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A Flight Prediction for Performance of the SWAS Solar Array Deployment Mechanism

Gary Sneiderman* and Walter K. Daniel**

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Abstract

The focus of this paper is a comparison of ground-based solar array deployment tests with the on-orbit deployment. The discussion includes a summary of the mechanisms involved and the correlation of a dynamics model with ground based test results. Some of the unique characteristics of the mechanisms are explained through the analysis of force and angle data acquired from the test deployments. The correlated dynamics model is then used to predict the performance of the system in its flight application.

Background

The Submillimeter Wave Astronomy Satellite (SWAS), one of NASA's Small Explorer missions, is designed to study the chemical composition of interstellar gas clouds which are invisible from beneath Earth's atmosphere. Its primary objective is to survey for water, molecular oxygen, carbon, and isotopic carbon monoxide emissions in a variety of galactic star forming regions. The 284 kg SWAS spacecraft is a three-axis-stabilized, stellar observatory with a pointing accuracy of 38 arcsec and jitter less than 19 arcsec. The spacecraft typically points the instrument at 3-5 targets each orbit.

SWAS was developed in the early 1990's for a launch in June of 1995. However, due to launch vehicle delays, the satellite remained in storage until mid-1998 when it was re-tested and successfully launched on December 5th 1998.

The SWAS power system includes four rigid deployed solar panels and one body-mounted panel. The 3.4 m^2 of solar cells provide 230 W of orbital average power for the spacecraft and instrument. The two requirements that were central to the design of the solar array were total area and natural frequency. The SWAS power consumption required rather large deployable solar arrays for a small spacecraft (see Figure 1). To prevent interaction between the attitude control subsystem and structural vibration of the arrays, the solar array natural frequencies had to be above 1.0 Hz.

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*NASA Goddard Space Flight Center, Greenbelt, MD
**Mechanical Dynamics, Inc., Austin, TX

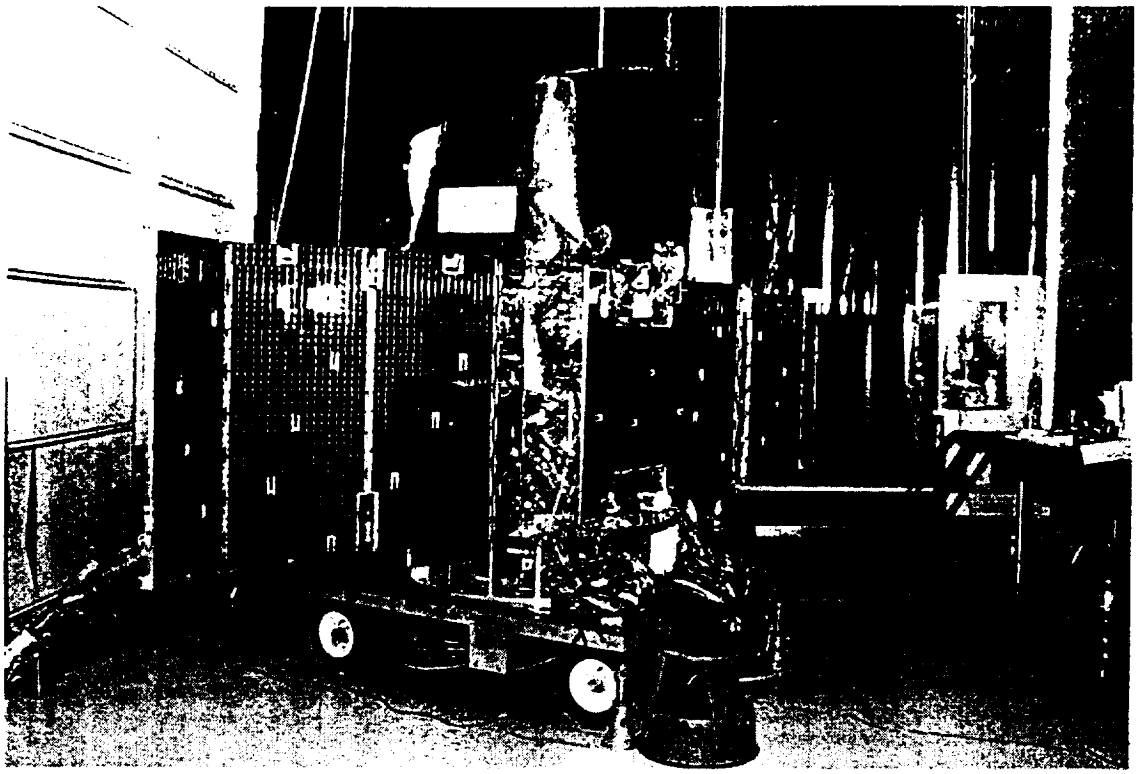
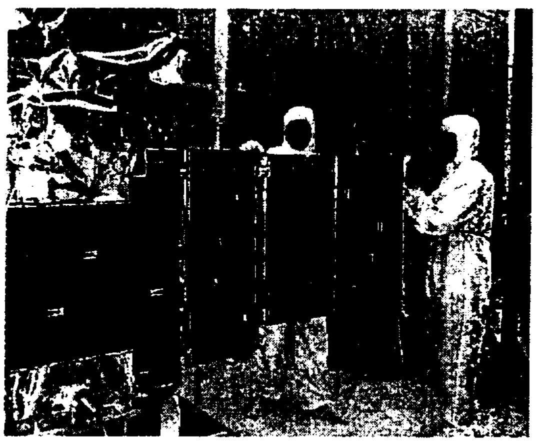


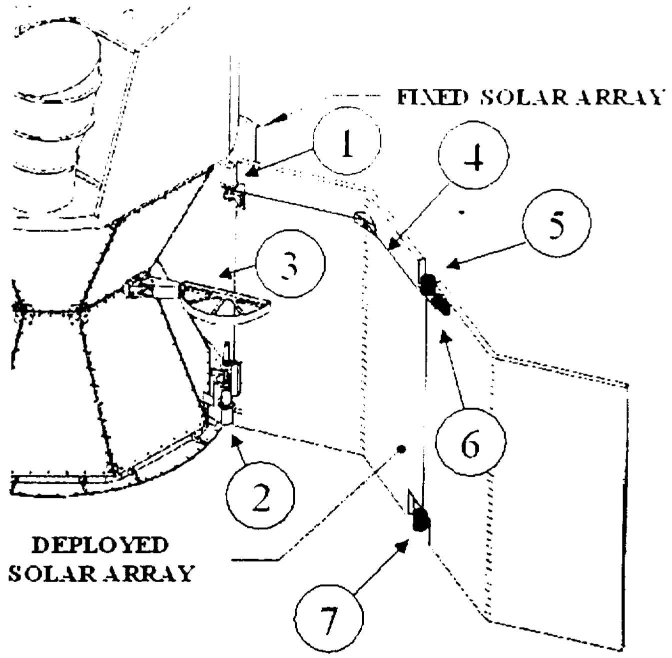
Figure 1. The Submillimeter Wave Astronomy Satellite

Array and Mechanism Design

The primary structure and the component panels of the spacecraft form an octagonal hourglass shape. This geometry accommodates a "wagon wheel" locking mechanism used to stiffen the array once it is deployed.

The substrate of the inner and outer panel of each deployed array is made with a continuous honeycomb core and aluminum facesheets with a 135 deg included angle at its center (see Figure 2). When deployed, each solar array extends 1.5 meter from the spacecraft body. The inner panel is cantilevered from the spacecraft structure by two non-locking hinges.





1 - Structure to s/a upper hinge and cable wrap
2 - Structure to s/a lower hinge and deployment mechanism
3 - Structure to s/a lock and bracket
4 - Cable
5 - Panel to panel upper hinge and lock
6 - Cable tensioning spring
7 - Panel to panel lower hinge and lock

Figure 2. Deployed Solar Array

The spring loaded wagon wheel locking mechanism prevents rotation about the hinge line after the array has opened. This helped the deployed array achieve the stiffness goal of 2.0 Hz as the first natural frequency. The outer panel is connected to the inner panel with a pair of locking hinges and pulled open by a cable. The upper panel-to-panel hinge is shown in Figure 3 with the housing for the spring-tensioned cable where it connects to the outer hinge. This lightweight aspect of the design significantly reduced the mass moments of inertia of the deployed panels. As the inner hinge opens, the cable wraps around a mandril and pulls the outer panel open. The cable and tensioning springs are tuned so the outer hinge locks before the inner panel rotates to its final position.

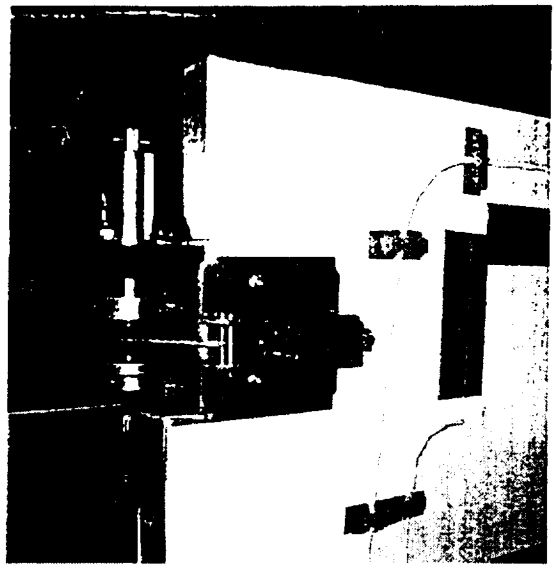


Figure 3. Spring Tensioned Cable around Upper Panel-to-Panel Locking Hinge

In the stowed position, the arrays are folded over each other and preloaded against the spacecraft body at two points with paraffin actuated release mechanisms. The release of the latch mechanism imparts very little or no momentum to the panels, so they essentially remain in the stowed position until deployed. Each array is actively forced open with a single paraffin actuator through the deployment mechanism (Figure 4). The paraffin actuator pushes a rod through a barrel cam that converts linear motion to the rotational motion used to open the panels. A force sensor located between the actuator pin and the cam shaft provides crucial diagnostic information used to assess the performance of the mechanism throughout the active deployment. The actuators are also equipped with thermistors mounted to the external case to measure temperatures throughout the deployment sequence. The SWAS paraffin actuators have Positive Temperature Coefficient (PTC) thermistors wired in series with each of the two internal heaters. Because the electrical resistance of these solid state thermistors changes sharply once they reach a specific temperature they can be used to power down the actuator after they have completed their function.

The –X array displays similar though not identical motion to the +X array (see Figure 6). There was always some variation in the forces due to the slight difference in tolerance build-up, inexact balancing of the spacecraft dolly, and friction effects. Note that the first motion occurred at 15 lbf on the +X array and 40 lbf on the –X array. This difference in the force required for first motion was observed consistently throughout the test program.

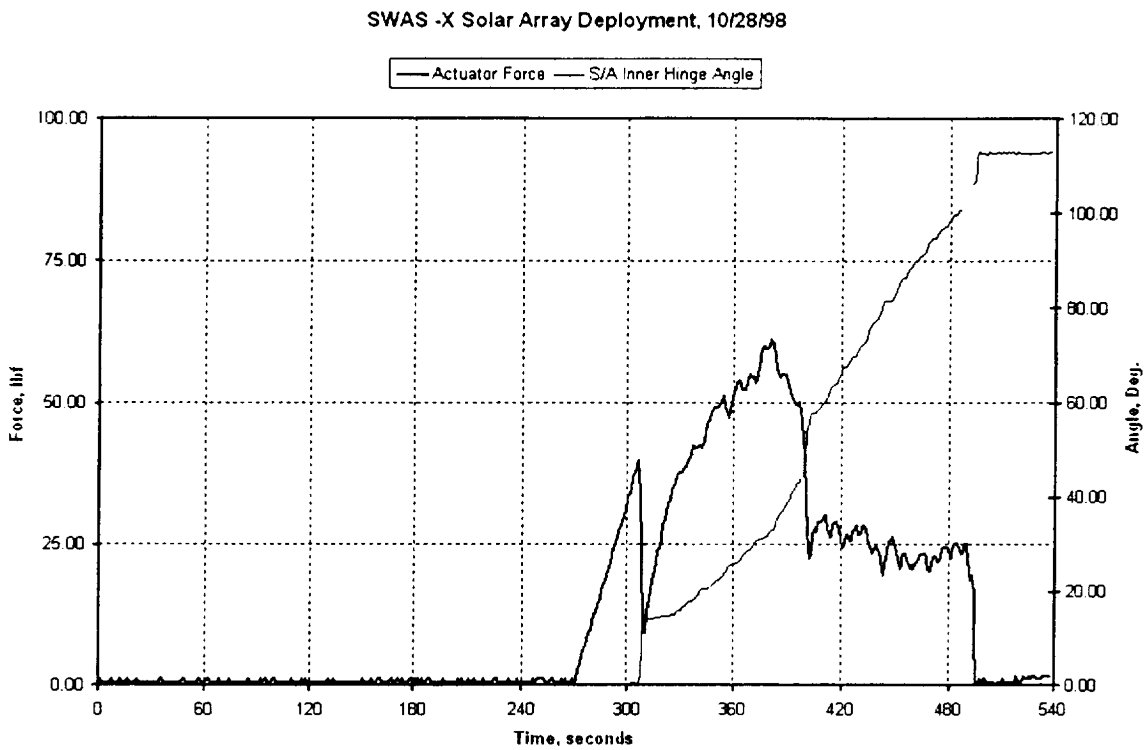


Figure 6. Ground Test Data for –X Array

One aspect of the motion that is difficult to see with these plots is the "stuttering" of the outer panel relative to the inner panel. As the paraffin actuator drives the inner hinge open the cable tensioning spring is repeatedly compressed then released. The force builds until it overcomes stiction in the hinges and the outer panel opens a few more degrees releasing the energy in the spring. In this fashion, the actuator only needs to match the force resisting the deployment of the panels. Unlike stored energy deployment systems, the force margin is unused. The result is a slower deployment, with minimal shock and impact forces at the end of the deployment.

After the array is completely deployed, the actuator piston must reach a hard stop to prevent the squeeze boot from inverting and damaging the actuator. Once the actuator reaches this stop, pressure continues to increase until the internal temperature is sufficient for the PTC thermistors to change state and reduce the power draw from the

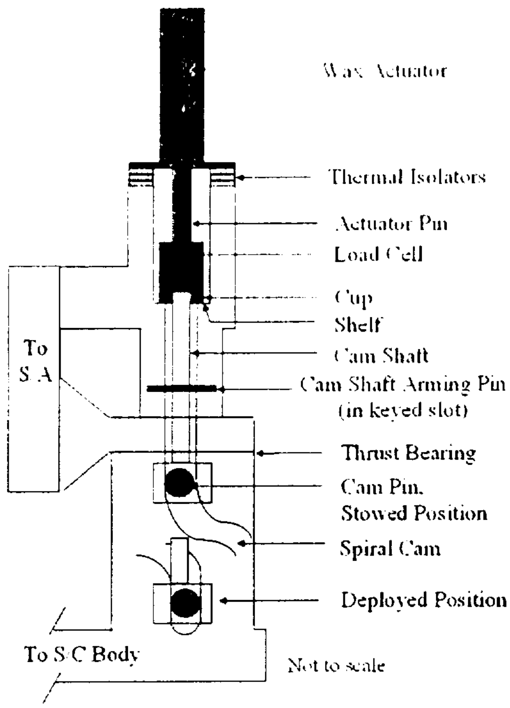


Figure 4. Schematic of the Deployment Mechanism

Operations

Since the arrays are deployed "in the blind" before the first telemetry pass of the satellite over a ground station, it is important to sequence the deployment properly and to minimize the total power drain from the battery. Immediately upon spacecraft separation from the third stage rocket motor the four release mechanism actuators are powered on. Since there is no means to directly measure of the function of the release mechanisms, the paraffin actuator case temperature is used to determine when it is safe to start deploying the arrays. The logic requires the release actuators to exceed 78°C before initiating an eighteen minute hold. This temperature was determined from extensive thermal-vacuum testing of the release mechanisms. The time delay adds a safety buffer against the possibility of deploying the arrays too soon and risking damage to the deployment paraffin actuator. Both the release and deploy commands can be initiated by hardware timers as discussed above, by an on-board software timer, or by ground command if the others fail.

Ground Testing

The solar panels and hinges were designed with sufficient strength and stiffness to be deployed in a one-g environment to simplify the ground testing. The paraffin actuator can be reused hundreds of times, so numerous deployments were performed throughout integration and testing in preparation for the launch. The arrays are always opened in clean rooms with the spacecraft carefully balanced against gravity. Because the paraffin actuator moves so slowly, the array never builds significant momentum. Therefore, the force applied by the paraffin actuator, and measured by the force sensor, was exactly equal to the system friction plus the resisting force of the springs.

The arrays were deployed at the NASA Goddard Space Flight Center on 28 October 1998 in the final ground test before shipping to the launch site. One more deployment test at the launch site would confirm that everything was flight ready. For the ground tests, the load cell and inner hinge potentiometer readings were recorded at a high rate (1 Hz sampling) and the tests were videotaped.

The curves in Figure 5 depict the measurements of the +X array motion, force (thick line) and inner hinge angle (thin line). The actuators require approximately four minutes to heat from room temperature to the paraffin melting temperature. Once the paraffin melts and starts expanding the force builds until the stiction and resisting spring forces are overcome and first motion occurs; note the jump in hinge angle and corresponding drop in force. The force then rises steadily to a peak of approximately 60 lbf as the cable reaches the tangency point with its guide on the outer hinge. As the outer hinge locks open the force drops to about 25 lbf. With the outer panel locked open, the two panels rotate together for the remainder of the deployment. Note the cycling effect of the tensioning spring loading and unloading at this time. At 110 degrees of rotation, the wagon wheel lock slides into position and the deployment is finished at 113 degrees.

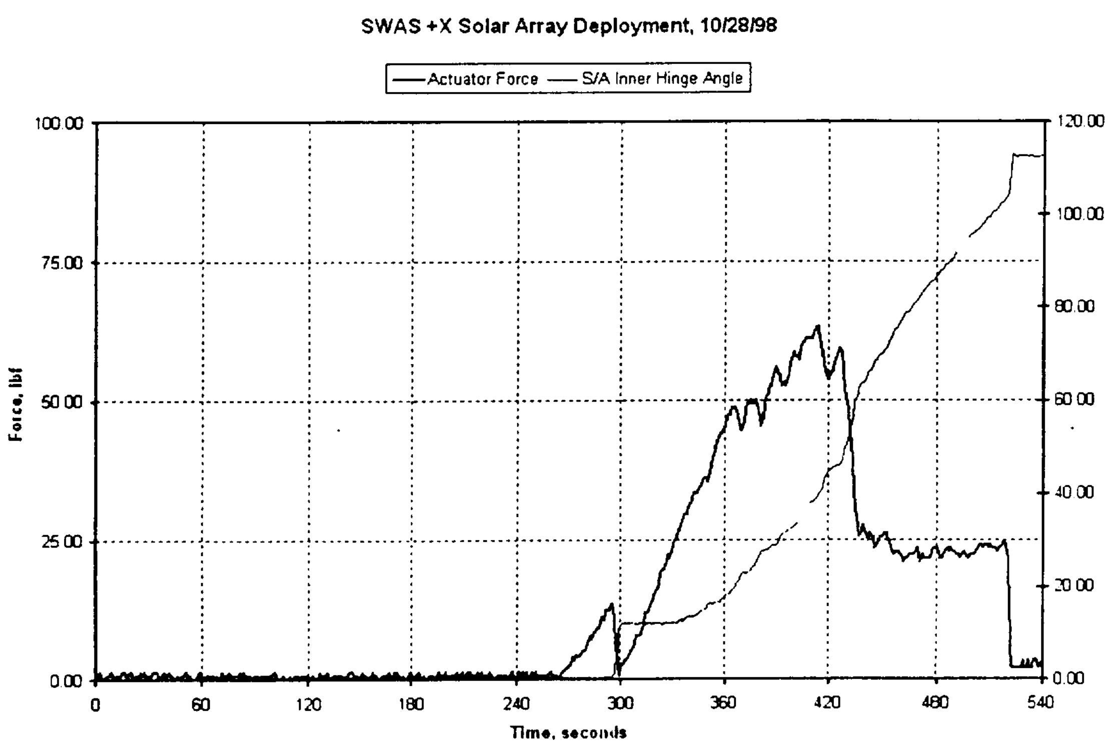


Figure 5. Ground Test Data for +X Array

battery. The actuator then begins to cool and the measured force drops. Figure 7 shows the result of the two thermistors switching off power by the two changes in slope near the peak of the force curve.

Note that the deployment is finished at approximately 500 seconds. A conservative indication of the force margin can be inferred by comparing the peak of the force at full extension with that required to open the array. This is conservative, because the actuator actually can exert a larger force when it is at less than full extension.

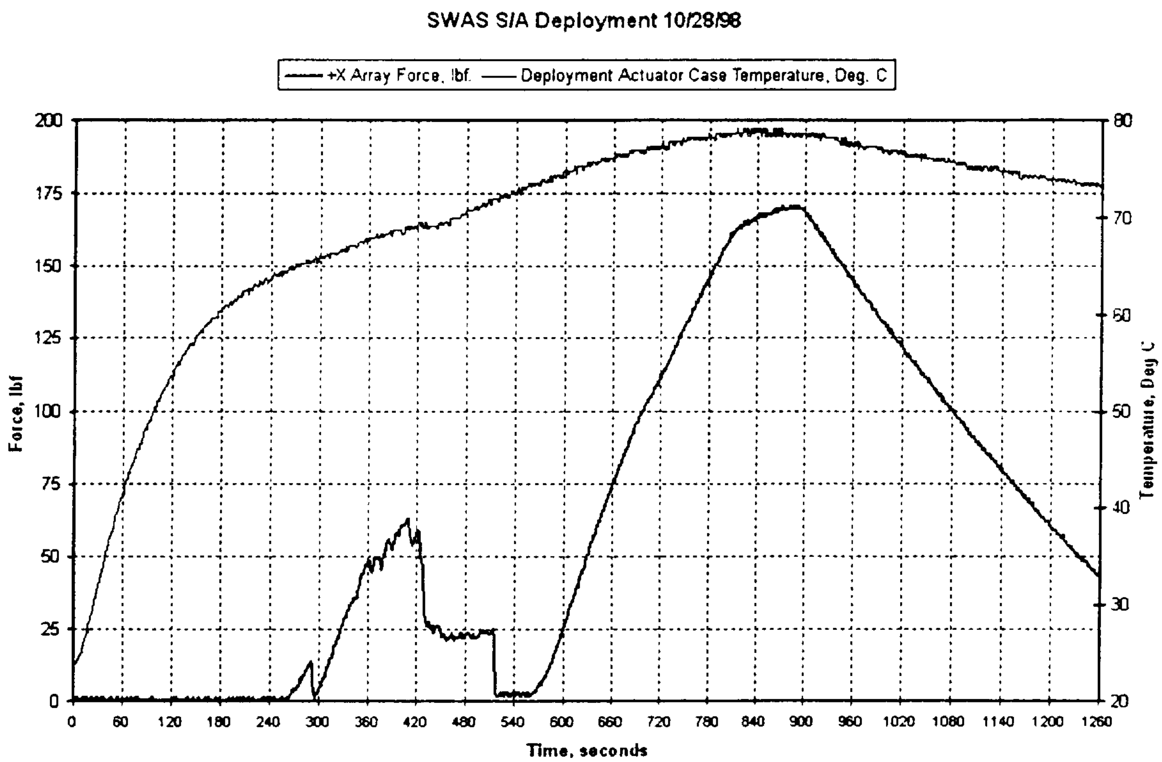


Figure 7. PTC Cutoff is Indicated by Change in Slope of Curves

Test Anomaly

Following the final deployment at the launch site, NASA personnel noticed that the peak force at which the PTC thermistor switched off had decreased. This change was only observed on one of the two deployment actuators, the –X one. Figure 8 shows this feature on the –X array while Figure 9 shows that the change did not occur on the +X array. Several tests were necessary to isolate the problem to the actuator. The spare actuator was integrated and two additional deployment tests were run to confirm that the spare was operating properly. The spare was flown on the mission. At the time of writing, a final disposition on the cause of the degrading performance had not been completed.

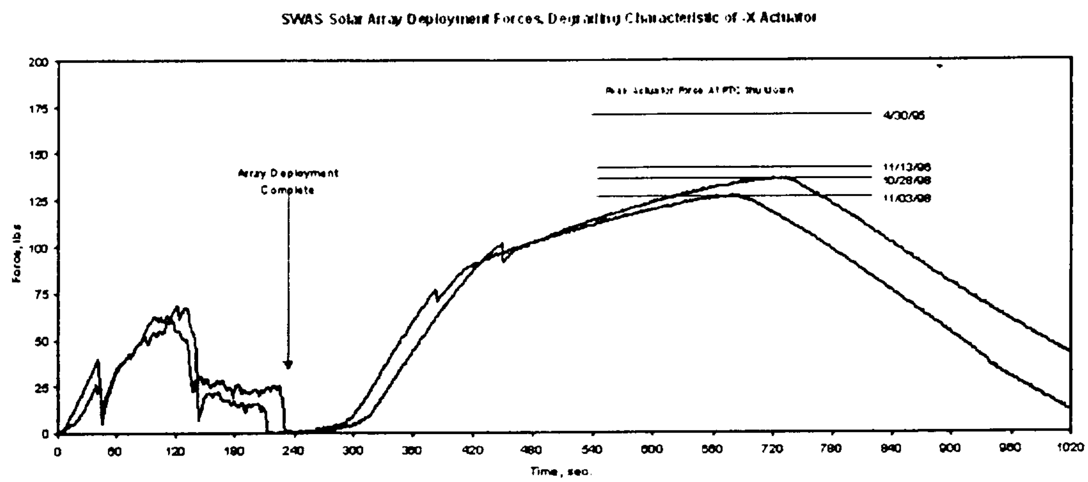


Figure 8. Degrading Performance of –X Deployment Actuator

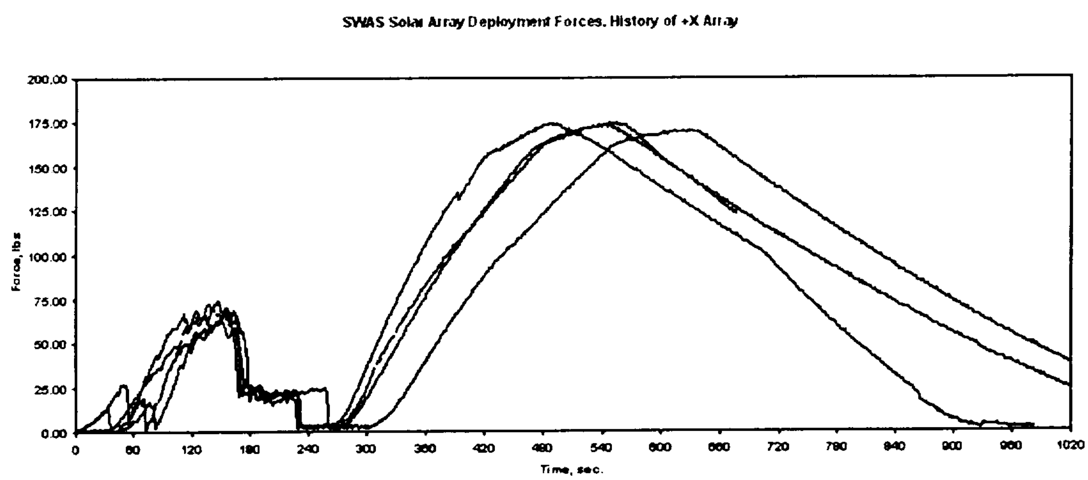


Figure 9. Nominal Performance of +X Deployment Actuator

Analytical Model

A model of the solar array deployment mechanism was developed using ADAMS (Automatic Dynamic Analysis of Mechanical Systems), a commercial, off-the-shelf dynamics analysis program from Mechanical Dynamics, Inc. [Ref. 1] The goal was to validate the model against ground testing results then use the model to predict flight performance.

be minimal. Instead, it appears that the tensioning springs dominated the force required to deploy the arrays. The fact that a slightly higher force was required in orbit is probably due to the difference in ambient temperature and lack of air as a lubricant in the mechanisms moving parts.

Ground testing remains of vital importance with the set of complex and often unknown flight-critical parameters that can effect flight-critical mechanisms. The interaction of friction and other higher-order effects can be difficult to model analytically but test data makes the job much easier. Computer models of solar array mechanisms can be used both for flight performance predictions and to gain insight into the system motion. One lesson learned from this project is to involve analysts in the test program. An engineer has much better access to key parameters such as spring rates and hinge radii when the hardware is in the lab.

References

1. *Using ADAMS/View*. Mechanical Dynamics, Inc., 1998.

2. Avallone, Eugene A. and Theodore Baumeister III. *Marks' Standard Handbook for Mechanical Engineers, Tenth Edition*. McGraw-Hill, 1996, page 3-23.

3. Smith, Bruce A. "SWAS Launched on Pegasus XL." *Aviation Week and Space Technology* (December 14, 1998), page 33.

There are three bodies in the model: the spacecraft body (fixed on ground), the inner panel, and the outer panel (see Figure 10). The inner panel is attached to the body with a revolute joint that represents the inner hinge. This first joint is driven directly by the deployment mechanism and is intended to lock at 113 deg although no latch is included in the model. The outer panel is connected to the inner panel with another revolute joint to represent the outer hinge. The second joint is opened by the tensioned cable and latches at 180 deg.

The paraffin actuator is represented by an applied motion with a value of 0.127 mm per second. This rate is derived from stroke of the actuator (25.4 mm or 1.0 inch) that takes approximately 200 seconds once powered. This actuator cannot apply an overly large force so a spring was used to represent mechanical slack in the system. The spring constant value was large (50,000 N/m) and selected to correlate with ground testing data. The spring pushes on a rod that is constrained by a translational joint that represents the input to the helical cam. An ADAMS coupler joint converts the linear motion of the rod to a rotational motion of the inner hinge to simulate the cam function. The coupler rate is 12.88 mm per radian to connect the 25.4 mm (1.0 inch) of actuator translation to the 113 deg (1.972 rad) of inner hinge rotation.

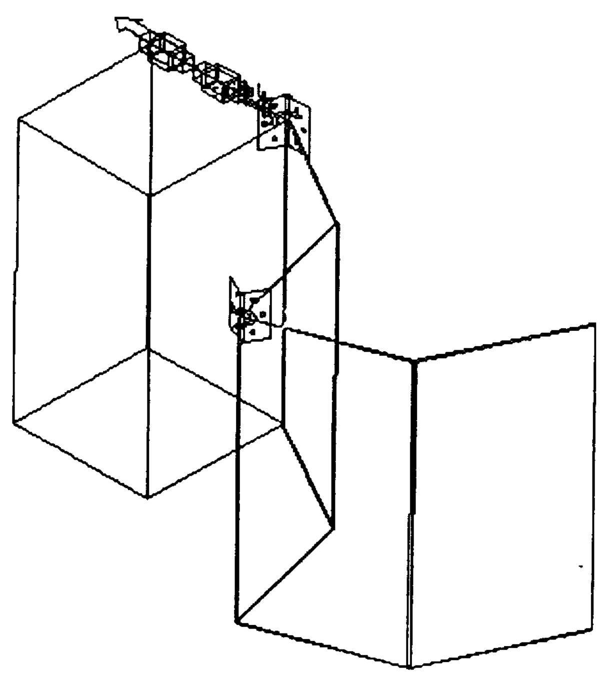


Figure 10. ADAMS Model of SWAS Solar Array

The cable opens the outer panel by applying torque at the outer hinge as the actuator drives the inner panel. A torque function was used to model the effect of the spring-tensioned cable. Apart from the initial tension, cable force is caused by stretching due to a difference in hinge angles. The displacement of each end is the product of the mandril radius and the hinge angle. The resulting force is the difference in displacements multiplied by the spring constant of the spring that tensions the cable:

$$F_{cable} = k(R_{outer}\theta_{outer} - R_{inner}\theta_{inner}) + F_{preload}$$

The spring constant k was measured to be approximately 6994 Newton per meter (i.e., 10 lbf to compress 0.25 inch). The cable force pulls on both ends—the spacecraft body and the outer hinge. The small preload force was neglected in the torque calculations. The torque on the outer hinge is therefore the spring force times the radius of the hinge:

$$M_{outer} = F_{cable}R_{outer} = kR_{outer}(R_{outer}\theta_{outer} - R_{inner}\theta_{inner})$$

ADAMS real-time measures of the hinge angles were created for use in the torque function as well as for plotting after each simulation run. Note that R_{outer}, the effective moment arm of the cable force, was treated as a constant to simplify the model. The actual moment arm varies as the cable lifts out of guide during deployment.

Friction in the outer hinge was needed for the model to accurately recreate the motion of the outer panel. The standard ADAMS revolute joint friction torque was used with a coefficient of friction of 0.04 to represent the Teflon-coated hardware in the joints [Ref. 2]. This friction torque includes stiction and axial preload effects.

Analysis of Ground Testing

Initial runs of the simulation were made of the ground test setup in order to finalize parameter selection. As shown in Figure 11 the model reasonably matches ground test results. The inner hinge opens steadily towards the 113 degree design position. The outer hinge does not start moving until the inner hinge has reached approximately 10 deg as designed. Note that the outer hinge motion exhibits the start-and-stop behavior seen in ground testing. This jerkiness is due to stiction in the outer hinge. When the inner panel has opened a sufficient amount, the torque due to the cable overcomes stiction and outer panel moves. When the outer panel has moved to the point that cable tension is relieved the stiction in the hinge stops the motion until the next cycle. Once the outer hinge has locked at 180 deg the inner hinge opens smoothly.

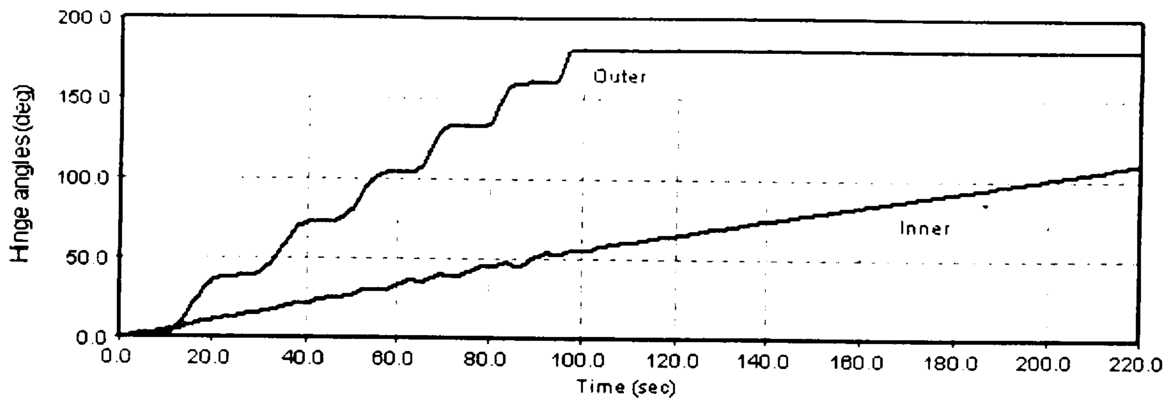


Figure 11. Hinge Angles for Ground Testing Model

Flight Prediction

For the prediction of flight performance of the mechanism a simulation run was made with no gravity. As shown in Figure 12, the results are not very different from the ground test run. The inner hinge motion is essentially the same while the outer hinge begins to open sooner than in the ground testing case. The start-and-stop motion of the outer panel is not as pronounced and the steps occur over slightly smaller angles. In general terms these results make sense because joint friction due to weight is no longer present. In addition, the solar array deployment mechanism was designed to be very robust so the absence of gravity does not cause a major change in behavior.

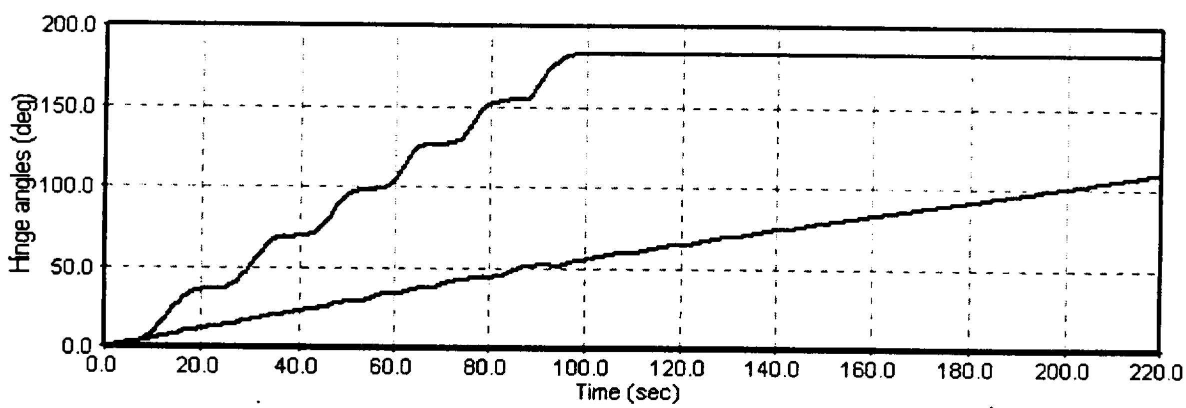


Figure 12. Hinge Angles for Flight Prediction Model

Flight Results

The SWAS spacecraft was successfully launched on a Pegasus XL vehicle on December 5, 1998 [Ref. 3]. Separation of the booster and deployment of the solar arrays were confirmed in telemetry. Figures 13 (actuator force) and 14 (inner hinge angle) show the flight performance of the +X array. In Figures 13 through 16, the thick lines represent the flight data while the thin lines represent the data from the final two ground tests. Note that the flight data was sampled at 0.2 Hz whereas the test data was sampled at 1.0 Hz.

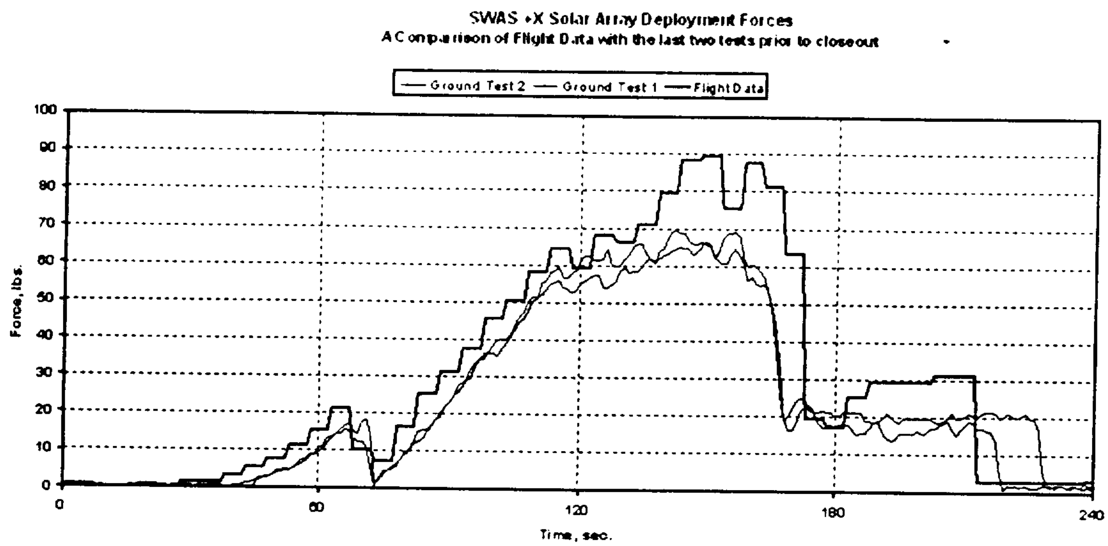


Figure 13. Flight Data for +X Solar Array Deployment Force

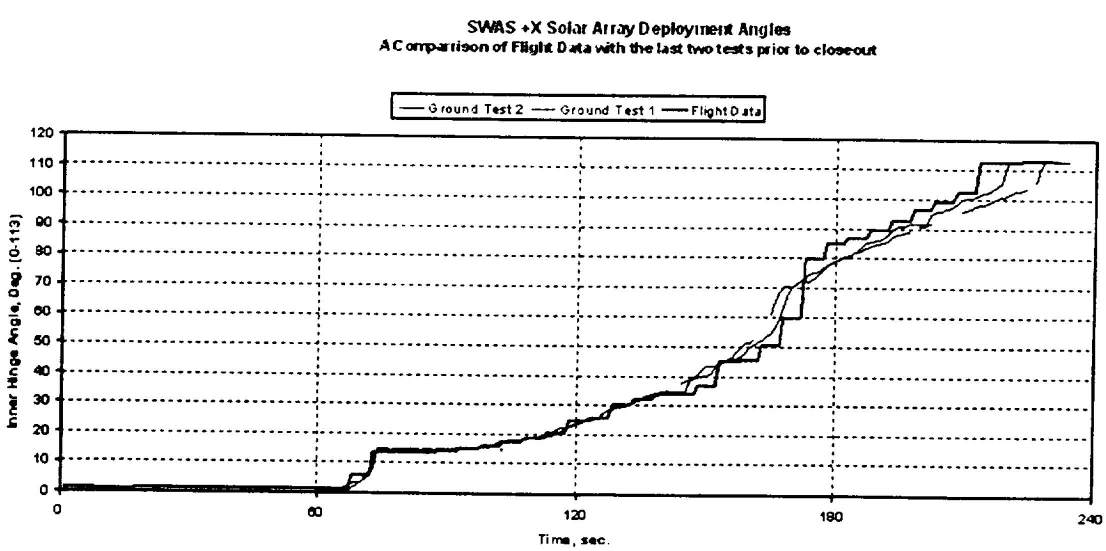


Figure 14. Flight Data for +X Solar Array Inner Hinge Angle

Figures 15 (actuator force) and 16 (inner hinge angle) show the flight performance of the -X array. Note that the peak force was approximately 90 lbf, a similar value to that for the +X array.

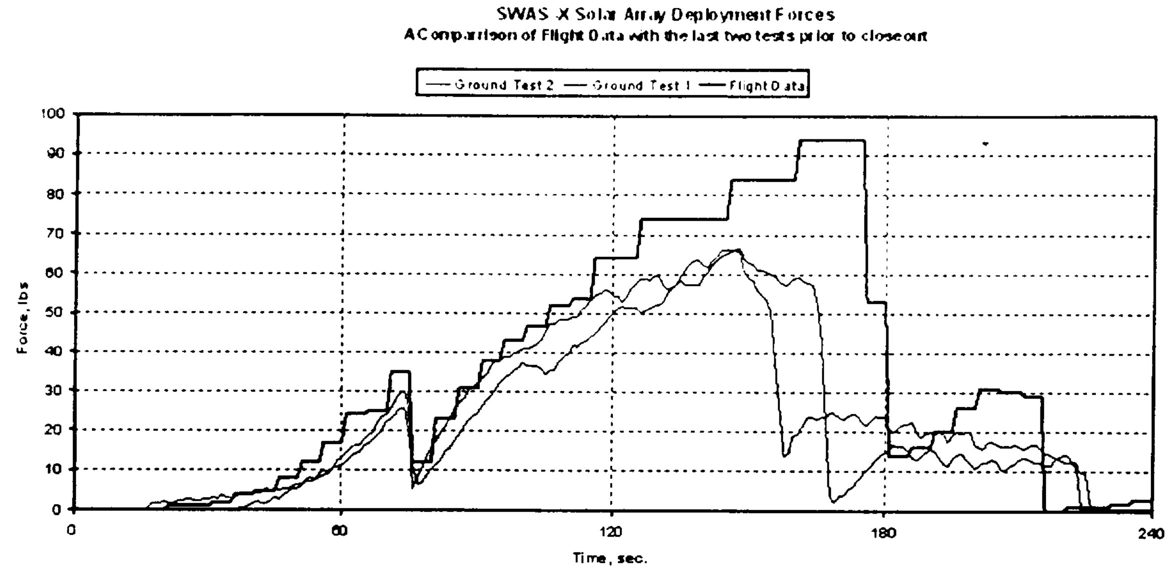


Figure 15. Flight Data for -X Solar Array Deployment Force

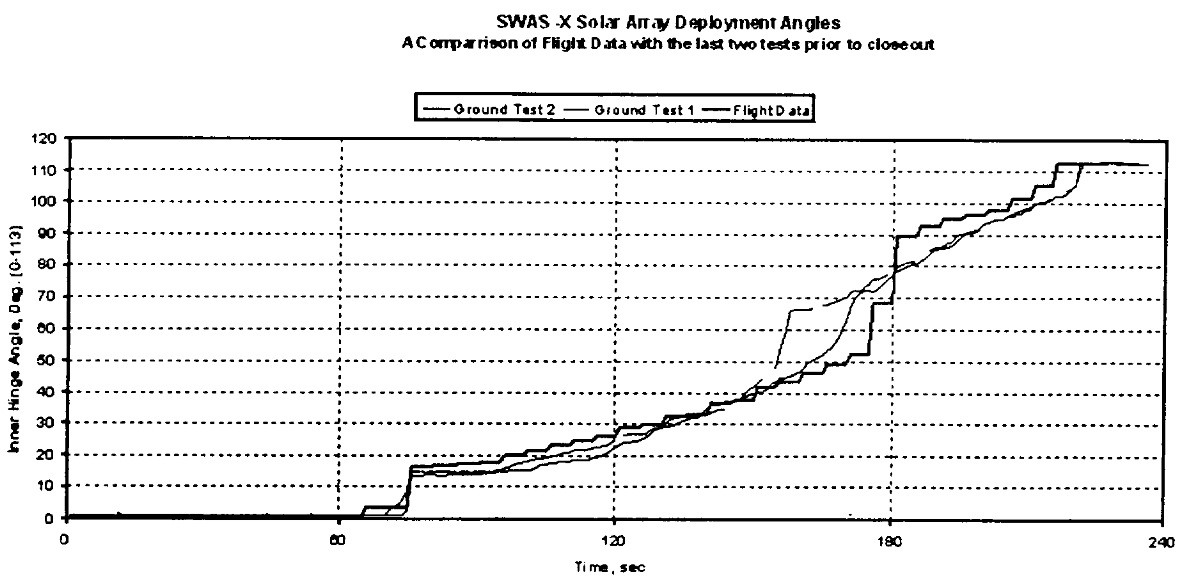


Figure 16. Flight Data for -X Solar Array Inner Hinge Angle

Conclusions

While the flight data was not what we had personally expected, it confirmed that the SWAS ground testing environment was remarkably similar to the orbital environment. The anticipated effect of gravity on increased moment, and hinge friction turned out to