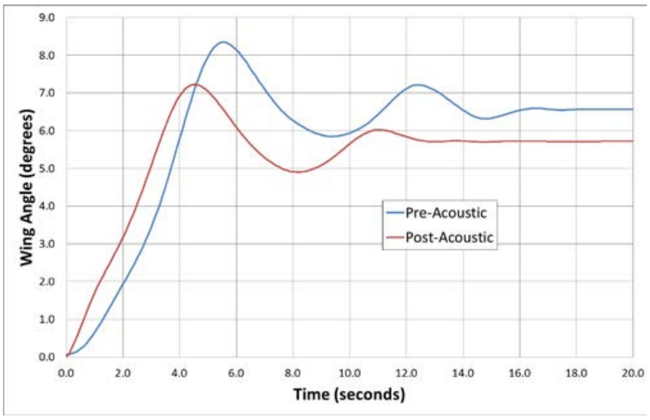

Figure 20: First Motion Test Results

3.7. Final Stow for Flight

After all vehicle environmental tests and the last hinge characterization test are performed, the wings are stowed and LRAs preloaded for flight. After the satellite is transported to the launch base at Cape Canaveral (Fig. 21), a final inspection is performed to verify the strut is verified to be seated properly in the strut guide, the springs are sitting nominally and free to rotate, all shearsert bondlines are intact, and all seams and interfaces are inspected to be free of any obstructions or blanket interferences that could hang the deployment.

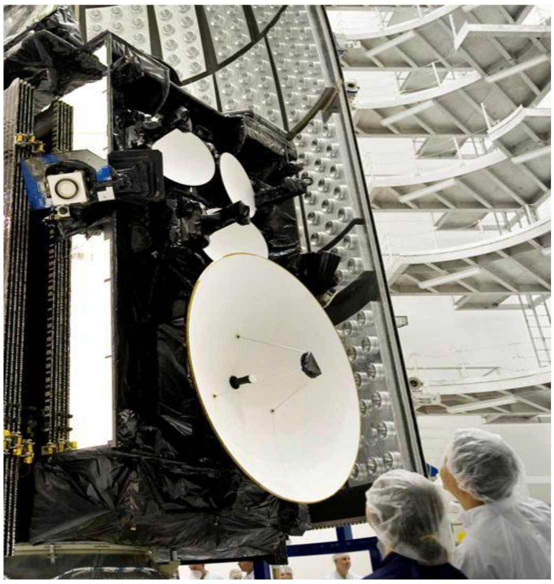

Figure 21: Payload Wing installed on the Vehicle and Encapsulated in the Rocket Fairing.

4. CONCLUSION

The Payload Wing Deployment Assembly began as a combination of mechanisms with proven flight heritage with little design modifications. However, development of the system uncovered several shortcomings that required extensive re-design efforts to meet the stringent requirements for the AEHF payload. Integration of the mechanisms also proved to be difficult due to characteristics of the heritage designs that were not optimized for the integration process. These difficulties were overcome, and resulted in a robust system for precision deployment of the heavy AEHF payload. The PWDA has performed very well to date, with all four payload wings on the first two AEHF satellites deployed successfully on orbit.

5. REFERENCES

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