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THE SOLAR ANOMALOUS AND MAGNETOSPHERIC PARTICLE EXPLORER (SAMPEX) YO-YO DESPIN AND SOLAR ARRAY DEPLOYMENT MECHANISM

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ABSTRACT

The SAMPEX spacecraft, successfully launched in July of 1992, carried a yo-yo despin system and deployable solar arrays. The despin and solar array mechanisms formed an integral system as the yo-yo cables held the solar array release mechanism in place. The SAMPEX design philosophy was to minimize size and weight through the use of a predominantly single string system. The design challenge was to build a system in a limited space, which was reliable with minimal redundancy. This paper will cover the design and development of the SAMPEX yo-yo despin and solar array deployment mechanisms. The problems encountered during development and testing will also be discussed.

INTRODUCTION

The Solar, Anomalous and Magnetospheric Particle Explorer (SAMPEX) was the first in a series of Small Explorer class satellites (see photos, figures 1 and 2). SAMPEX, with its cluster of particle detectors, was launched into a near polar orbit aboard a Scout launch vehicle from Vandenberg Air Force Base on July 3, 1992. The SAMPEX spacecraft carried a yo-yo despin system and deployable solar arrays. The launch of SAMPEX and the successful operation of the yo-yo despin and solar array deployment culminated about two and a half years of development effort at NASA's Goddard Space Flight Center.

The SAMPEX yo-yo despin and solar array deployment mechanisms formed an integral system as the yo-yo cables held the solar array release mechanism in place. Tying the yo-yo and solar array operation together allowed one spacecraft command to both despin the spacecraft and deploy the solar arrays, thereby reducing the number of actuators, relays and wiring. Once the yo-yo cables unwrapped, the release mechanism was free to unlatch and deploy the solar arrays.

SAMPEX is a small spacecraft weighing 157 Kg (347 lb) with a launch size of .74 meters in diameter by 1.4 meters in height. The width after solar array deployment grew to over 2 meters with the arrays providing about 1.6 square meters of solar cell area. The solar arrays consist of two mirror image wings, each wing comprised of two hinged honeycomb panels. The array deployment was spring driven with viscous fluid damping.

The SAMPEX design philosophy was to minimize size and weight through the use of a predominantly single string system. The design challenge was to build a reliable system in a limited space with minimal redundancy that would function under extreme conditions. A major hurdle was to have the system operate at the worst case test temperature of -75°C.

This paper will discuss the design of the SAMPEX yo-yo despin and solar array deployment mechanisms. The problems encountered during testing and their resolutions will also be covered.

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DESIGN REQUIREMENTS

Yo-yo Despin Mechanism

The SAMPEX spacecraft was launched atop a Scout launch vehicle which has a spin stabilized 4th stage. A system was needed to despin the spacecraft to a rate at which the attitude control system could operate. The Scout vehicle offers a standard yo-yo despin system but we chose to provide our own for two reasons: first, in the early design stages we were very weight critical and could save weight by designing our own despin system; second, because our solar arrays and lower antenna extend below the vehicle separation plane, a spinning separation was desired to minimize any tip-off that might occur during separation from the 4th stage. The design requirements of the yo-yo despin mechanism were:

Despin the spacecraft after 4th stage separation to an absolute spin rate of less than 3 rpm from an initial spin rate of 141 ± 10 rpm.

Yo-yo despin operation must initiate the Solar Array deployment.

The total despin system mass must be less than 2.27 Kg (5 lb), the mass of the Scout provided despin system.

Solar Array Deployment Mechanism

The solar array design requirements were determined by a number of factors. Most important of these were; spacecraft power requirements, attitude control system constraints, instrument pointing requirements, launch vehicle interface, spacecraft dynamic and thermal environment, and spacecraft testing requirements.

The instrument pointing scheme and attitude control plan allowed the use of fixed solar arrays. The arrays were sized based on the spacecraft power requirement of approximately 100 watts, using fixed arrays with gallium arsenide cells. The design requirements, both given and derived, of the solar array deployment mechanism were:

Provide 1.67 square meters (18 ft^2) of fixed solar array area.

Withstand launch and spin loads while in the stowed position. For this launch the thrust loads were 10g, lateral loads were 4.5g with spin loading of 12g. There were also significant shock and random loads.

Fit within .735 meter (30 inch) diameter envelope of the Scout .86 meter (34 inch) diameter heatshield in the stowed position.

The fundamental mode of the solar array panels in the stowed position must be greater than 30Hz. This prevented coupling between the spacecraft and the solar arrays (the spacecraft had a requirement for first bending mode between 15Hz and 30Hz).

The minimum natural frequency of deployed array must be greater than 2Hz to prevent coupling with the attitude control system.

Withstand spacecraft spin rate up to 3 rpm during deployment.

System must withstand its own induced dynamic loading during deployment.

Deployment must be possible in a 1g environment for ease of testing.

System must operate at temperatures from +30°C to -75°C.

SYSTEM DESIGN

The solar array system consists of 2 symmetric wings, each comprised of 2 solar array panels as shown in figure 3. The panels are attached to the spacecraft primary structure and to each other by spring loaded hinges with dampers to control the rate of deployment. Each wing was held in the stowed position by a primary release mechanism restrained by the yo-yo despin cables. The cables were wrapped twice around the spacecraft circumference and the despin weights at the end of the cables were held captive by electro-explosive pin pullers.

Firing the pin pullers released the yo-yo weights, allowing the cables to unwind and fly free, thereby despinning the spacecraft. The release levers, unrestrained by the yo-yo cables, slowly rotated out releasing the arrays. Upon release the panels were pushed out about a centimeter by kickoff springs and then slowly deployed to their operating position where they locked in place.

Yo-yo Despin System

The yo-yo despin mechanism consisted of a pair of weights and cables wrapped around the spacecraft and was required to despin the spacecraft to an absolute spin rate of less than 3 rpm after 4th stage separation. A cable guide (top view, figure 3) consisting of nine separate sections, formed a circular path on which the cables were wrapped. The total mass of the despin system including cables, weights, pin pullers, cable guides and other hardware was 1.8 Kg (3.9 lb).

The yo-yo weights were held in place by electro-explosive pin pullers. When the pin pullers fired the weights came free and the cables unwrapped. At the point where the cables completely unwrapped and reached a point radial to the spacecraft they flew free. Due to the conservation of angular momentum and kinetic energy the spacecraft was despun. Through the use of well defined equations, the system components were sized to despin to the desired rate.[1,2] The variables needed to size the yo-yo system and the values for the SAMPEX spacecraft were:

Spacecraft Moment of Inertia, $I = 9.79$ Kg-m^2 (7.23 sl-ft^2)
Spacecraft Radius, $a = .38$ m (1.24 ft)
Total Mass of Tip Weight, $m = 154.15$ grams
Length of Yo-yo Cable, $l = 4.7$ m (15.443 ft)
Lineal Density of Cable, $p = 21.9$ gm/m (.0147 lb/ft)
Spacecraft Initial Spin Rate, $W_o = 141$ RPM
Spacecraft Desired Final Spin Rate, $W_f = 0$ RPM
Release Angle of Cable Relative to Radial, $\Theta = 0°$

Yo-yo despin systems have been used on many spacecraft and sounding rockets with great success and the basic design is quite straightforward. The SAMPEX design though, had several unique features. One, the spacecraft has a tank containing 8 Kg of isobutane fluid. There was great concern that this fluid would affect the ability to balance and despin the spacecraft accurately. Next, the yo-yo cables held the solar array deployment mechanism in place. Finally, due to the criticality of meeting the ±3 rpm final spin rate, a sensitivity analysis was performed to assure normal variations in the input variables would not significantly affect the despin accuracy.

The isobutane tank on the SAMPEX spacecraft is unbaffled and mounted directly on the spin axis. We had two main concerns about the tank: the first was that the fluid would affect the accuracy of the spacecraft moment of inertia measurement, a value needed for the despin analysis; the second was that the fluid would affect the accuracy of the despin itself. To deal with the inertia measurement concern the fluid inertia was calculated as if it were a solid and found to be about .6% of the total spacecraft inertia. This number was added to the inertia tolerance numbers used in the sensitivity analysis. To determine the effect of the fluid on the despin itself two extreme cases were analyzed. The first case assumed the fluid would act as a solid and therefore not affect the despin at all. The second case assumed the fluid was completely decoupled from the spacecraft and after despin would slowly dissipate its momentum into the spacecraft. Fortunately, the difference between these two cases was only about half an rpm, well within the 3 rpm limit.

The fact that the yo-yo cables held in the solar array release also posed two main concerns; one, that the spring loaded release levers would push out during despin, altering the spacecraft radius; and two, that the arrays may begin deploying before despin was complete, possibly damaging the arrays or affecting the despin. As it turned out, once despin had begun, the tension in the cables was sufficient to hold the release levers in place. To prevent the arrays from deploying prematurely, dampers were added to the release mechanism to slow down the release time to several seconds, whereas the despin time was less than one second.

A sensitivity analysis was performed because it was very critical to meet the ±3 rpm despin target. This was mainly due to power limitations, as the solar arrays needed to be pointed at the sun before the spacecraft battery was depleted. At spin rates higher than 3 rpm this could not be guaranteed. At even higher rates the attitude control system would become unstable, unable to stabilize the spacecraft, and the mission would be lost. This analysis varied all input variables simultaneously to obtain worst case positive and negative final spin rate. The tolerances applied to all the input variables were:

Spacecraft Moment of Inertia, I:	±1%
Spacecraft Radius, a:	± 6.3 mm (.25 in)
Total Mass of Tip Weight, m:	± .1 gm
Effective Length of Cable, l:	± 12.7 mm (.5 in)
Lineal Density of Cable, p:	± .06 gm/m
Spacecraft Initial Spin Rate, W_o:	± 10 rpm
Release Angle of Cable Relative to Radial, Θ:	± 5°

The worst case spin rate using these conservative tolerances with no fluid effect came to 1.98 rpm. The worst case rate including the fluid correction (vary I ± 1.6%, decouple fluid during despin) came to 2.42 rpm, within the 3 rpm target.

Solar Array Panels

The 4 SAMPEX solar array panels are aluminum honeycomb, 1.15 meters tall by .37 meters wide (45.2 by 14.5 in.). The core is 9.5 mm (3/8 inch) thick with face sheets of .2 mm (.008 inch) 7075-T73 aluminum alloy. Hardpoints for hinge and component mounting are integrally bonded into the panels. The bare panels weigh about 1.4 Kg (3 lb) each. The panels with the gallium arsenide cells installed weigh just over 2.3 Kg (5 lb) each. The minimum mode of a panel, simply supported at the hinge locations is about 45 Hz.

Panel Hinges

The panel hinges (figure 4) act in pairs, upper and lower, to form a hinge line. All hinges use spherical bearings with a 6.35 mm (1/4 in) stainless steel shaft. The upper hinge bearings are fixed to the shaft to carry the thrust loading of the array. The lower hinge bearings are free to slide along the shaft to allow for misalignment and thermal expansion.

Each upper hinge has a potentiometer to provide position data while a fluid damper is incorporated into each lower hinge. The dampers dissipate the deployment energy and minimize impact loads at panel lock in. The stainless steel torsional springs are adjustable in 45° increments.

The main difference between the inboard and outboard hinges is the inboard hinges travel 90° while the outboard hinges travel 180° (see figure 5). This required a slightly different configuration and different torsional springs but otherwise the designs are identical.

To enable the deployed array to meet the 2 Hz minimum frequency requirement, each hinge includes a locking feature. This locking feature consists of a simple pin/detent design as shown in figure 6. Each hinge contains a spring loaded stainless steel pin. When the hinge reaches its fully deployed position, the pin slides into a detent, locking the array in the open position. The surface where the pin slides on the hinge is finished with teflon impregnated, hard anodize. The deployed panel frequency is over 2.6 Hz.

Rotary Viscous Damper

There are a total of 6 dampers on the solar array deployment mechanism, one on each solar array hinge line and one on each primary release mechanism. These dampers are D.E.B Manufacturing, Sesco Model 1080 Sub-Miniature Rotary Viscous Damper. The dampers are very small, 3.5 cm diameter by 5 cm long, and have the following properties:

Damping Rate: 2.8 N-m/rad/sec (25 in-lb/rad/sec)
Friction Torque: .03 N-m (4 oz-in)
Maximum Torque Capacity: 11.3 N-m (100 in-lb)
Weight: 85 grams
Damping Fluid: McGhan Nusil CV-7300 silicone fluid

Primary Release Mechanism

There are two symmetric primary release mechanisms, one for each solar array wing. The primary release mechanisms held the arrays in the stowed position and were released by the deployment of the yo-yo cable. When the yo-yo cable unwrapped, the spring loaded release levers were free to move. Each release lever turns out thereby rotating a deployment shaft which is linked to an upper and lower release plunger. The rotary motion of the shaft is converted to linear motion at the release plungers. The plungers pull clear from their mating slots in the outboard hinge bodies and the arrays are free to deploy. The deployment shaft is tied by means of a linkage to a rotary damper to slow down the deployment and prevent any possibility of the arrays deploying before despin is complete. A primary release mechanism is shown in figure 8.

The entire arrangement is set up to prevent backdriving the system. The loads carried by the release plungers are not taken by the release mechanism drivetrain. The upper plunger carries only thrust direction loads and panel out of plane loads. The lower plunger carries only panel out of plane loads. These loads are reacted by the plunger housing which is mounted directly to the spacecraft primary structure. Loads in the direction of the plunger are taken by the opposing inboard hinge.

All of the release mechanism parts, except pins, bushings and fasteners, are made from 7075-T73 aluminum alloy. This alloy has high strength and low susceptibility to stress corrosion cracking. All areas of sliding friction are finished with teflon impregnated hard anodize. A light coating of Braycote 601 grease was also applied to these surfaces. The deployment shaft is mounted using teflon bushings.

Outboard Panel Release Mechanism

The outer panel release mechanism consists of a passive ball/detent arrangement as shown in figure 7. A hook with a steel ball is mounted to the outer panel opposite each hinge location. When the arrays are stowed, the ball fits into a detent in the fixed half of the inboard hinge assembly. Upon primary release, the inboard and outboard panels rotate together for about 30 degrees. At this point, due to the geometry of the system, the ball is clear of the detent and the outer panel is free to deploy, driven by its damped spring system.

Kick-Off Plungers

There are two spring loaded kick-off plungers on each inboard solar array panel. The plungers are located at each edge of the panel directly under the yo-yo cable location. The plungers serve a dual purpose. First, upon release of the primary release mechanism, the kick-off plungers provide an initial impetus to the arrays to help overcome any static friction. Second, the plungers act as standoffs between the outer solar array panel and the spacecraft structure to prevent the tensioned yo-yo cable from deflecting or damaging the panel.

Thermal Concerns

Our main thermal concern during design was that thermal expansion or contraction would causing binding and lockup the mechanism. This was accounted for in the hinge and release mechanism design. The upper hinges and upper panel release mechanisms carried the panel loads in the thrust direction. The lower hinges and release mechanisms were free to slide in the thrust direction allowing for thermal expansion. Similarly the hinged side of the panel (inboard hinge for the inboard panel) carried the lateral loads in the plane of the panel where the release mechanisms allow expansion and contraction in that direction.

We had two other thermal concerns. One was damper freezing at cold temperatures and the other was heat from the arrays conducting into the spacecraft through the inboard hinge. The solar arrays get very hot due to the solar flux and the power system shunts mounted to the back of the arrays. This heat could flow into the spacecraft and cause overheating under certain conditions. To solve this problem the hinge piece mounted to the inboard solar array was made from titanium; thus reducing heat flow from the arrays but allowing heat to reach the inboard dampers, preventing them from freezing. In the event of a cold deployment, heat flow from the spacecraft would protect the inboard dampers; however, the outboard dampers were at risk of freezing. A heater was applied to the outboard damper to prevent possible freezing.

SYSTEM TESTING

The testing of the SAMPEX spacecraft consisted of a combination of environmental, functional and measurement tests. Since the SAMPEX spacecraft was a completely new design, vigorous testing was required to qualify it for flight. To accomplish much of the testing, a mechanical Engineering Test Unit (ETU) of the spacecraft was built. The ETU was structurally and mechanically identical to the SAMPEX flight unit. All instruments, electronics and other components were represented by mass models. The tests performed on the SAMPEX ETU were:

- Yo-yo despin mechanism deployment test, ambient and vacuum
- Deployment testing of the solar array, ambient and thermal vacuum
- Vibration testing
- Spin balance
- Mass properties measurement (Spin axis moment of inertia only)

The tests performed on the SAMPEX flight unit were:

- Deployment testing of the solar array, ambient and thermal vacuum
- Vibration testing
- Spin balance
- Mass properties measurement
- Acoustics testing

Other "non-mechanical" tests are not enumerated here. Despin testing was not performed on the flight unit as it was deemed too risky.

Despin Testing

Our first set of despin tests was performed at NASA's Wallops Flight Facility in October 1991. The tests were performed outdoors since the test required an area 10.5 meters in diameter and we could not find a suitable indoor facility. At that time our predicted initial spin rate was 162 rpm, it was later revised to 141 rpm when the true spacecraft mass properties were known. We initially wanted to run 3 tests, one nominal spin rate, one high spin rate, and one low spin rate. Due to an anomaly the first test was repeated. Each test was recorded on video tape and high speed film. We were initially wary of allowing the arrays to deploy after despin because they could be damaged if the despin did not function properly. After several runs, the system appeared to be functioning properly and the arrays were released on the final test. The result of each deployment test were as follows:

Test 1: 162 RPM, Arrays Locked (not deployable)
Result: Spin table failed to declutch, and one cable failed to release due to out of spec ball end on cable.

Test 2: 162 RPM, Arrays Locked
Result: Test appeared successful but only despun to about 4.6 RPM

Test 3: 192 RPM, Arrays Locked
Result: Test appeared successful but only despun to about 8 RPM

Test 4: 142 RPM, Arrays Unlocked
Result: Test successful, despun to about 2.3 RPM, arrays deployed successfully

An anomaly occurred in Test 1 where the swaged ball on one cable failed to release from the cable retainer. The spin table also failed to declutch on this test. We originally thought that the table malfunction caused the failure by not allowing the cable to reach the radial (release) position. Upon review of the high speed film however, it was apparent that the cable did reach the radial position and should have released. Inspection of the hardware found that the cable ball was caught in its release slot because the ball was .1 mm oversize and the slot was .08 mm too small. Due to the small nominal clearance (.15 mm), these out of specification conditions caused interference and the ball wedged in the slot. To resolve this problem the slots were opened up to provide 1.7 mm of nominal clearance, all hardware was reinspected to assure compliance with the drawings, and an analysis was performed to assure no interference could occur due to thermal contraction or expansion. Only one of the despin tests performed met the ± 3 rpm despin requirement. We attributed this to air resistance since the lower the initial rate, the closer the despin was to zero. To assure the accuracy of the system and checkout the fixes made to correct the release problem, we decided to run another despin test in vacuum.

The final despin test was performed in a 16 meter diameter vacuum chamber at NASA's Langley Research Center in April 1992. This test was performed using the spacecraft ETU with the flight cable retainers and an improved spin table. This time the spacecraft despun to exactly zero rpm with no detectable residual rate.

Solar Array Testing

The solar array deployment mechanism was tested in ambient conditions and in thermal vacuum at hot, nominal and cold conditions. The system functioned flawlessly in all cases except extreme cold.

During our initial ETU solar array testing the system functioned properly at hot and nominal temperatures but the outboard panels failed to deploy at cold temperature (-65°C). Upon warming of the thermal vacuum chamber, the arrays deployed. In this test the outboard arrays failed to deploy. Upon investigation, it was discovered that the damper fluid freezes between -41°C and -42°C.

To alleviate the freezing problem, 1 watt heaters were applied to the outboard dampers. This modification incorporated a limit switch to turn off the heaters when the arrays started to deploy. The inboard dampers being thermally coupled to the spacecraft did not have a freezing problem.

The system was tested a second time with the heaters installed and again functioned properly at nominal temperatures. At cold temperature (-75°C) however, the outboard arrays deployed to over 90% of their full open position then stalled. The dampers, even with heaters, were running about -33°C before deployment and the temperature dropped rapidly (below -40°C before movement stopped) when heater power was cut off at initial deployment. The arrays deploy very slowly at these temperatures, due to the high viscosity of the damping fluid. It was assumed (incorrectly) that the failure was due to the damper fluid freezing before the deployment was complete. Upon warming the chamber, the arrays fully deployed when the damper temperature read -38°C. The arrays deploying when warmed made trouble shooting difficult because the arrays could not be inspected in the failed configuration.

To solve the problems of the second test, three fixes were employed. Thermal isolation of the damper was increased by adding a non-conductive coupling to prevent heat loss through the damper shaft. The spring torsion in the hinge was increased to provide a faster deployment rate, thereby minimizing the effect of the temperature drop. The heater power was increased to 2 watts to provide a higher initial temperature in the damper. The increase in heater power was employed on only one solar array damper to determine if the other fixes would be sufficient to solve the problem.

After these fixes were employed a third ETU test was run at cold temperature (-75°C). The damper with the 2 watt heater was running about -19°C, the damper with the 1 watt heater was running about -41°C. We predicted that the 2 watt side would open because we had successful deployments below -19°C and that the 1 watt side would fail because it had failed the previous test at a warmer temperature.

To our surprise the 1 watt side opened, while the 2 watt side opened about 90% then stalled. After verifying that the 2 watt heater was on the correct side the chamber was brought to ambient. Again the array deployed fully upon warming.

Upon inspection of the mechanisms we discovered a problem with the 2 watt side that was not evident on the 1 watt side. On the upper outboard hinge, the rotating leg of the torsion spring was rubbing on the fixed hinge, causing a noticeable amount of drag that was not evident on the 1 watt side. This occurred because the torsion spring was catching in a gap between the spring mandrel and the hinge (see figure 9). The condition also put the torsion spring under a bending force which caused some galling between the coils of the spring. Apparently the spring would catch in the gap intermittently. The drag that occurred was not severe enough to prevent deployment at warmer temperatures. Apparently the aluminum hinge would contract at colder temperatures, aggravating the problem and preventing deployment. To fix this problem the spring mandrel was modified to eliminate the gap.

Based on our testing to this point we selected a flight unit configuration to minimize changes from the latest ETU test. This configuration consisted of 2 watt heaters on both outboard dampers, the outboard hinge spring torsion was increased, the dampers were thermally isolated, and the mandrels were modified to eliminate any gap.

The flight unit was tested at -50°C and deployed successfully. Unfortunately the cold temperature was limited by the cold limit on the SAMPEX instruments and the mechanism was not technically qualified to the worst case temperature of -75°C. A fourth ETU solar array test was then run at -75°C with the configuration identical to the flight unit. The arrays deployed very slowly at this temperature, but after sufficient hand wringing they finally deployed to the full open position and locked in place.

CONCLUSION

The SAMPEX launch took place on July 3, 1992 at 7:19 AM Pacific time. About ten minutes after liftoff the spacecraft separated from the Scout 4th stage with the despin command set to occur 12 seconds later. Although we had to wait four and half agonizing hours before the first ground contact confirmed the successful operation of our system.

ACKNOWLEDGEMENTS

The author would like to thank **Suk Yoon** of General Electric Government Services for his outstanding design work and dedication in the development of this system. I would also like to thank **Steve Patton, Gary Cooper** and **Mike Adams** of NASA Goddard for their invaluable contributions towards the integration and testing of this system.

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2. Flatley, Tom, A Yo-yo Despin Analysis (NASA GSFC X-732-70-476, October 1970).

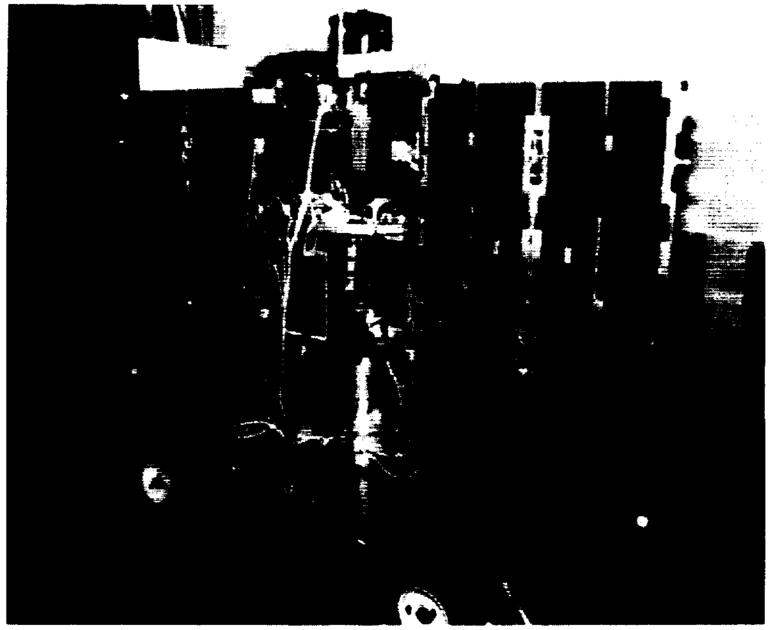

Figure 1: SAMPEX During Integration at Goddard Space Flight Center, Greenbelt, MD

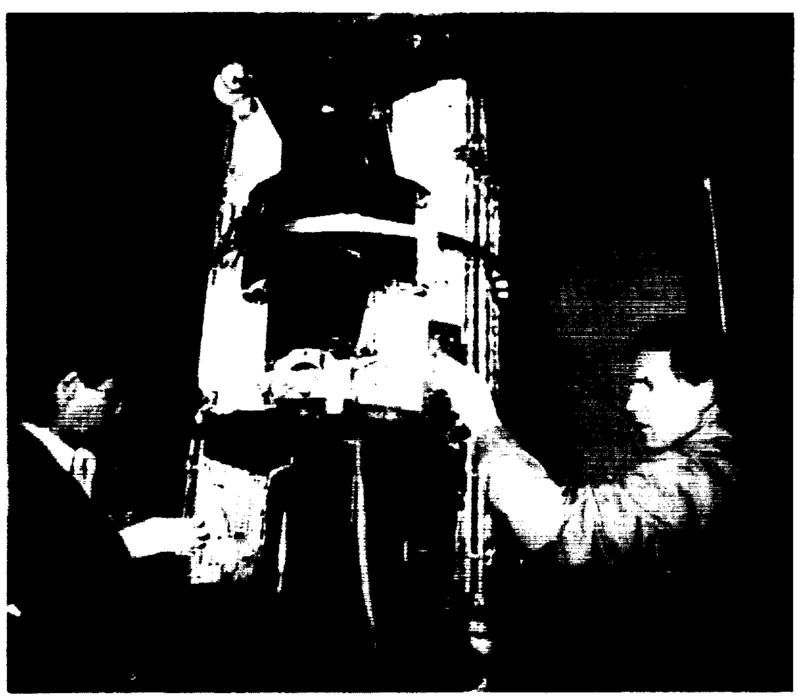

Figure 2: SAMPEX Prior to Fourth Stage Mating in Scout DBF, Vandenburg Air Force Base

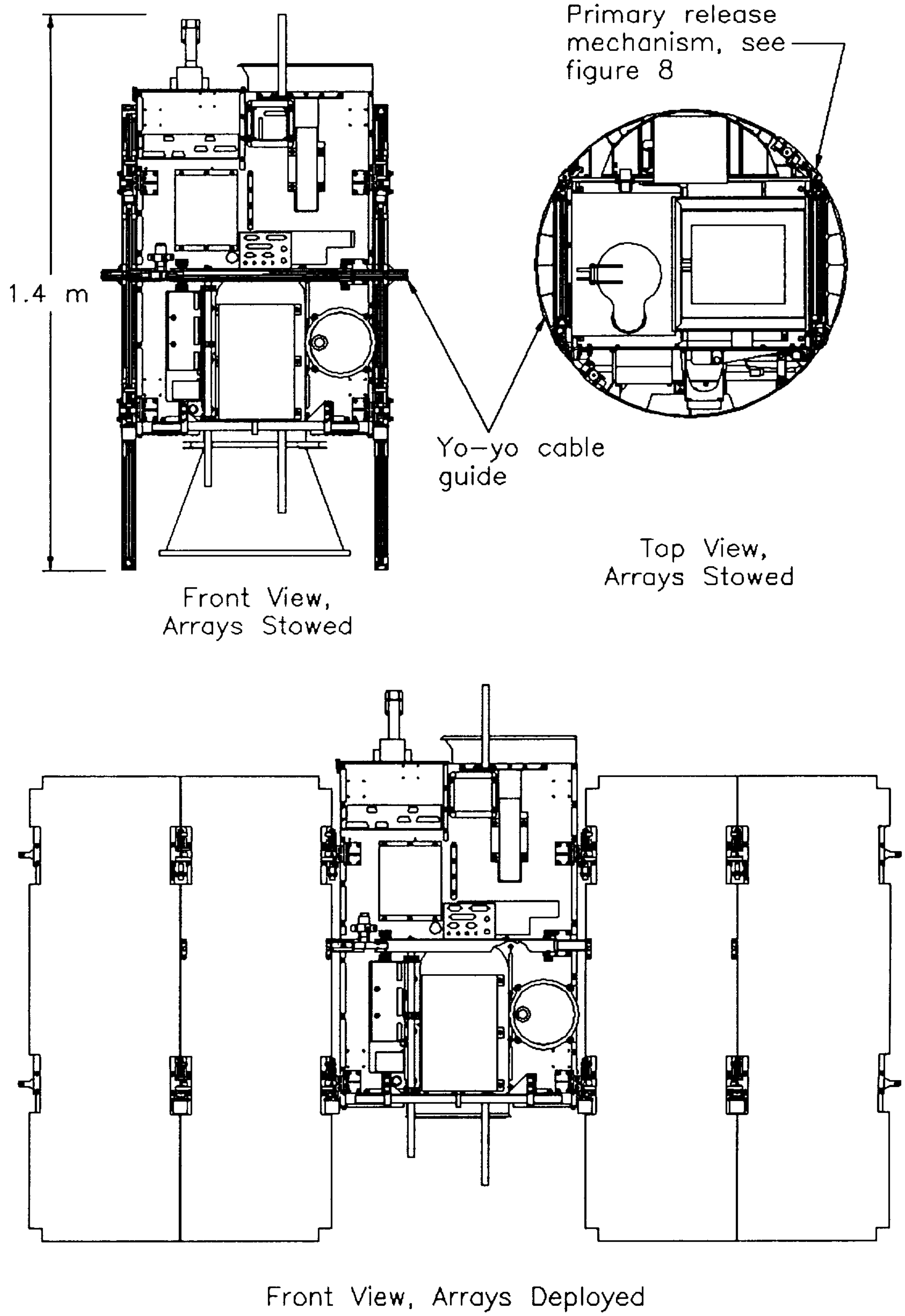

Figure 3: SAMPEX Spacecraft

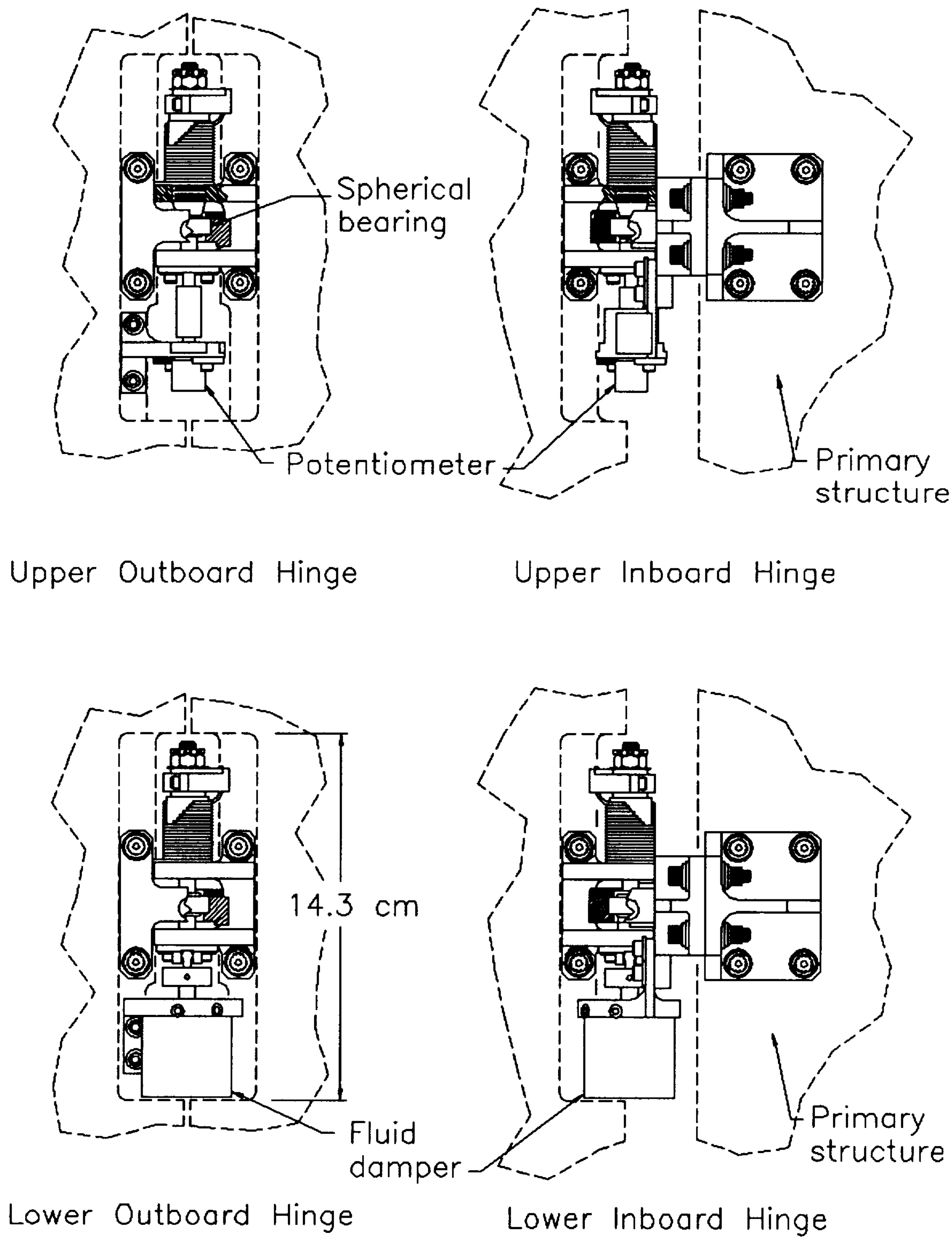

Figure 4: Solar Array Hinges

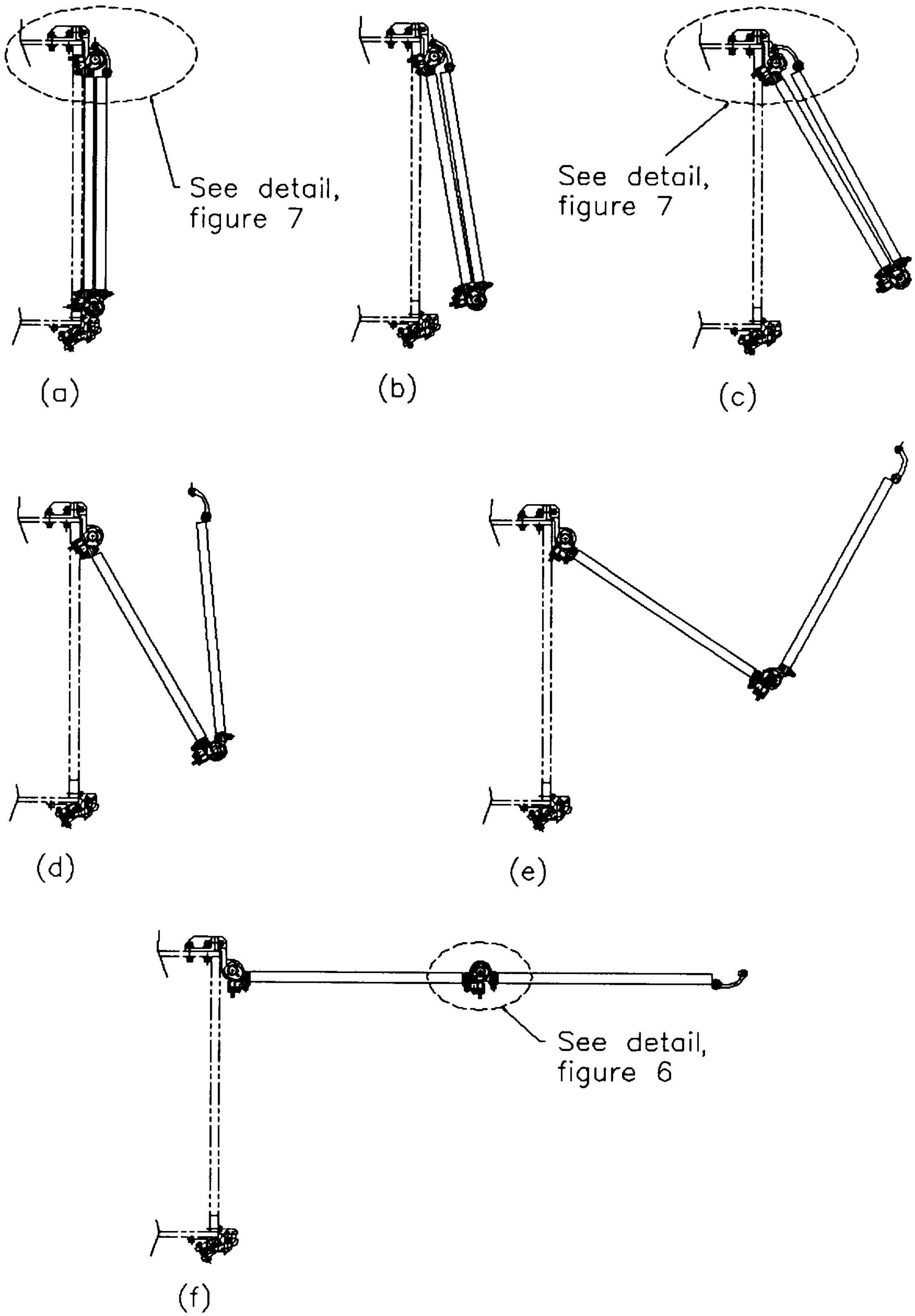

Figure 5: Solar Array Deployment Sequence

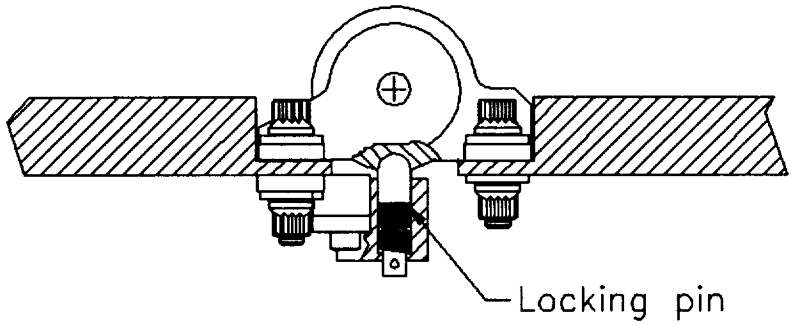

Figure 6: Outboard Hinge Locking Mechanism

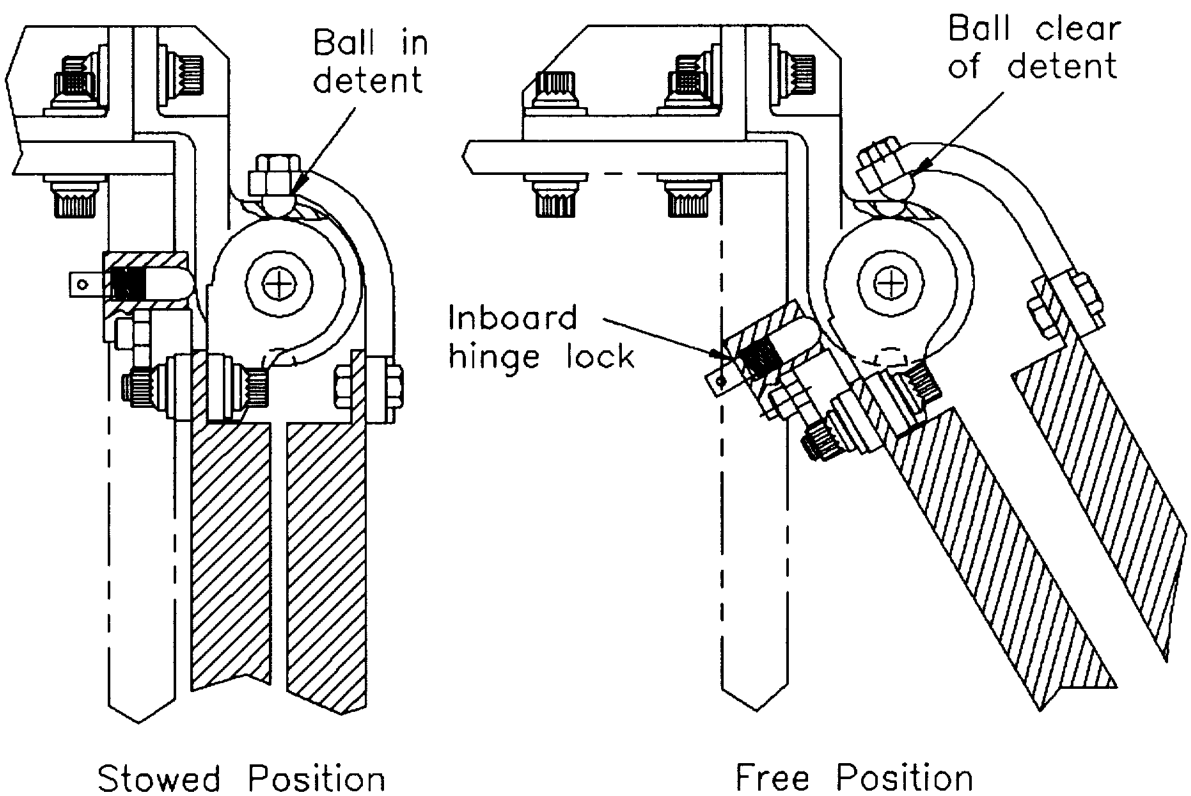

Figure 7: Outboard Panel Release Mechanism

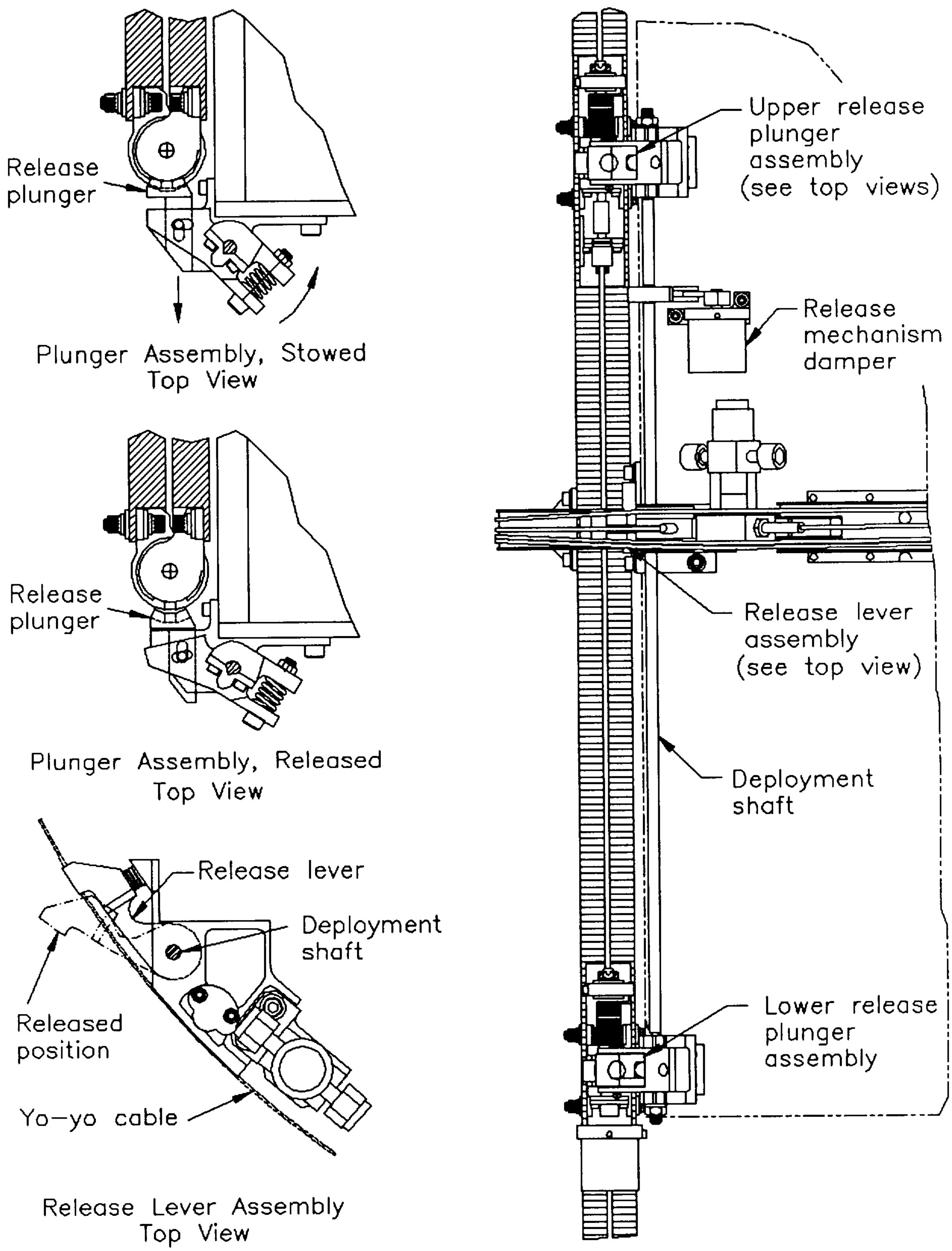

Figure 8: Primary Release Mechanism

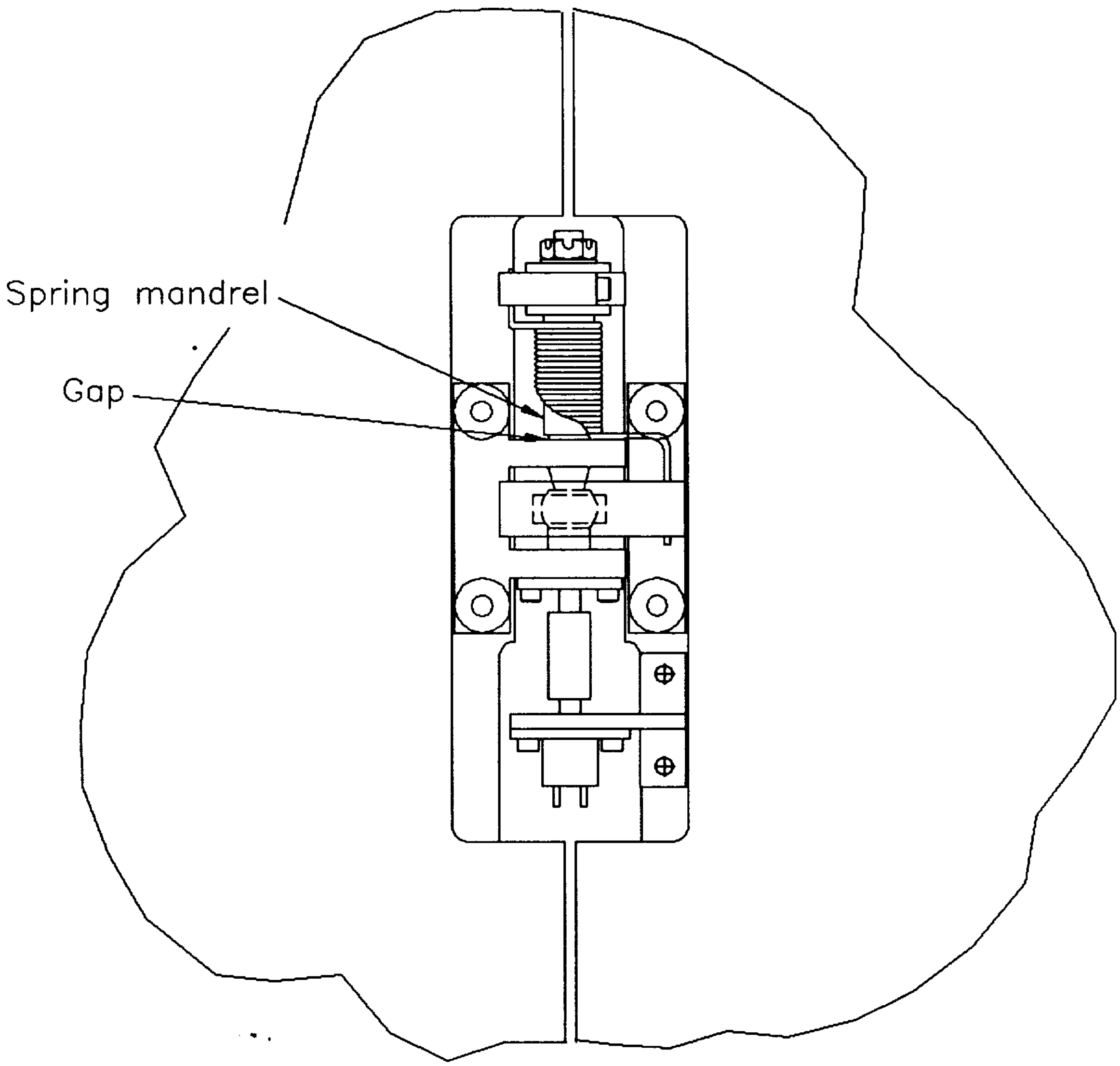

Figure 9: Upper Hinge Problem