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SPACE STATION FREEDOM SOLAR ARRAY CONTAINMENT BOX MECHANISMS

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Introduction

Space Station *Freedom* will feature six large solar arrays, called solar array wings, built by Lockheed Missiles & Space Company under contract to Rockwell International, Rocketdyne Division. Solar cells are mounted on flexible substrate panels which are hinged together to form a "blanket." Each wing is comprised of two blankets supported by a central mast, producing approximately 32 kW of power at beginning-of-life. During launch, the blankets are fan-folded and compressed to 1.5% of their deployed length into containment boxes (*figure 1*). This paper describes the main containment box mechanisms designed to protect, deploy, and retract the solar array blankets: the latch, blanket restraint, tension, and guidewire mechanisms.

Design Heritage

SAFE

The technologies and mechanisms used on the Space Station *Freedom* (SSF) wing were first demonstrated in 1984 on the Solar Array Flight Experiment (SAFE) aboard Shuttle mission STS-41D (*figure 2*). However, different requirements for SSF led to major differences in the implementations of the latch and blanket tensioning mechanisms, as well as the addition of a blanket restraint system. SAFE's smaller, single blanket design was latched and preloaded in a single containment box using cams and the initial motion of the extendable mast. In contrast, the two containment boxes of the much larger two-blanket SSF design (*figure 3*) were required to swing 90° into a more compact configuration for stowage aboard the Orbiter. The 90° rotation of the two boxes necessitated an all new design for the latch mechanism (see below). The smaller wing and very short operational life of SAFE allowed its tension mechanisms to be weight-optimized for low load at high stress, without concern for thermal cycles and related mechanism fatigue. Increased tension and life requirements for SSF, as well as limitations in the partially deployed mast capability, caused major redesign of the tension mechanism. In contrast, SSF's guidewire mechanisms are direct descendants of SAFE. Both designs use constant-force spring driven takeup drums to deploy and retract over 30 m (>100 ft) of wire rope. This cable passes through every other blanket hinge, guiding the fanfolding blanket during deployment and retraction. Finally, a new blanket restraint system was designed to accommodate the weight and size of the SSF blankets.

Milstar

Though a later design, Milstar's mechanisms have less in common with SSF than do SAFE's. The primary reason for this is that Milstar has no requirement to retract its wing on-orbit. Its latch mechanism is preloaded on the ground and released by pyrotechnic pinpullers. The guidewire tension is required only during initial deployment, so may be provided by a small slip-clutch. Though the Milstar tension mechanism is similar to the SSF design in providing nominally constant force over a wide range of thermally induced blanket and mast motion, it was sized for only a fraction of the tension required by SSF.

Special Requirements

In addition to the typical requirements for spacecraft mechanism design which include vacuum, temperature extremes, zero gravity, light weight, and remote operation, the Space Station *Freedom* program dictated several unique requirements for the solar arrays that significantly impacted the design of the containment box mechanisms. The most onerous of these requirements was that for repeated deployments and retractions: 35 extension/retraction cycles and 15 unlatch/latch cycles over the operational life of the wing. This requirement resulted from a system level desire to retract the arrays to allow on-orbit servicing of the remainder of the electrical power system and to avoid excessive wing loads that potentially result from the plumes emitted by the Orbiter's thrusters impinging on a deployed wing. Not only did this requirement preclude the use of single action release devices from being used on the containment box, but it also necessitated the ability to passively restow and align 33 m (107 ft) of solar array blankets and the related tensioning hardware within the containment box to sufficient accuracy to allow relatching without damaging the solar array.

A second category of unique requirements were those necessary to allow assembly and servicing on-orbit by astronauts during Extra-Vehicular Activity (EVA). The two most significant items in this category were requirements for manual backup capability to the automated mechanisms and the ability to remove and replace an individual containment box on-orbit. In addition to necessitating additional mechanization for the EVA to bypass the automatic mode and manually actuate the latch, these requirements necessitated separable interfaces and consideration in the mechanism designs of EVA limitations and risks.

The final category of special requirements was the severe design life which included a four year storage requirement, a one year dormant condition on orbit in the stowed configuration, and a 15 year operational life in low earth orbit (LEO) with a significant Atomic Oxygen (AO) flux. The space station orbit required the mechanisms to withstand 87,000 thermal cycles during this exceptionally long life. Finally, the long life in the specified AO environment of LEO provided very severe constraints on the use of lubricants and non-metals.

Mechanism Descriptions

Latch Mechanism

Function & Requirements

When stowed for launch, the folded blanket is preloaded within the containment box. This prevents "chatter" between the blanket panels during the vibratory/acoustic loading of ascent, as well as providing some measure of lateral restraint by inter-panel friction. The SSF latch mechanism is required to provide 24.9 ± 1.8 kN (5600 ± 400 lb) of preload (*figure 3*), distributed over eight locations on the containment box, using available motor output with 100% torque margin and a maximum of 20 seconds. It must capture and release the box cover anywhere from 0–9 cm (0–3.5 in) above the nominal compressed blanket stack height and be capable of 15 operations over a 15 year on-orbit life. It also must provide actuation force for the blanket restraint system and tension mechanisms.

Physical Description & Performance

To evenly distribute the preload into 17.2 kPa (2.5 psi) over the stowed blanket, there are eight latch points on the perimeter of the containment box, four per side (*figure 4*), and foam pads between the box and blanket. The motor drive assembly (*MDA*) is located at the inboard end of the box to minimize wire harness length and cantilevered mass. Its minimum output is 12 N-m (110 in-lb) at 180 RPM. This torque is transmitted by a *drive shaft* to tandem, opposing *ball screws* in the center of the box (*figure 5a*). The ball screws are lightly lubricated with a Braycote 600 grease plate, protected from AO by the box structure. Their support bearings are treated with a sputtered MoS_2 dry film solid lubricant. Small radial bearings support the extreme ends of the screws, while larger face-to-face mounted angular contact bearing pairs support the thrust loads (11.6 kN, or 2600 lb max). The thrust loads are reacted out locally by a common central bearing housing so that little load is transferred to the honeycomb panel mounting surface. *Ball nut flanges* on the ball screws are driven toward the center of the box during a latch operation. A pair of short *tie rods* are pinned between each ball nut flange and two arms of a *torque tube*. This slider-crank mechanism transforms the horizontal motion of the ball nut flanges into rotation of the torque tubes.

Each torque tube has two latch *hooks,* pivoted and sprung on lobes at each end (*figure 5b*). When the hooks engage *pivot pins* on the box cover, the rotation of the torque tube is transferred into vertical motion of the cover with a second crank-slider mechanism. There are four torque tubes but only two ball screws: the torque tubes furthest from the box center are driven by long *tie rods* from the central torque tubes. This method saved the weight and complication of a second pair of ball screws and associated support bearings.

The latches start in a self-locking, over-top-dead-center position. Unlatching turns the torque tubes, raising the latch hooks which are held against the cover pivot pins by *hook springs* (*figure 5c*). Some distance after the blanket preload is relieved, the hook springs are overpowered by a cam feature on the torque tube, swinging the hook out of the cover pins' path (during blanket extension). After the wing is retracted, the latch hooks are able to recapture the cover by reversing the motion.

Primary and redundant limit switches provide telemetry for the latched and unlatched positions, while hard stops protect against overtravel if the limit switches fail. Each pivot location features redundant pivot paths and lined bushings (PTFE impregnated) for controlled friction and low edge wear. The stowed preload is set at assembly by adjusting the length of the latch hooks with their central turnbuckles.

In the event of power loss or a failed motor, the latch mechanism may be operated by an astronaut using a rotary power tool. The *manual backup assembly* is located inline with the drive shaft, near the motor. A dog clutch transmits rotary power during nominal operation. This spring-loaded clutch may be disegaged by an astronaut using the *lever.* The mechanism is then driven by the astronaut's rotary power tool via a 1:1 miter gear pair. This gear mesh is never disengaged—it freewheels during nominal, motorized operation.

A kinematic analysis of the latch mechanism utilized conservative friction factors (0.30 for PTFE-lined bushings and MoS_2 surfaces, 90% efficient ball screws) and blanket compression characteristics (*figure 6*). The predicted performance satisfied the design requirements for 100% torque margin and < 20 seconds operation time (*figure 7*).

Blanket Restraint System

Function & Requirements

The Blanket Restraint System (BRS) for the SSF containment boxes is a spring actuated retractable pin mechanism designed to restrain the blanket within the containment box during launch then retract prior to solar array deployment on orbit. The functional requirements of the BRS include: restraint of the blanket during launch (with a maximum clearance ≤ 0.089 mm, or 0.0035 in, to limit transient impact loads), ability to retract in on-orbit environments, use of only the available latch drive motion for pin release, adequate telemetry to verify retraction, and reset capability during ground test with no access to the actuation system. The quantitative requirements are shown in Table 1. In addition, the multiple deployment/retraction requirement of the SSF wing requires that the BRS be resettable during ground test with minimal test operations interference. This turned out to be a driving requirement for the design of the mechanism.

Table 1: Blanket Restraint Pin Performance

Parameter	Requirement	Measured Value
Release force	≤ 222 N (50 lb)	58 N (13 lb) max
Allowable Sideload during retraction:	≥ 227 N (51 lb)	240 N (54 lb) min, 418 N (94 lb) max
Limit Load	> 5.8 kN (1,300 lb)	> 7.1 kN (1,600 lb)
Ultimate Load	> 12.5 kN (2,800 lb)	> 12.9 kN (2,900 lb)
Operational Temperature Range:	-73 to +37 C (-100 to +100°F)	-85 C (-121 °F) (hot case not tested)
Design Life: On-orbit In Test	≥ 1 retraction ≥ 50 retractions	not tested

Mechanism Description & Performance

Unlike previous smaller and lighter flexible solar arrays which relied on inter-panel friction to provide lateral restraint of their blankets during ascent, the SSF blankets are positively restrained during launch by a retractable pin system. This was required due to the weight of the folded SSF blanket assembly—over twice that of SAFE's and six times the weight of Milstar's. The use of friction alone to provide the lateral restraint of the stowed blanket was not adequate for this system without undue compressive forces that threatened to crack solar cells and cause large weight penalties to the containment box structure and latch mechanism. Thus a non-pyrotechnic, retractable pin system was determined to be needed after efforts to either increase inter-blanket friction or provide "interlocking" panel segments were deemed unreliable or impracticable (largely due to the on-orbit retraction requirement).

The pin of the BRS extends through the honeycomb structure of the box and is inserted through slots machined in aluminum stiffeners in the blanket. Some slots are in the x direction resulting in only y lateral restraint while others are slotted in the y direction resulting in x direction lateral restraint. There are a total of seven pins per box assembly. Two pins restraint the blanket in the x direction and six restrain the blanket in the y direction (one stiffener hole is circular). The slots provide allowance for relative thermal growth between the glass/Kapton/fiberglass blanket assembly and the aluminum containment box to limit thermally induced pin loads. The BRS pin will be retracted within the containment box structure once on-orbit prior to the first solar array deployment.

The heart of the mechanism is a titanium *tapered pin* nested within a stainless (303) *"expandable" pin* (*figure 10 & 11*). The expandable pin is sectioned along its length to allow for expansion when the MoS_2 lubricated tapered pin is inserted. The pins are precision machined to calculated profiles such that the expandable pin will achieve (ideally) line contact with the tapered pin upon its complete insertion into the expandable pin. After wire Electrical Discharge Machining (EDM) of the expandable pin longitudinal slots and insertion of the tapered pin, the outer surface of the expandable pin is precision ground to 20.32 +0.00/-0.04 mm (0.800 +0.000/-0.0015 in) along its interface with the blanket assembly.

The pin assembly contains a 53 N/cm (30 lb/in) *spring* compressed to 222 ± 22 N (50 ± 5 lb) for extraction of the tapered pin from the expandable pin. This retraction allows the expandable pin to contract (a maximum of 2.5 mm, or 0.100 in, diameter at the tip) in order to relieve all sideload from the pin during retraction. At this point, a 10.5 N/cm (6 lb/in) *spring* compressed to 111 ± 22 N (25 ± 5 lb) retracts the entire expandable pin assembly from the blanket into the *mounting tube* assembly. This results in release of the blanket and allows unhindered deployment of the folded blanket assembly during mast extension. In ground testing, the system can be reset to the "extended" position to allow rethreading of the blanket over the "collapsed" pin. The unit then can be cocked into the expanded position, securing the blanket into position with minimal clearance. The blanket side loading on the expandable pin is transferred to the titanium tapered pin then through the mounting tube into inserts in the honeycomb structure.

The BRS assembly employs a *pin lock* attached by *actuation cables* to a *trip lever* on the latch mechanism (*figure 12*). During unlatch of the blanket box, the lever pulls open the pin lock door resulting in release of the system. In the event of a "stuck" pin, a lockout plunger prevents the pin lock from resetting. This will allow the pin to retract on its own if an unanticipated transient event (*e.g.,* unpredicted thermal gradients) causes an initial failure to retract.

When the pin fully retracts, it releases the *lockout plunger* to allow resetting the pins and pulls two additional plungers from redundant limit switches to close a series circuit. In addition to this electrical confirmation, a yellow 3.8 cm (1.5 in) long "visual indicator" protrudes out of the end of the mounting tube and will allow an astronaut to determine if any pins have failed to retract. The retracting expandable pin assembly is captured by a padded stop at the end of the mounting tube. An interface for a reset tool was designed into this stop so that all forces required to reset an expandable pin will be reacted into its mounting tube structure. No additional bracing on the ground support equipment or flight structure is required.

Tension Mechanism

Function & Requirements

When deployed, two tension mechanisms apply tension to the flexible, hinged blanket to maintain its flatness and achieve a minimum natural frequency of 0.085 Hz for the deployed wing. The load requirement is bounded by 245 N (55 lb) minimum for the frequency requirement, and 423 N (95 lb) maximum for blanket strength (hinge loading). The operational life requirements include 35 full stroke cycles for array extensions/retractions, and 87,000 partial stroke cycles for on-orbit thermal cycles (operational and ground test cycles are doubled for qualification testing). The blanket length tolerance and thermal distortions require the full stroke to be 71 cm (28 in), and the partial stroke 8–15 cm (3–6 in). In addition, strength limitations of the partially deployed mast require that the tension be limited to less than 53 N (12 lb) until after full mast extension.

Physical Description & Performance

Each tension mechanism is a spring-driven cable drum. A constant-force spring, while providing a convenient flat force profile, was unacceptably large when designed to withstand 200,000 fatigue cycles at the design load. Instead, a pair of power springs were utilized to provide a more weight and space efficient design. The nonconstant moment produced by these springs is converted to a nominally constant force by the increasing radius of a helical cable drum. Solid film (MoS_2) lubricated ball bearings are used in the cable drum and mechanism pulley to minimize friction at these points. A complete discussion of this mechanism is given in the paper "Space Station *Freedom* Solar Array Tension Mechanism Development."

The single blankets deployed by SAFE and Milstar are tensioned during mast extension, but SSF's large power requirements and stowage envelope constraints required a split blanket/twin box design. This introduced the possibility of differing blanket lengths. Such an imbalance would mean blanket tension loads may be applied to one blanket before the other, imparting unacceptable dynamic loading on the mast during the final seconds of deployment. The solution was a two-stage tension mechanism that provides full 333 N (75 lb) only for launch restraint and when the wing is completely deployed. This was accomplished by linking each tension mechanism with the motion of the latch mechanism ball screws.

Miscellaneous Mechanisms

To control the motion of the blanket during extension and especially retraction, three guidewire mechanisms on the box base pay out over 30 m (100 ft) of wire rope attached to the box cover. A single constant-force spring powers each wire drum, producing 5.3 ± 0.9 N (1.2 ± 0.2 lb) over the considerable stroke. SAFE used multiple springs per mechanism, but the single spring design provides similar forces and reliability, saving the weight of additional spring drums, bearings, and associated fasteners. The mechanism's life requirements are similar to the tension mechanism. Reliable, even winding of the guidewire cable during retraction is ensured by a proper "fleet angle" (the angle over which the cable alternates when winding on the drum).

Other minor mechanisms on the box are an astronaut-operated soft dock mechanism, swing bolts, and an electrical connector separation mechanism where the box Orbital Replaceable Unit (ORU) interfaces with the rest of the wing. Proper stowage of a retracted and compressed blanket is maintained by small deployer bars and over 300 small extension springs at the extreme blanket ends.

Development Testing

Latch Mechanism Performance Test

This test was necessary to evaluate the overall function of the mechanism, including correlation of kinematic analysis & drag predictions, calibration & adjustment of the preload, capture & release of the box cover, proper motion of the drive train & linkages, and interaction of the limit switches & hard stops.

The test equipment consisted of a complete development latch mechanism (without the manual backup assembly). An aluminum plate and frame structure simulated the box base, and an offloaded aluminum plate simulated the box cover in zero gravity (*figure 4*). The folded blanket compression characteristics (*figure 6*) were simulated by a foam pad and appropriate spacers. A test motor with separate controller provided representative torque (up to 12.4 N-m, or 110 in-lb), though at 10% of flight motor speed (15 RPM). A torque reaction transducer measured motor output, and a single LVDT measured vertical cover motion. As for flight production, each latch hook featured a full bridge strain gauge for measuring the "axial" force in each hook (the offset pivot point at the hook end induces some bending).

The test successfully demonstrated the latch motion, adjustment, and operation. Torque measurements exceeded expectations by 0.2–0.9 N-m (2-8 in-lb, *figure 8*), but were well within the flight motor's capability with 83% torque margin. This discrepancy was attributed to additional losses in the drive train. There was slight rubbing on the cover pivots and hook spring leading to minor redesign of those components.

Blanket Restraint System Performance Test

The BRS was tested for both structural load capability as well as retraction performance. The development test employed both a full BRS pin assembly and a representative section of the containment box honeycomb (*figure 13*). The pin was loaded using 82 representative strips of "solar array blanket" with sections of aluminum stiffeners to simulate the blanket loading of the flight pins.

For retraction capability, the BRS demonstrated release at -73 C (-100 °F) with no internal binding due to thermal growth. The maximum sideload under which retraction reliably occurred was 418 N (94 lb). However, the minimum retraction of one pin assembly was just 240 N (54 lb). This was lower than expected and was attributed to internal pin loading caused by a shortening of the moment arm of the titanium pin due to pin bending during loading. The flight design was improved by providing a shoulder on the titanium pin to ensure the moment arm of the pin remains relatively constant and the internal loading more predictable.

The structural capability of the pin was very close to what was predicted. The yield of the system occurred in the titanium pin at 10.2 kN (2,300 lb) and was very benign. Ultimate failure occurred in the honeycomb insert bond line to the honeycomb and was evidenced by "crimpling" of the honeycomb around the insert.

As can be seen from the load vs. deflection curve (*figure 14*), there is a hysteresis in the system. This is due to the friction between the expandable and tapered pins. Calculations showed that this hysteresis indicated a relatively high effective friction coefficient between these members of 0.27. The development unit used Braycote 601 grease on the tapered pins with uncontrolled surface finishes. Improvements made for the flight units that will reduce the internal hysteresis and fiction include providing controlled surface finishes on the tapered expandable pins, increasing internal clearances and lubricating with sputtered MoS_2 (grease was used during development testing due to schedule constraints).

The lessons learned from the development testing included: the need for increased internal clearances between the tapered and expandable pins allow for minor pin bending; the need for a functional "break-in" test to allow initial wear of the pin stop; and the need for controlled surface finishes to improve internal friction properties.

Tension Mechanism Performance and Life Cycle Tests

The tension mechanism first exhibited unacceptable hysteresis and wear during the performance and life tests, leading to incorporation of power springs lubricated with sputtered MoS_2 and Bray oil. The paper "Space Station *Freedom* Solar Array Tension Mechanism Development" contains a full description of this test.

Integrated Box Mechanisms Performance and Life Cycle Test

Once the major box mechanisms had undergone development testing at the component level, they were assembled together on the latch mechanism test stand to verify correct interaction. The test configuration consisted of the latch, manual backup, two tension mechanisms BRS pin assemblies. Using the same instrumentation as previous latch testing, this test configuration underwent numerous simulations of all operational sequences: the combined unlatch/detension/BRS release sequence, tension wire extension, full tension application, detension sequence, tension wire retraction, and latch/tension sequence. The test indicated proper performance of the integrated mechanisms with only minor enhancements necessary to the BRS release hardware. These enhancements were to provide adjustment of the release cables during assembly and to provide increased stroke from the torque tube lugs.

The set of mechanisms were exercised through 70 tension/detension cycles, and 30 latch/tension/unlatch/detension cycles—twice the on-orbit life requirement. At the end of the testing, all mechanisms were still functioning as designed. Post-test inspection of the mechanisms revealed no adverse wear but some organic wear debris on the ball screw assembly. The development ball screws were tested unlubricated, but were not cleaned of the residual coating applied by the supplier for storage. Flight ball screws will be thoroughly cleaned and lightly lubricated with Bray 601 grease plate.

Future Testing

Funding caps and system level redesign of the space station have delayed the qualification testing of the wing, including the containment box mechanisms, until late 1994 through 1995. The testing will include qualification of the tension mechanism at the component level to demonstrate performance, after exposure to severe random vibration, for twice the operational life cycles (100 extension/retraction cycles and 176,000 thermally induced cycles). The life cycling will be performed under full thermal and vacuum conditions in an accelerated life test. The guidewire mechanism will undergo similar life cycle testing. At the wing assembly level, the containment box mechanisms will be qualification tested for full functional performance of both automatic and manual backup modes before and after exposure to acoustic environments and periodically during operational life cycle testing (>100 full extension/retraction cycles and >50 unlatch/latch cycles). Life cycle testing at the wing level is being performed at ambient conditions due to the large size of the deployed array (7.6 by 33.5 m, or 25 by 110 ft, for the test configuration utilizing only one of the two containment boxes and blankets). Functional testing of the latch mechanism and blanket restraint system at the containment box and wing assembly level under thermal and vacuum conditions will be performed on a "protoqual" basis on each flight wing. This test will include a first motion demonstration of the wing extension as well as simulation of worst case containment box thermal gradients during the operation of the mechanisms.

Conclusion

The major containment box mechanisms for the Space Station *Freedom* solar array wing have been design, built, and undergone component and integrated development testing. Performance of the mechanisms and their interactions was successfully verified by the development testing and minor enhancements to the hardware have been incorporated. Production of qualification units has begun, to be tested during 1994. First flight is scheduled for 1997.

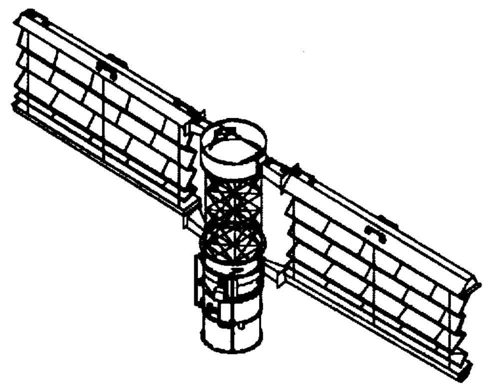
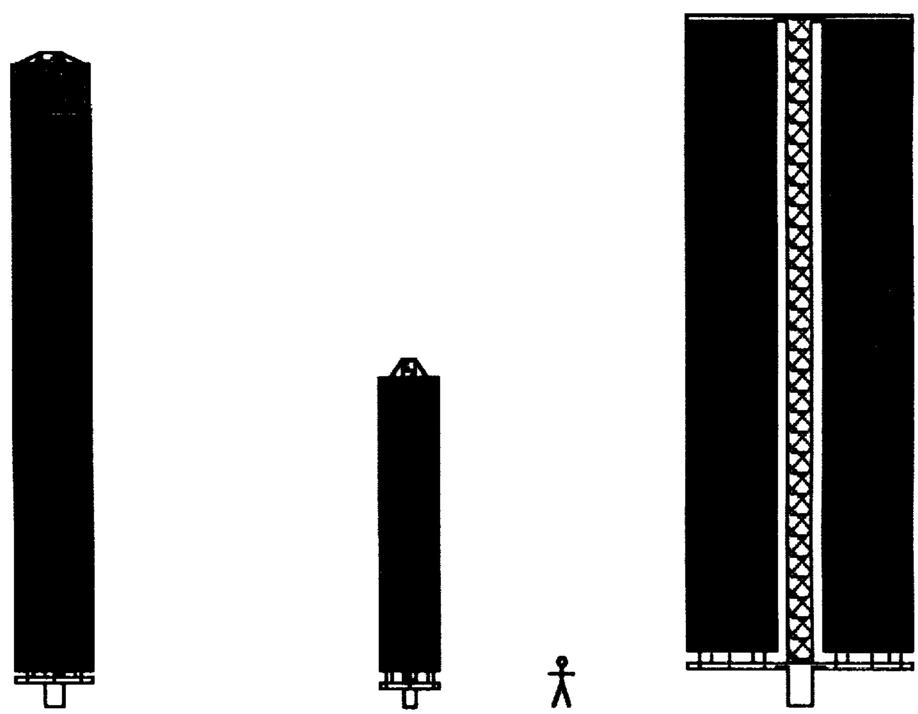


Figure 1: Space Station Freedom Solar Array Wing (Deploying)



SAFE	Milstar	SSF
4 by 33 m (13 by 109 ft)	3 by 16 m (10 by 52 ft)	12 by 35 m (38 by 115 ft)
426 kg (940 lb)	88 kg (195 lb)	1082 kg (2386 lb)
Retractable single blanket	Single blanket	Retractable split blankets
n/a	8.5 kW	32 kW

Figure 2: SAFE, Milstar, and SSF Solar Array Wings

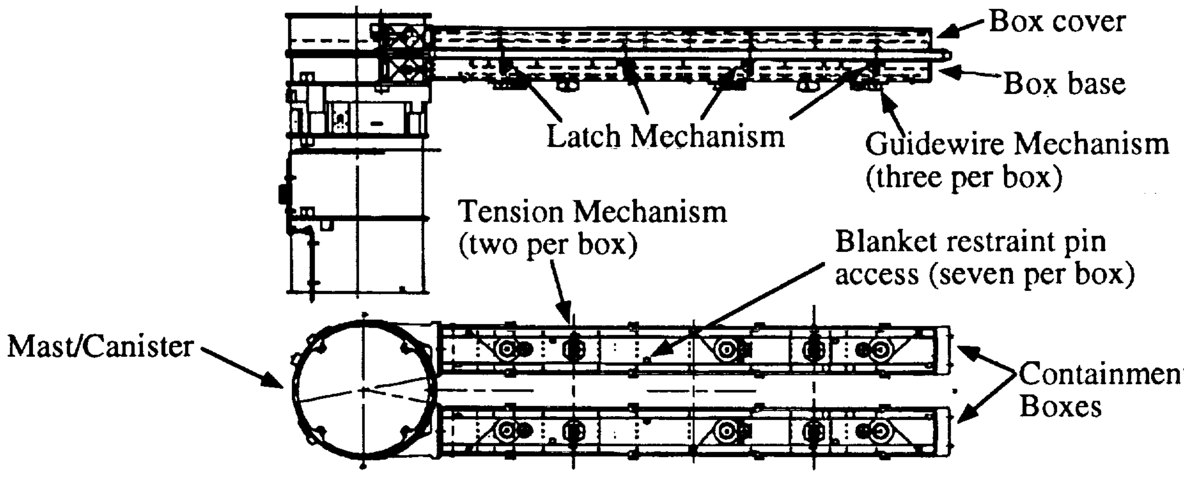


Figure 3: SSF Stowed Wing Layout

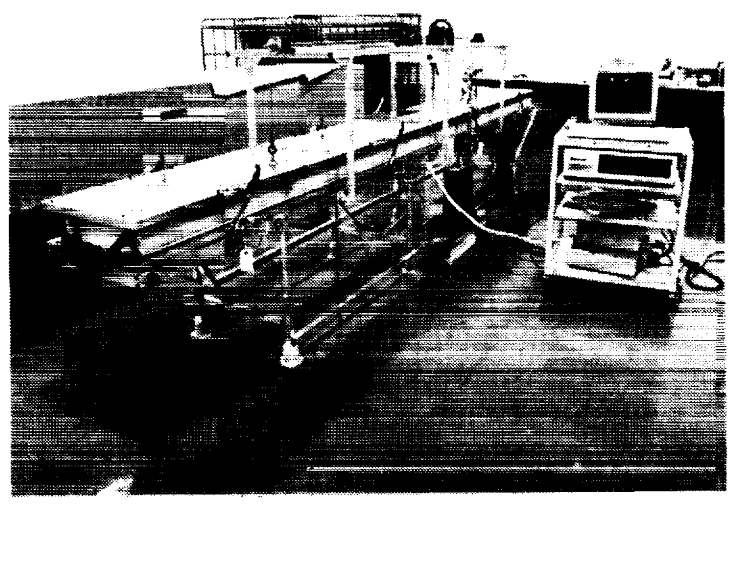


Figure 4: Latch Mechanism Development Test

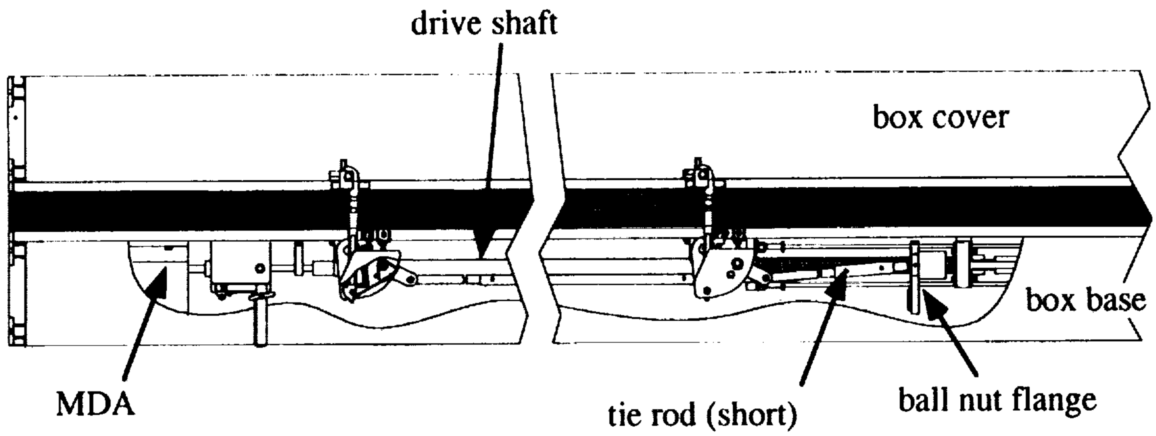


Figure 5a: Latch Mechanism Kinematics (Latched)

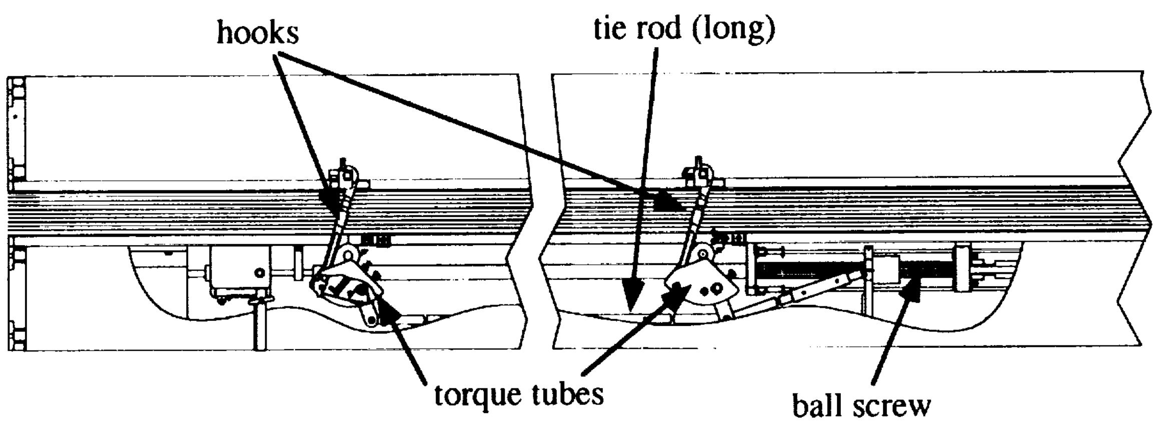


Figure 5b: Latch Mechanism Kinematics (Mid-operation)

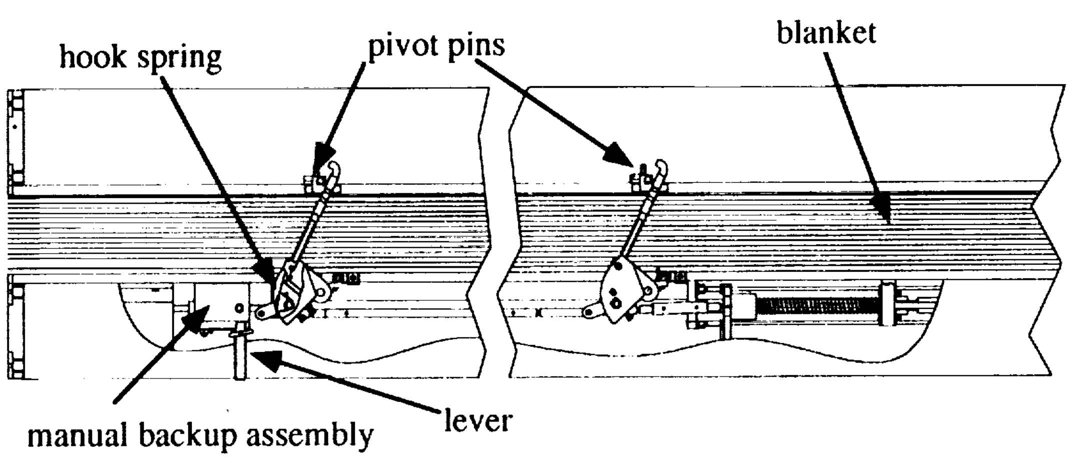


Figure 5c: Latch Mechanism Kinematics (Unlatched)

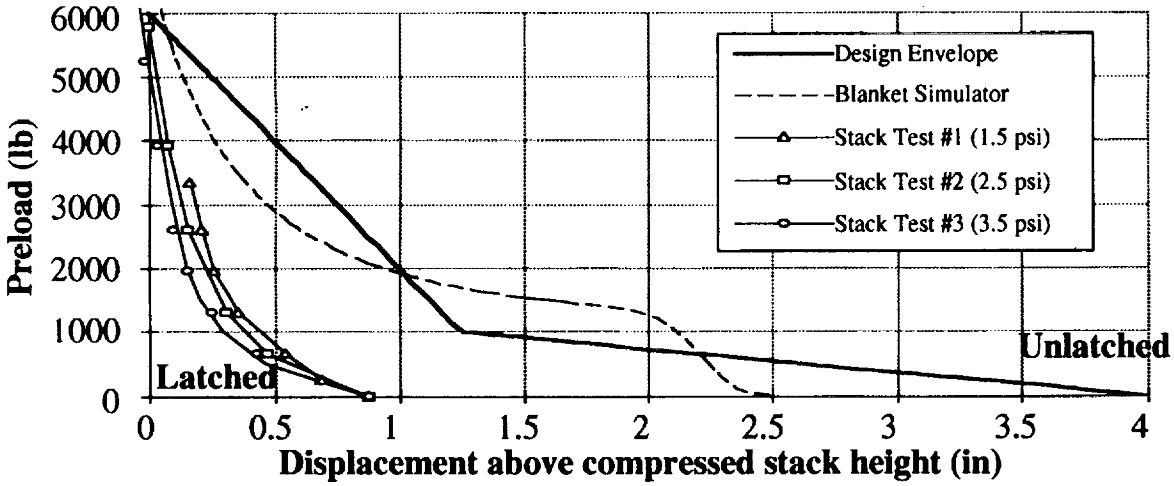


Figure 6: Blanket Compression Characteristics

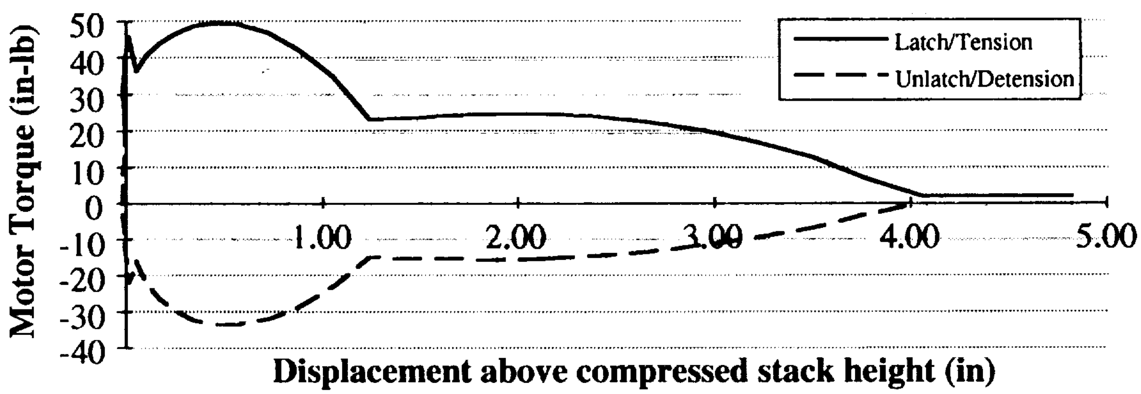


Figure 7: Latch Mechanism Performance Prediction

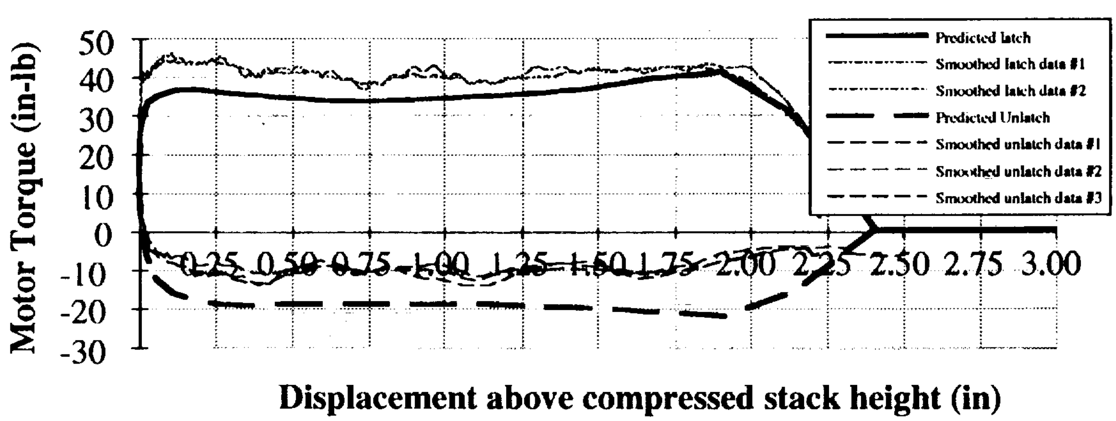


Figure 8: Latch Mechanism Development Test Results

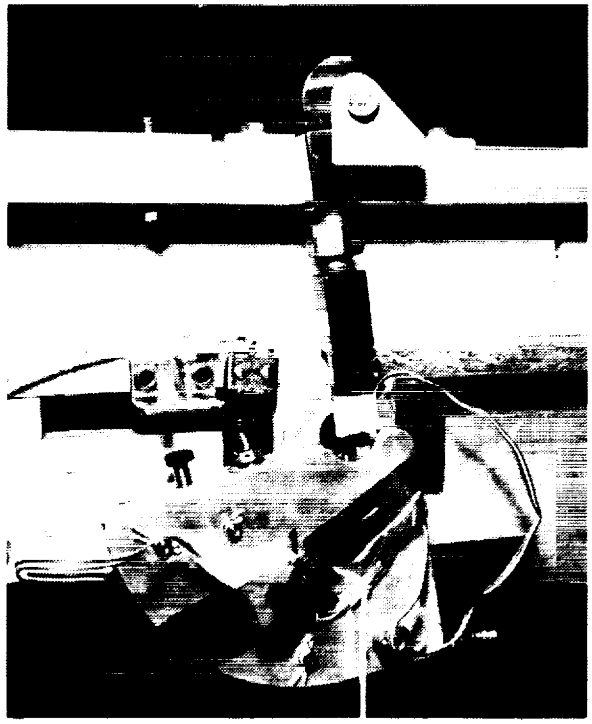


Figure 9: Development Latch Mechanism Hook

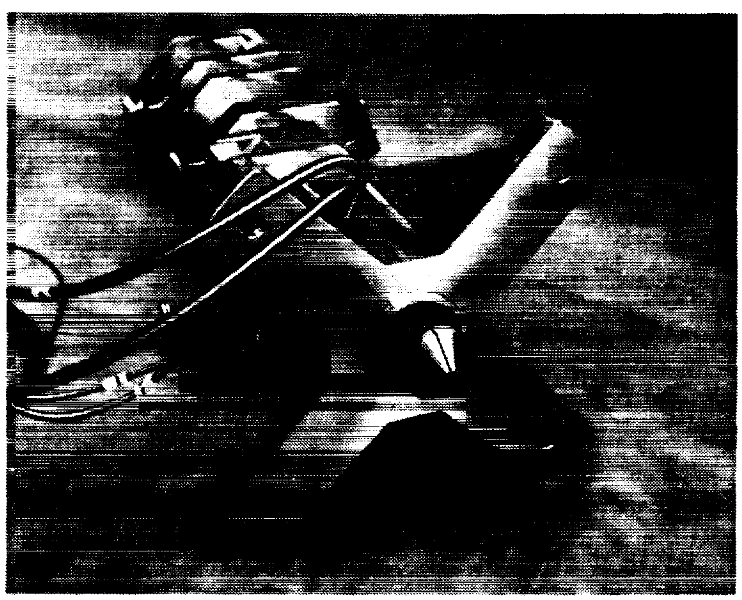


Figure 10: Blanket Restraint System Pin Assembly

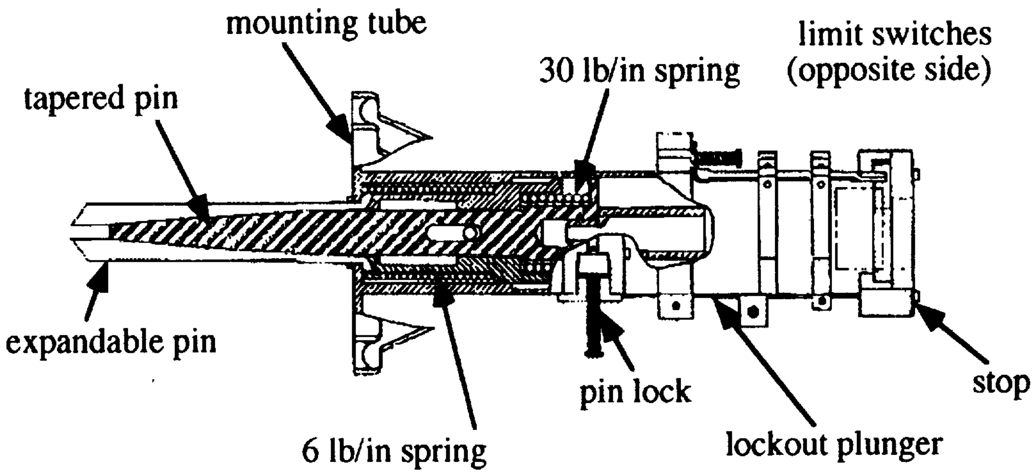


Figure 11: Blanket Restraint System Pin Assembly Cross-Section

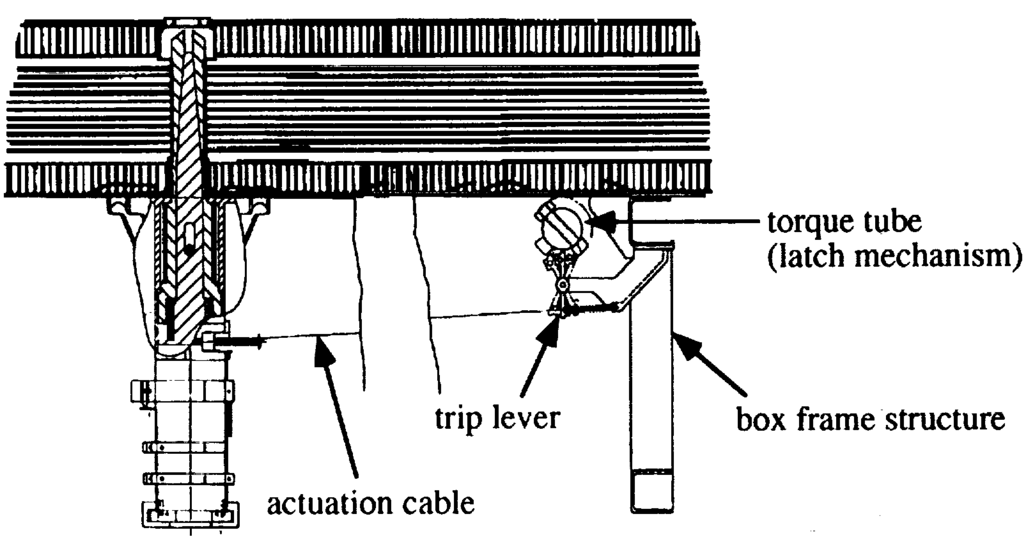


Figure 12: Blanket Restraint System Actuation

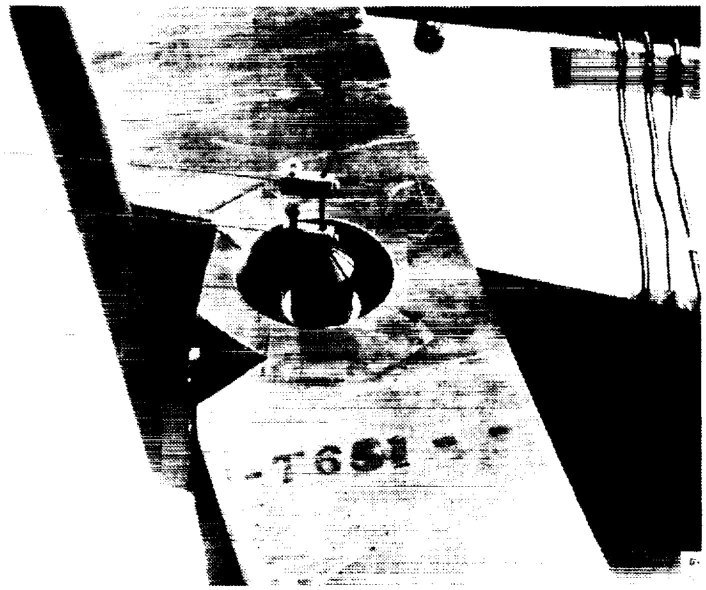


Figure 13: Blanket Restraint System Development Test

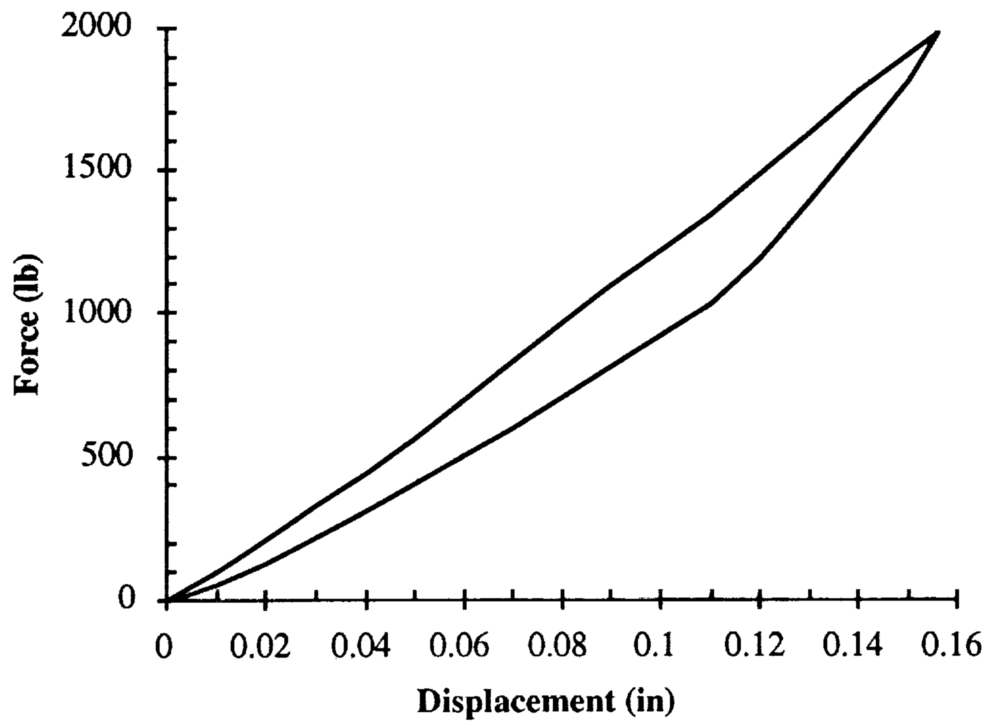


Figure 14: Blanket Restraint System Test Results