

## The Use, Evolution and Lessons Learnt of Deployable Static Solar Array Mechanisms

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### Abstract

This paper focuses on the mechanisms incorporated into SSTL's static deployable arrays; namely the sprung-hinges and hold down and release mechanism (HDRM). Combined, the HDRM and hinges form the hold down release system (HDRS). The deployable static solar array HDRS has been successfully used on several missions, first launched upon the DMC-CFESAT spacecraft in 2007 for a U.S. customer (Figure 1), and later used on DMC-UK2 and EXACTVIEW-1 launched in 2009 and 2012, respectively. The simple, robust and low-cost solution HDRS has been evident in allowing missions to satisfy an ever-increasing power demand, allowing the solar arrays to increase in size and have a preferable sun angle for increased cell efficiency. The system is now being employed on the first mission out of SSTL's U.S. office (SST-US) on the Orbital Test Bed platform. This paper shall cover details of the original design and development program, problems incurred on latter missions, and evolution of the HDRS for the present Orbital Test Bed mission. Both the original development and recent evolutions have taken place in rapid timescales, to satisfy the high-turnaround of SSTL missions.



**Figure 1. DMC-CFESAT platform with deployable solar panels partially deployed**  
(Ph. Credit LeRoy N. Sanchez94)

### Introduction

#### The SSTL approach to Spacecraft Engineering

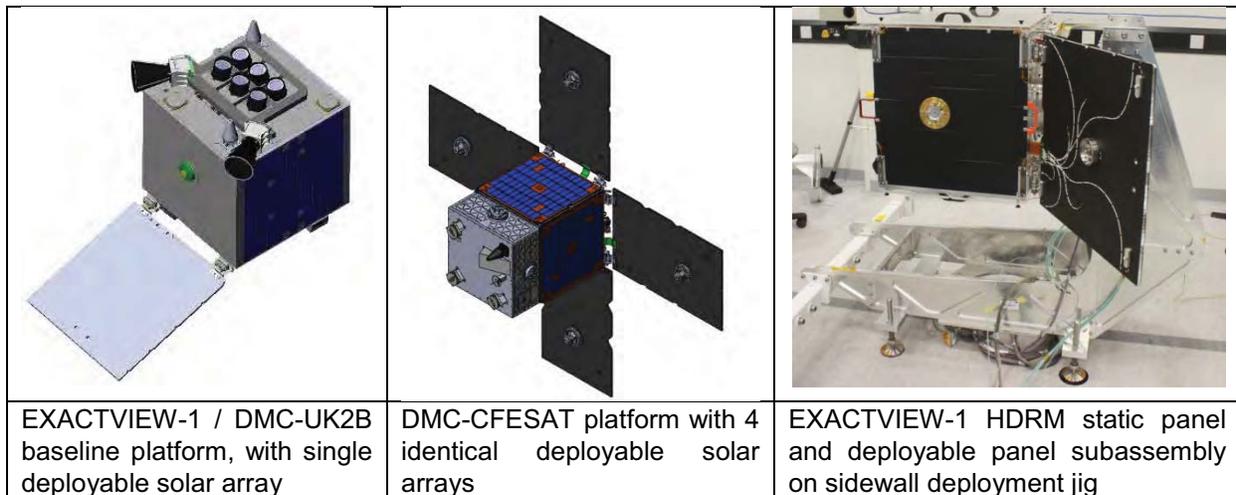
Surrey Satellite Technology Ltd (SSTL) is a unique supplier of low-cost satellites to international customers from its headquarters based near London, UK. SSTL's success within the industry and to meeting the increasing market need for more cost-effective satellite solutions has meant SSTL expanding into the U.S. with a dedicated office (SST-US) with design and assembly integration and test (AIT) facilities. SST-US' first mission, Orbital Test Bed, is now live, with AIT of the spacecraft commencing in summer 2014. SSTL-UK shall be supplying the modules and platform bus, while SST-US provides the payloads and conducts the remaining AIT campaign on the entire spacecraft. SSTL's traditional simple, low-cost platforms used fixed body-mounted solar arrays on all 4 side-panels (non-Earth and Space facing facets) as the satellite's power source. As SSTL's mission capabilities grew and payloads became more power hungry, SSTL had to achieve higher-power capacity platforms. Options to do this included higher-performance solar cells; static deployable solar arrays; and tracking deployable arrays. Static deployable arrays were selected over the other two options for the most popular SSTL-100 class of

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platforms (100 kg) to give a noble compromise between increased power delivery and attributed cost (development and recurring). Static deployable arrays vary from the more complex tracking arrays in as much as the array is deployed from the body of the craft once and to a fixed position which does not change throughout life. This means that the sun angle is improved over the traditional fixed body-mounted panels, but not as optimized as a tracking array which actively follows the sun. The cost and complexity of a tracking array was not deemed necessary for SSTL-100 platforms since the needed power could be achieved via the simpler static-deployable arrays (Figure 2).

SSTL operates uniquely compared to traditional space companies as it makes extensive use of commercial off the shelf (COTS), which gives the benefit that technology is typically advanced ahead of space-rated hardware. SSTL’s matrix structure and use of COTS components both allow for rapid development schedules with the manufacture and test of Qualification Models (QM) taking place early in the project cycle. SSTL invests a focused effort and duration on design and analysis, and spends significant time on thorough test campaigns in order to gain confidence in module developments and fix problems quickly along the way. In this way, SSTL can build and test Flight Models ready for satellite integration quickly from ‘blank-sheet’ design specifications. The deployable static solar arrays and attributed mechanisms demonstrate this approach to spacecraft engineering.

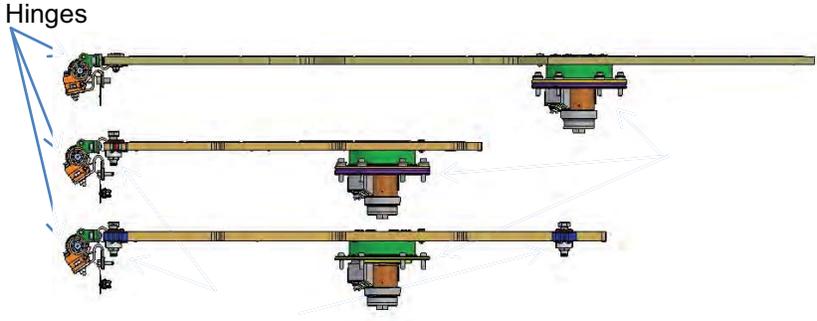


**Figure 2. SSTL-100 platforms with deployable solar panels**

Deployable Static Solar Arrays Overview

Existing SSTL platforms utilize static deployable arrays, with each deployable array having 2 spring-driven hinges, damping interface brackets, and a HDRM. The complete system (hinges, brackets and HDRM) is termed the Hold Down Release System (HDRS). The original HDRS philosophy was to form a generic set of units that could cope with a variety of panel sizes and launch loads to keep future platform non-recurring engineering costs low. The HDRM unit constrains the array against the panel craft during launch, and once actuated, releases the panel. During launch, the HDRM therefore limits the movement of the panel both laterally and axially and thus share the loads with the hinges to prevent their bearings being overloaded. The HDRM incorporates push-off springs to physically release the array from the craft, however, full array deployment is only possible because of the hinges. Once released, the sprung-hinges drive the panel through deployment and a sprung latch locks into position to prevent over travel and recoil upon reaching full deployment.

SSTL has utilized the HDRS successfully on several SSTL-100 platform missions, as captured in Figure 3. Other requirements of the platform types mean that the panel size and position of the HDRM has varied across the missions. Depending upon the position of the HDRM, size of panel, and launcher choice, some platforms also incorporate array snubbers to further control panel displacements during launch.

	<p>DMC-CFESAT (launched 2007) No snubbers. 908-mm panel length. Original HDRM and Hinges</p>
	<p>DMC-UK2 (launched 2009) Lower snubbers only. 482-mm panel length. Original HDRM and Hinges.</p> <p>EXACTVIEW-1 (launched 2012) Lower and upper snubbers. 641-mm panel length. Original HDRM and Hinges.</p> <p>Orbital Test Bed (launch expected 2015) No snubbers. 908-mm panel length. New HDRM evolution and original Hinges.</p>

**Figure 3. Evolution of the SSTL deployable Solar Array**

### History of HDRS Original Development for DMC-CFESAT

#### Original Hinge Development for DMC-CFESAT

Initial concepts were based on the Surrey Rigid Array program, which took place at a similar time and used tape springs for a 2-panel 1-yoke system with ~1.2 x 1.5 m panels for ~1-kW power generation (see Figure 4). The tape springs provide simple low part count mechanisms but due to their nature during 180° rotation, there are points of negative torque margin, and when coupled with the high retarding torques of the flexprint power harness, a positive torque margin cannot be guaranteed. Initial testing proved that this could be overcome by giving the array an initial push-off, however, there was permanent deformation (creases) exhibited in the tape springs and although they still worked, it was deemed not suitable for flight systems which require multiple deployments, so a new hinge concept was undertaken.

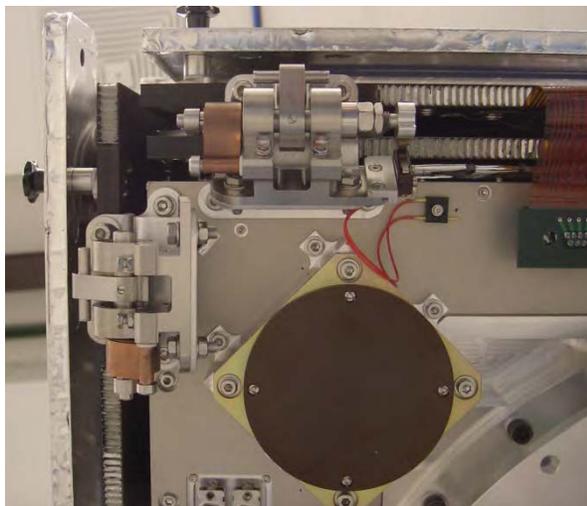


**Figure 4. Surrey Rigid Array in deployed position**

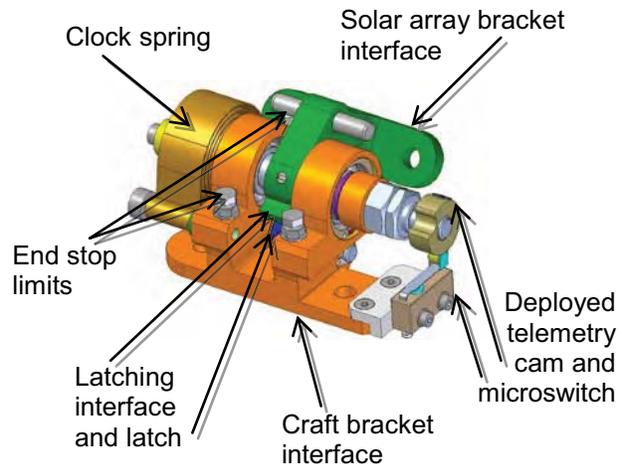
The subsequently developed hinge system design is relatively simple, utilizing two hinges very similar in design; one fixed and one free to slide on its shafts to manage the material thermal expansion mismatch from the CFRP solar panel and the aluminum body of the spacecraft. The constraints of limited volume and mass for the hinges meant that a damped system could not be implemented. Instead, the system

energy needed to be carefully balanced so as not to over-design the parts or cause the brackets to break at the end of deployment latch up. The hinge design is kept generic and latches in the same position, and hence different deployment angles between missions are achieved by simply changing the bracket geometry and not touching the hinge mechanism itself. Each hinge has a simple latch which has a PTFE-coated running surface with a carefully profiled end to give positive locking at the end of deployment. Early versions of these did disengage if enough moment was applied on the hinge so a change was made to bead blast the end surface and after this they never disengaged.

The first design used plain bushings; however, the torque of these was too high and varied considerably over temperature, so this simple and small volume solution was replaced with deep groove radial ball bearings with dry-lubricated PGM-HT cages similar to the ones used in SSTL 10SP reaction wheels. These had a large radial load capacity and very low and consistent running torque (see Figure 5). The initial hinge design used a regular coil torsional spring, but this proved to give unreliable torque performance, and to keep the stresses low in the spring it ended up being quite long and thus not fitting in the design volume on the spacecraft. Instead, a custom beryllium copper clock spring was designed, which had good torque performance through the rotation angle.



DMC-CFESAT hinges insitu with platform



Details of fixed hingewith microswitch

**Figure 5. An overview of heritage deployable array hinges and interface brackets**

The deployable panel went through an extensive unit-level qualification; the same unit also went through spacecraft-level testing doubling some of the typical tests performed. Over 20 deployments were performed at ambient pressure cold / hot and well as a Thermal Vacuum Test (TVT) campaign to remove the air drag on the system. During TVT, one test was performed hot with the maximum system energy and the other cold with one spring removed, the other at low torque and no push-off springs to represent different possible failure modes and a worst case deployment case; all tests were successful.

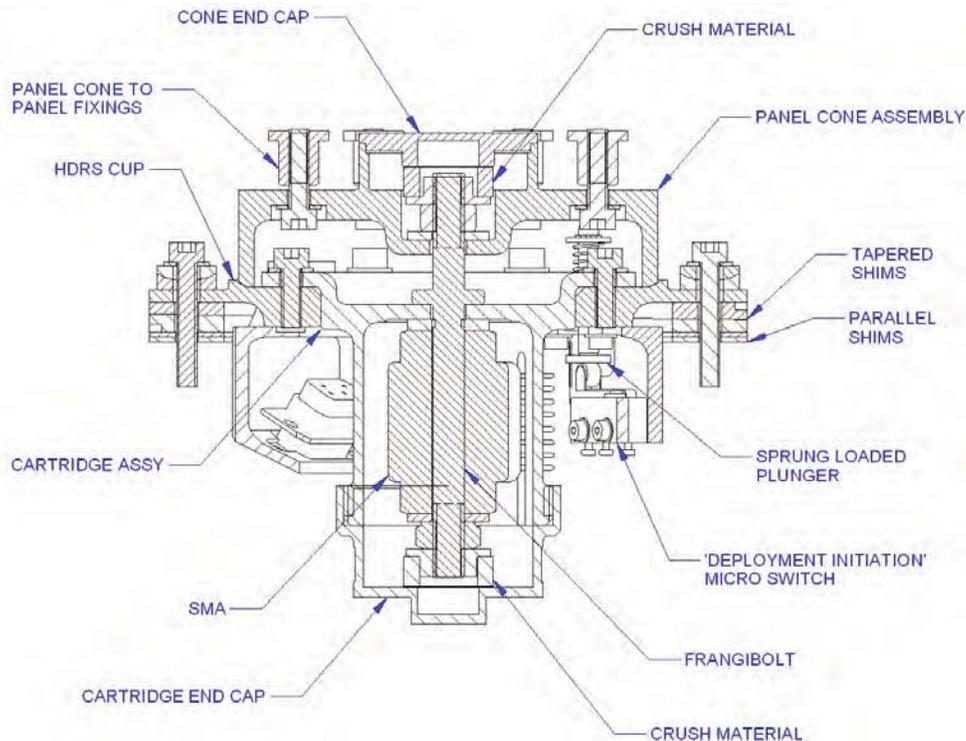
#### Original HDRM Development for DMC-CFESAT

SSTL started investigating the use of Frangibolts (see [www.tiniaerospace.com](http://www.tiniaerospace.com)) in early 2000's for use on its static and multiple panel deployed tracking solar arrays due to the simple design, ease of use, volume, mass and cost of these devices. The development for DMC-CFESAT was particularly mass and volume critical which suited the Frangibolt design. This section discusses the initial Frangi-HDRM (i.e., Frangibolt-based HDRM) development through engineering trials, qualification campaigns, and into flight manufacture and in-orbit deployment of multiple modules. The development was challenging at times; an early release trial on some ground support equipment resulted in a non-fracture of the bolt, however, these problems were overcome to fulfill the successful design qualification and release of multiple devices in-orbit.

The HDRM unit is predominantly bespoke SSTL design in terms of the separation interfaces and housings; however, the central release actuator is a purchased unit (Frangibolt and Shape Memory Alloy (SMA)). The basic principle of the Frangibolt relies on a pre-compressed SMA tube and a titanium bolt with a special notched section which is positioned through the cylinder and restrained at either end. When heated up to a temperature of  $\sim 90^{\circ}\text{C}$ , the SMA undergoes a phase change returning to its original size thus developing enough tensile force to fracture the notched section of the titanium Frangibolt. Redundant heaters wrapped around the SMA are used to apply the heat which only requires unregulated DC power from 10 – 100 V to activate, and with temperature sensors built-in this makes a very simple mechanism with no real moving parts. The Frangi-SMA actuator was selected because it was cost effective, could be reset in-house, and energized via low-power switches.

The Frangibolt is thus designed to fracture in a controlled fashion at a specific tensile load, which occurs because of a stress concentration region via a notch – the frangible portion of the bolt. The notch however limits the load capacity of the Frangibolt and in turn, of the HDRM. In order to ensure that the Frangibolt performs as intended, tight control over the applied preload must be established in order to absorb all clearances so that the subsequent SMA expansion  $\sim 0.1\text{ mm}$  is successful in fracturing the bolt, while not overloading the Frangibolt.

There were a few design requirements on the HDRM over the aforementioned. One requirement was that the parts of the cup-cone should require no match-machining of parts or special alignment impact to the spacecraft structure or arrays. Instead, the cup-cones simply had to ‘bolt-on’ over multiple interfaces which could create up to 0.5 mm of axial and lateral alignment error. To do this, the HDRM was designed to include multiple parallel and tapered shims in the form of a fixed swash plate to allow for parallel and angular misalignments. However, the shimming capacity (parallel and angular) were never used and always set at nominal as the spacecraft assembly errors were minor during the flight builds. The actuator also had to be easily re-set without dismantling the spacecraft or misaligning previously set HDRM and panel hinges. Figure 6 shows a cross section of the Frangi-HDRM assembly.



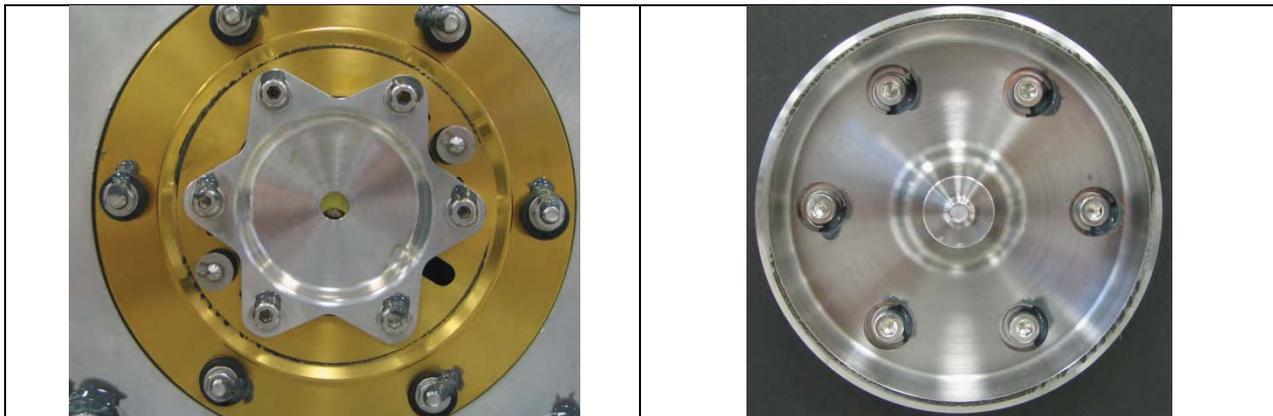
**Figure 6. Cross section of the Hold Down and Release mechanism (HDRM)**

Redundant push-off springs mounted on COTS bushings are configured inside the cone to give the panel an initial separation. A COTS microswitch (~\$2) is housed behind one of these push-off plungers and used to confirm separation of the hold down. A similar microswitch used on the hinges then confirms full deployment and latching. The solar arrays use triple-junction gallium arsenide solar cells which are in close proximity of the HDRM in order to achieve maximum panel coverage. The induced shock from the HDRM bolt fracture was a concern on the micrometer-thick fragile cover glass on the cells, in addition to the hinge's deep groove radial ball bearings and other electronic components within the spacecraft. To reduce the shock, thin film aluminum honeycomb core material was located at each end of the Frangibolt which serves to absorb the shock forces when the bolt fractures.

A ¼-inch Frangibolt was selected for the HDRM and is assembled into a removable cartridge which attaches to the cup. The Frangibolt passes through the cup (craft side) and protrudes through the mating cone (array side) to allow the array to be attached. The cup-cone is thus fastened together via the Frangibolt, and a preload is applied to the front side via torquing a high-tensile nut up to ~5 N-m. The preload of this bolt is critical to prevent the cup-cone slipping and gapping.

In order to prevent cold welding during launch, the Titanium 6Al-4V cup is coated with Titanium Nitride, while the Titanium 6Al-4V cone is left un-plated. To help accommodate the alignment and assembly constraints at spacecraft level, a design trade was made and it was decided that the cup-cone interface did not bear on the cone angle itself as in a typical cup-cone interfaces. Instead, the cup-cone elements bear on flat portions of the corresponding elements with a very small gap (<0.2 mm) to the cone interface so if slipping did occur, it would then engage.

The practical limit of tensile load capability of the Frangibolt is ~6.5 kN. If a larger preload is applied on the cone-side over the maximum in the actuator section, there is risk of creating a gap which could risk non-fracture of the bolt. There was no space in the design to implement a load cell to give accurate preloads so the torque scatter was accommodated in the design. Early testing did lead to further calibration of the test up and an increase in this value to 6.5 N-m as slipping did occur on the engineering qualification model (EQM) vibration test which was evident by visible fretting (Figure 7), but it should be noted that this did not prevent any deployments.



**Figure 7. Engineering Qualification Model cup / cone interfaces showing fretting**

Early on in the design process of the hold downs it was acknowledged that the temperature of the device is critical to its operation. Some work was undertaken to better understand the upper temperature limits of the SMA in the Frangibolt as thermal mission predictions show high temperatures in excess of 80-90°C during early phases of the missions prior to array deployment. This environmental temperature could result in premature array deployment. It was during this testing that one bolt failed to fracture, upon disassembly there was nothing obvious to show why and the actuator in question subsequently successfully fractured a bolt under normal actuation. The full root cause was never clearly identified but

during the detailed investigations several lessons learned were made which were to be later implemented in the flight designs:

- Radii under some of the mating interfaces could have caused a gap if they interfered so tolerances were tightened and 100% optical inspection was made on all Frangibolts and mating parts.
- There was a hexagon broached section in the end of the Frangibolt used to stop rotation with an Allen key while the nut torque was applied during assembly. It was realized that this section could be too long and extend into the working section of the bolt under the main nut and thus changing the local stiffness of the bolt possibly causing elongation of the bolt during extension of the Frangibolt. This was tested and found to be inconclusive although during one tensile test the broached section did fail at a low load and thus it was removed from the design – subsequently changed to a male hex feature.
- Upon disassembly of the setup, a slither of metal was found in the nut and the thread of the bolt was damaged. It was thought that debris could have prevented the nut from being correctly torqued up thus insignificant preload being applied which when the SMA went through its elongation it was not restrained and could extend / gap.

The latter concern coupled with the fretting required indicated that accurate knowledge of the preload in the frangible section of the Frangibolt was needed and so a new setup (Figure 8) was defined in a tensile test machine. Here a known load  $\sim 7$  kN is applied and then a locking nut installed to 1 N-m thus guaranteeing the preload in the assembly is greater than the newly defined 6.5 kN required on the front.

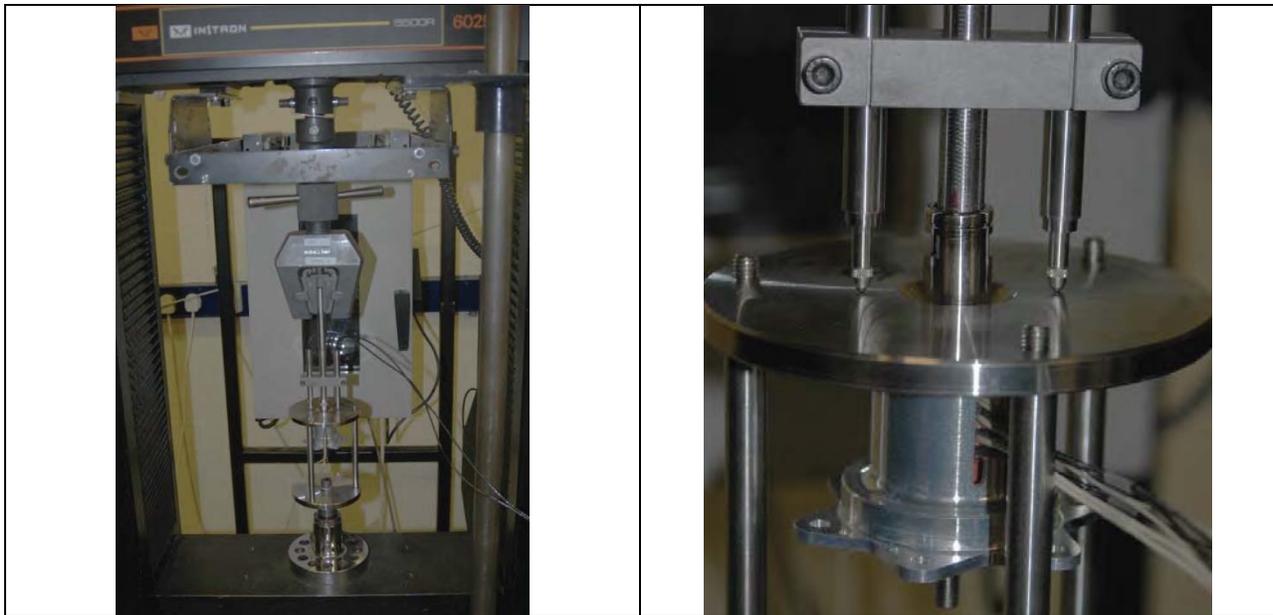


Figure 8. Frangibolt preloading set up

### Challenges Experienced with HDRM Actuator Selection on Later SSTL-100 Missions

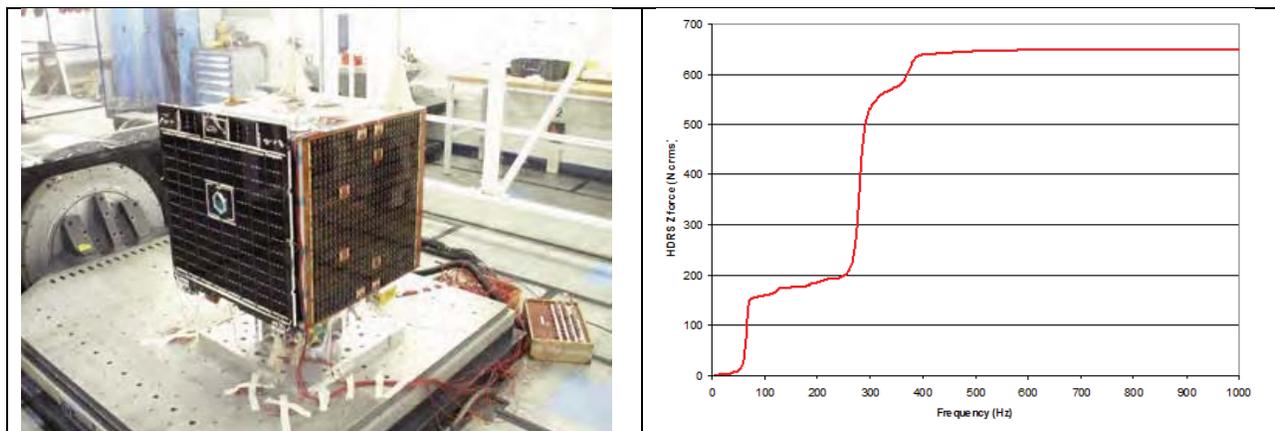
#### DMC-UK2

The HDRS successfully deployed all four DMC-CFESAT solar arrays in orbit following launch in 2007. Following this success, the HDRS was employed on another SSTL-100 platform type, DMC-UK2. For this mission, the deployable panel was significantly smaller than the original DMC-CFESAT (Figure 3), therefore direct use of the HDRS was deemed feasible. While the HDRS had sufficient load capacity to cope with the panel's induced forces during launch, the launch environment was harsher and solar array panel deflections were a concern. In order to avoid possible damage to the solar cells, then snubbers

were incorporated into the design to limit the array deflection. DMC-UK2's single solar array deployed in-orbit in 2009 - a successful application of the heritage HDRS design.

#### Problems Experienced During EXACTVIEW-1 Environmental Test Campaign

The load capacity was suitable for the baseline two platforms – DMC-CFESAT & DMC-UK2 – however problems arose during the third platform (EXACTVIEW-1). For this mission, the panel size grew. While still smaller than the original DMC-CFESAT panels, the relative distance from the hold down to the top of the panel increased and a different launcher was selected late in the program, which imparted higher loads into the structure. Subsequent analysis showed a coupling of a spacecraft mode at ~280 Hz (Figure 9) with the deployable panel so two additional upper corner snubbers were incorporated into the design, which helped, but the design margins at the hold down against gapping were low to negative. As this was late into the spacecraft EVT program with all hardware manufactured and built onto the spacecraft, the spacecraft was taken for a low-level vibration transmissibility test which confirmed the analysis / negative margins and lead to a speedy re-design.



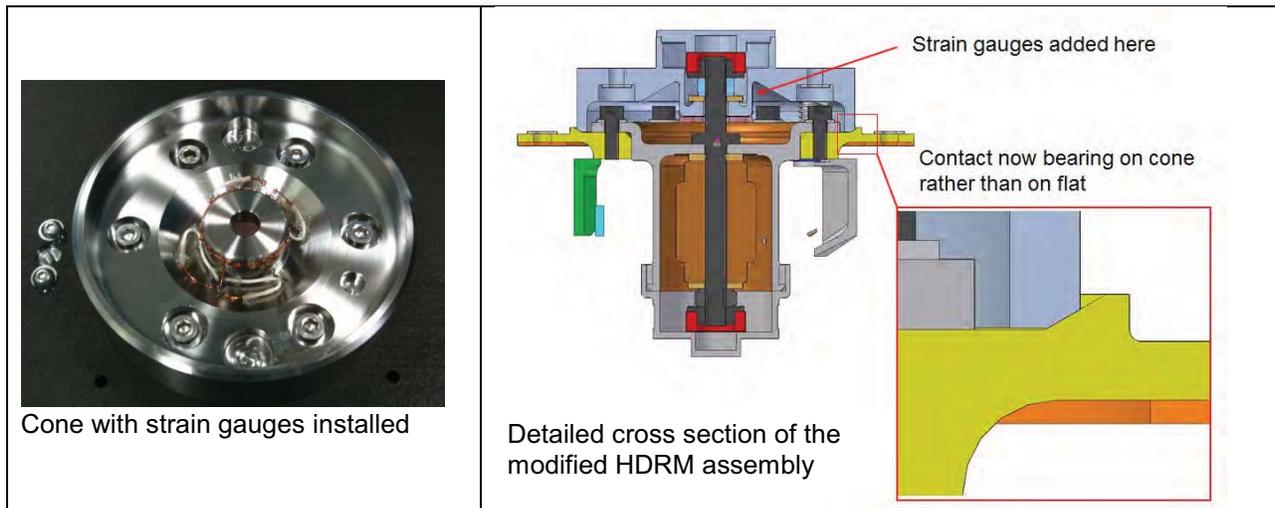
**Figure 9. EXACTVIEW-1 at vibration test with HDRM sine loading shown**

All the interfaces were fixed and there was no time to procure / implement another hold down device with a higher load capacity, so the preloading of the nut was investigated as preloading either side of the actuator using a torque wrench is known to have a large scatter. The torque scatter means that the resulting preload in the bolted joint could vary by as much as  $\pm 30\%$  due to friction on the thread and washer. In order not to exceed the Frangibolt's 6.5-kN load capacity, a nominal preload of 4.55 kN would be necessary and at the lower end of the scatter would result in a preload of 3.19 kN. At this lowest tolerance preload, the resulting margin of safety against HDRM gapping was negative (gapping is the ability of the mating release interfaces to separate temporarily), this could lead to:

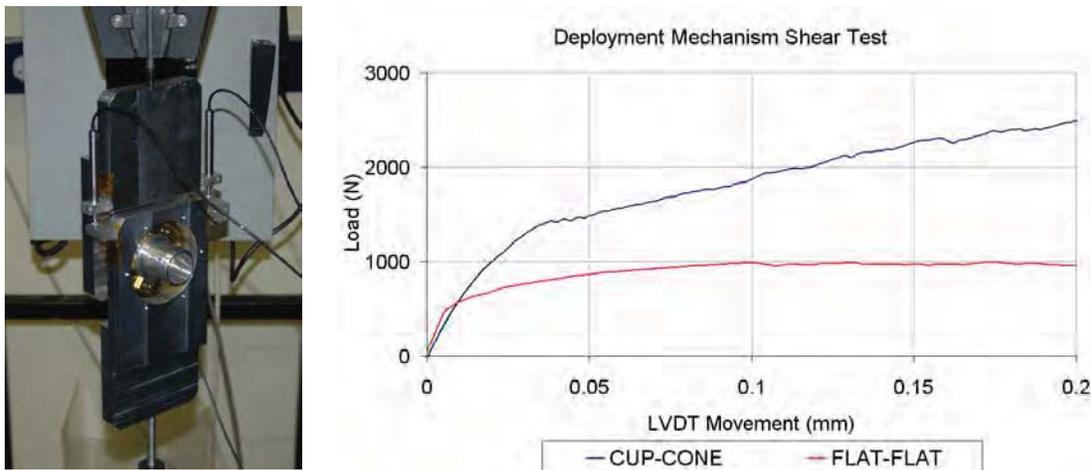
- HDRM pushing additional loads onto the hinges, and the hinge bearings being overloaded - thus increasing friction and risk of deployment failure
- During dynamic vibrations, fretting on the release interface could occur, which could lead to cold welding preventing the panel from deploying
- Gapping hold down portions could in turn induce additional lateral loads and bending moments on the Frangibolt which could cause it to fracture prematurely and deploy during launch.

It was not possible to implement a load cell or load washer into the design so the HDRM housing was re-designed and instrumented with strain gauges to monitor deflection to better determine the preload reducing the scatter to  $\pm 10\%$  (see Figure 10). As a result of this method, the nominal preload could be increased to 6 kN, however, this alone was only just sufficient to avoid the potential for gapping so a further change was made to achieve more favorable margins of safety by employing the true cup-cone interface instead of the previous flat-on-flat friction contact used on CFE-SAT and DMC-UK2. Shear Tests

were conducted, that showed the true cup-cone interface could support approximately 25% increased lateral forces for the same preload than a flat-on-flat friction contact (see Figure 11).



**Figure 10. EXACTVIEW-1 HDRM Modifications**



**Figure 11. Deployment interface shear test setup comparing flat-flat interface with cup-cone interface, for a nominal preload of 6075 N**

The cup-cone interface and reduced preload scatter was sufficient to change the baseline negative design margins to positive for the EXACTVIEW-1 mission, whose panel successfully deployed in-orbit in 2012. While the campaign was successful, it highlighted the design's load capacity limitations and the criticality of the preload method. The man effort now associated with using this design due to the different assembly and calibration steps had increased the overall unit price of the system. When also considering the increased analysis non-recurring effort which would be required on each future program due to the Frangi-HDRM's sensitivities, a recommendation was made by the material review board to implement a different higher load capacity solution on the next mission. The material review board's analysis suggested a preload greater than 12 kN was required to give confident margins.

It is also worth noting that DMC-UK2 and EXACTVIEW-1 platforms provided the benefit over the DMC-CFESAT craft such that the HDRM and hinges are mounted directly to a single fixed body-mounted panel interface, allowing the HDRS, static and deployable panels to be assembled and tested separate to the rest of the satellite structure (see Figure 2). Whereas, on DMC-CFESAT, the hinges mount to one spacecraft interface, and the HDRM to another driven by the craft's need for an alternative payload bay

arrangement. The stacked tolerances on the multi-interface of DMC-CFESAT means that after craft de-stack and re-stack operation, the hinges and HDRM need realigning carefully, whereas DMC-UK2 and EXACTVIEW-1 can remain as a separate single subassembly, saving time and reducing variability.

### **HDRM Evolution and Actuator Change**

#### Hold Down Actuator Change

Following on from the successful in-orbit deployment of EXACTVIEW-1 solar panel, there were substantial lessons learned regarding the heritage hold down actuator choice. Firstly, the load capacity proved more restricting than previously thought to the extent where a launcher change caused the hold down to have negative margins on gapping and slippage. Secondly, the attraction of being able to perform in-house refurbishment via frangible bolt change after hard deployments was offset by problems experienced during the setup whereby the exact preload achieved was questionable. This latter issue could be resolved for future use by better methodology and with additional instrumentation – for example by monitoring the deflection of the hold down cup body.

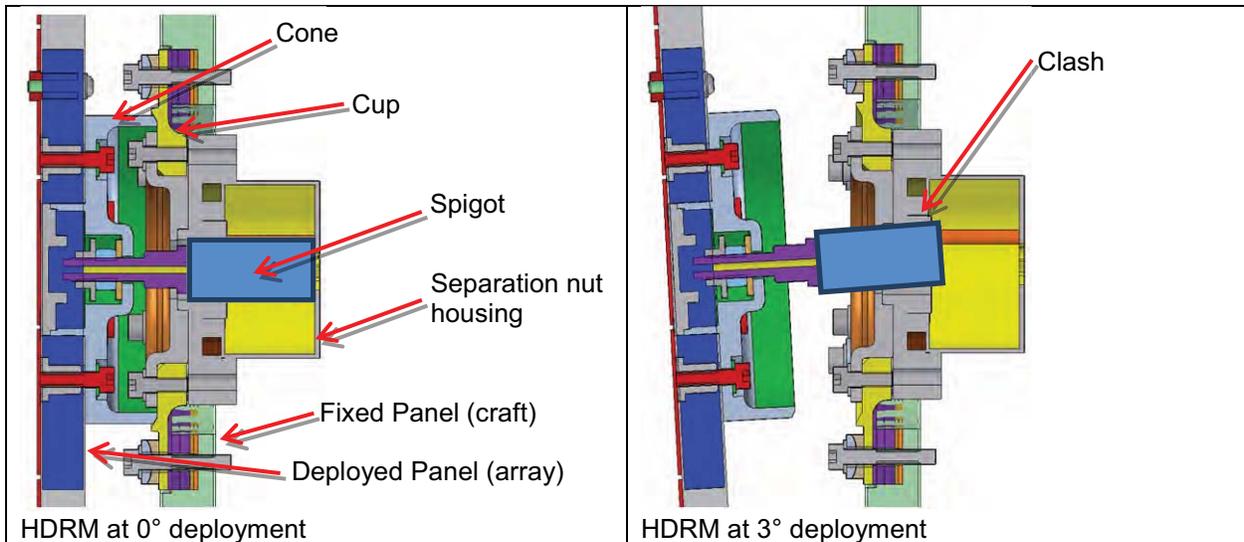
However, the decision was taken that a much higher load capacity was needed in order to avoid future problems similar to EXACTVIEW-1, such that in terms of HDRM loading, the launcher selection could largely be ignored and the need for detailed analysis would almost disappear. Therefore, upon signing the contract for the SST-US mission Orbital Test Bed, the decision was taken to change the hold down actuator. SSTL could have alternatively changed other design features of the HDRM, such as the cup-cone angle, however, the actuator change and experience with a viable alternative proved favorable.

Another actuator market study was conducted, but revealed that the availability was very constrained based on the recurring cost budget permitted. Due to the criticality of the application, only established actuators were considered and the possibility of new developments in-house or external was ignored. This market study identified two possible solutions: either select a larger size in the frangible bolt and SMA family from the existing selection (1/4") or change to a non-explosive separation nut. The larger frangible-bolt was rejected since the corresponding SMA body would impart a significant increase in volume of the HDRM unit. Thus the non-explosive separation nut (see [www.neaelectronics.com](http://www.neaelectronics.com)) was chosen. This was also based on good experience gained with the separation nut on the SSTL NigeriaSat-2 mission which used these to support a deployable optical payload bench.

The separation nut actuator works on the basis that a spigot shaft is constrained by a multi-piece nut, which is held radially by a wrapped spring and fuse wire. When the fuse wire burns, the released spring allows the nut to separate and release the spigot. The separation nut housing is fastened to the cup portion of the HDRM in place of the previous SMA and the mating cone is fastened by screwing a stud into the separation nut's spigot. Therefore, the spigot shaft is merely released in contrast to the fracturing nature of the frangible bolt in the previous system.

For a similar size separation nut to the previous SMA, the load capacity increased threefold, with a rated release load of >33 kN. This allows a Sep-Nut-HDRM to be assembled efficiently without any instrumentation by torqueing alone. A nominal preload of 22 kN is sufficient to ensure that the achieved preload always exceeds the 12-kN minimum requirement and is well within the actuator's load capacity, even factoring in  $\pm 30\%$  torque scatter. The significant load capacity allows the Sep-Nut-HDRM to remain a generic solution for many more applications supporting larger panels/payloads over higher launch loads. The requirements for the HDRM development are summarized in Table 1.

Due to the hinged deployment nature of the solar arrays, once released, the spigot tip's movement describes an arc and thus the spigot tip would catch and snag on the bore of the cup. This was not a problem on the existing SSTL separation nut application on the deployable optical payload bench, since the bench was released parallel to the static panel, as opposed to being hinged from one edge. Any snagging of the spigot tip on the cup could result in partial deployment and corresponding mission degradation via loss of power.



**Figure 12. Separation Nut and HDRM arrangement and arcing effect of panel deployment profile**

**Table 1. HDRM and Actuator Requirements**

Parameter	Magnitude	Notes
Hold down actuator load capacity	Minimum load capacity 15 kN	The selected Separation Nut is rated to loads <33 kN, hence providing margin even given $\pm 30\%$ torque scatter
HDRM mass	< 1.0 kg	
Operating voltage	20 – 33 VDC	
Operating current	< 3 A	Traditionally actuated by a low-power switch.
Survival temperature	-70°C to +80°C	Based on preliminary mission analysis.
Low Shock	<< 3000 g	
Physical Envelope	Compatible with heritage interfaces	The redesigned HDRM shall have the same spacecraft interfaces to both the static and deployable panels. The overall volume of the HDRM shall be largely unchanged, and compatible with existing craft layouts.

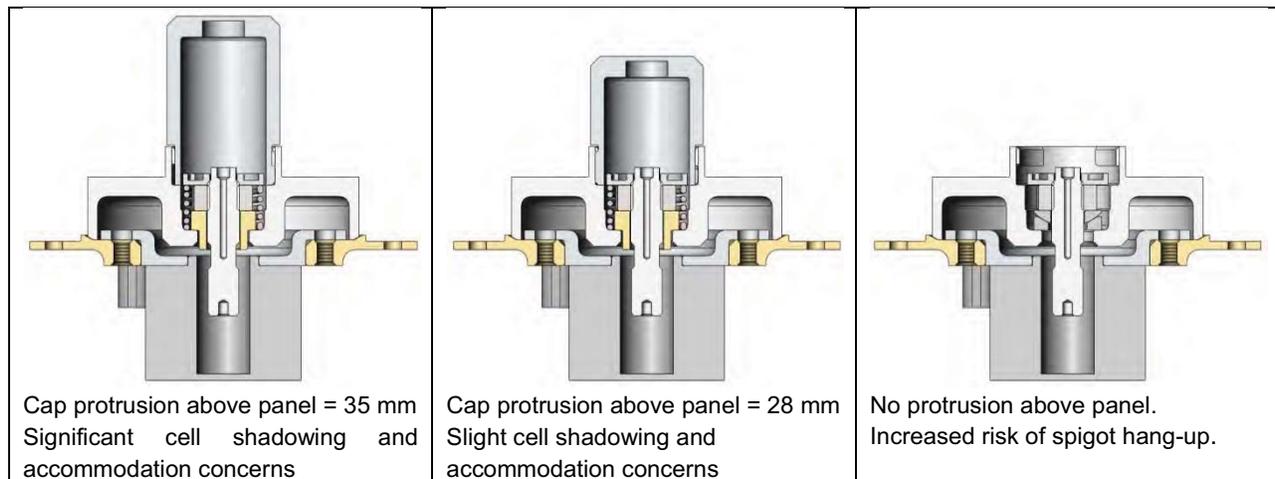
Due to the complex way in which the separation nut releases the spigot, the separation nut actuator provides the limitation that it cannot be refurbished in-house and needs to be sent back to the supplier, in contrast to the original actuator where the frangible bolt could merely be replaced. This external refurbishment limitation was outweighed by the confidence obtained through SSTL's existing use of the separation nut and large margin on load capacity. The other limitation was that the separation nut actuator choice was also ITAR-restricted which limits the applications and slows the procurement process. However, there is a significant lack of European cost-effective, high-load capacity, low-shock and proven hold down actuators, so there was no alternative at this time.

#### Retraction and Non-Retracting Mechanisms

Due to the spigot arcing hang-up effect described in Figure 12, it was foreseen necessary that the released spigot shall be withdrawn through the cup bore, prior to significant panel movement in order to prevent the possibility of hang-up. The most elegant way to do this is to incorporate a retraction spring which pulls the released spigot through the deployable panel immediately upon deployment to ensure that the spigot is out of the way and will not clash. This is done using a compression spring mounted between the cone and stud to act as a spigot retraction spring. The nature of the NEA's function means that the deployment is low-shock; however, the retracting spring on the spigot will impart some low-magnitude shock which shall be absorbed by a simple O-ring, instead of the previous honeycomb core material.

Initially, it was conceived that a retraction cap would be integrated onto the cone assembly to 'catch' the released spigot and stud shaft during deployment. Figure 13 shows that the retraction cap for a full-retraction system would protrude significantly above the panel surface with two notable drawbacks: the likelihood of cell-shadowing in-orbit and the restriction of launch vehicle accommodation due to growth of craft footprint. The cell shadowing effect was deemed low risk since the small diameter (30 mm) protrusion above the array surface would only cause significant shadowing at large angles of incidence (60°), at which point the cell's efficiency is significantly reduced and little power produced anyway.

As an alternative to the retraction spring, it was also conceived feasible that the spigot could merely be provided with a degree of freedom (rather than rigidly constrained to the cone), and thus without retraction, the spigot would clash with the cup during deployment but rotate and drag along the surface, rather than snagging and preventing deployment. Therefore, 3 configurations were envisaged: full retraction, partial retraction, and non-retracting (see Figure 13). The non-retracting variant has the advantage that the cone can be kept low profile without a retraction cap, but the mechanism details are less elegant. The degree of freedom is provided by a spherical washer interface, and the amount of spigot deflection needed to drag across the cup is <math><3^\circ</math>.



**Figure 13. Partial-retraction, Full-retraction and Non-retraction variants of HDRM**

#### HDRM Variant Breadboard Engineering Testing

The 3 HDRM design variants were taken forward to breadboard engineering testing in order to assess which one to select and evolve to the final mechanism. For this testing, a simple separation nut spigot was used instead of a complete separation nut actuator, and rapid-prototype additive-layer manufactured plastic parts represent the cup, cone and retraction cap. This low-cost and quickly compiled test setup was invaluable for providing some initial indications of design success.

All testing was done on the already existing sidewall deployment jig, with qualification model (QM) hinges and panels. Two sized panels were used: the smaller DMC-UK2 QM with short offset between the hinge rotation axis and HDRM position, worsening the arcing effect; and the larger DMC-CFESAT QM panel. The deployment tests were conducted in two stages: initially a 'light' retraction spring was used ( $k = 0.189 \text{ N/mm}$ ) to reduce risk of damage to the plastic; then the testing was repeated with a 'hard' spring (the intended design,  $k = 0.98 \text{ N/mm}$ ). To give confidence, three deployment tests were done for each variant across several deployment scenarios, varying the hinge spring torque and the push-off spring presence to mimic different failure scenarios.

The breadboard engineering tests provided the following conclusions:

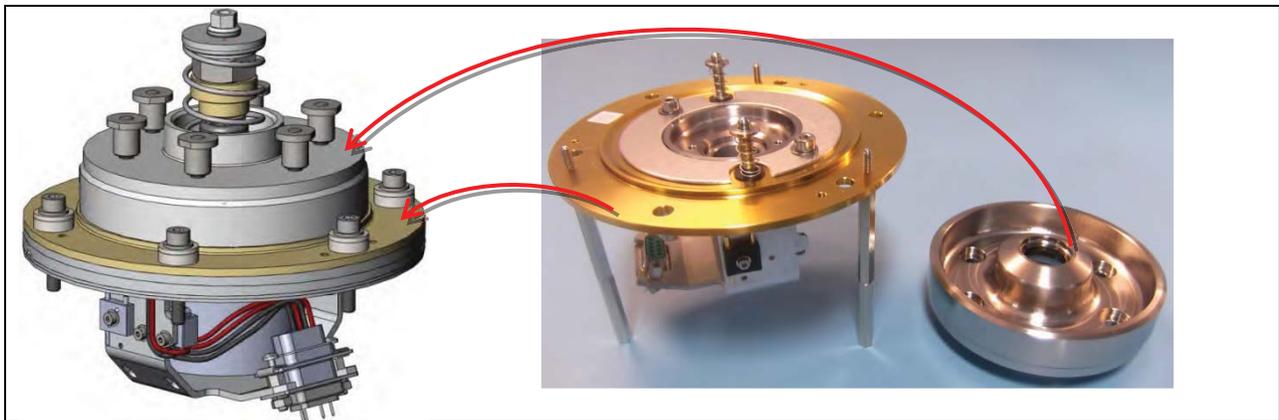
1. Deployment was successful in all attempts for all variants for all feasible scenarios and the choice between full, partial and no retraction does not influence the reliability of deployments.
2. Retraction spring energy imparts notable energy into the panel deployment – the effect on hinge bracket energy dissipation shall need further work.
3. The time for full retraction with the hard retraction spring is 6 ms corresponding to 0.3° panel deployment. This provides margin on the spigot-cone collision which occurs at 1° deployment.
4. The light retraction spring still provides fast enough retraction time (12 ms to full); however under 1g conditions, the spring was so light that the spigot would droop under its own weight once released, and this meant that it could be heard to make contact with the cone during deployment.
5. Spigot-cone collision does not prevent deployment provided the spigot has >2° free rotation.

This test campaign has demonstrated that all variants (partial-, full- and non-retraction HDRMs) are suitable to ensure reliable deployment of both the DMC-UK2 and DMC-CFESAT solar arrays.

#### Design Finalization and Qualification Plan

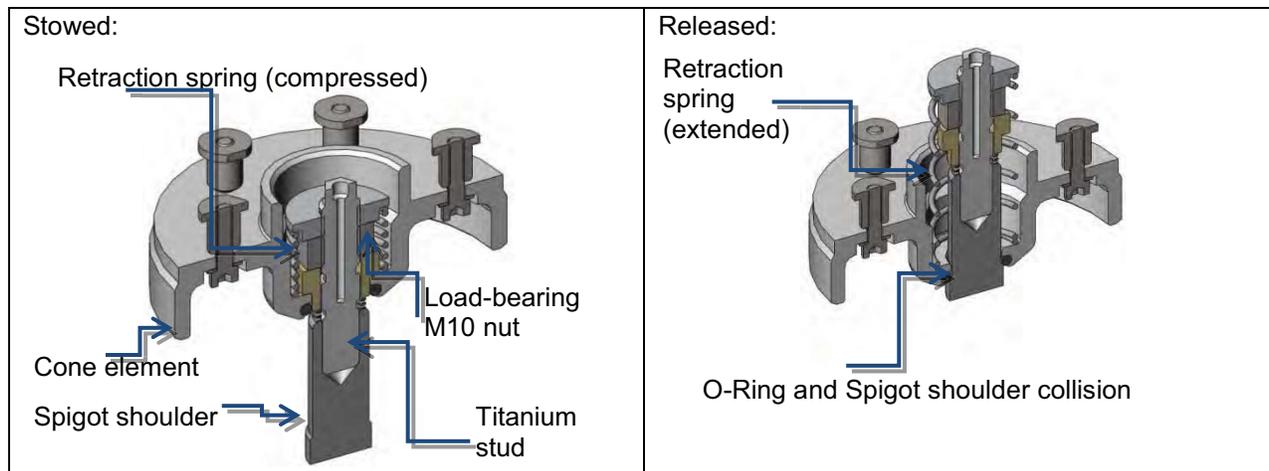
In true SSTL-style, the early breadboard engineering test campaigns gave opportunity to make observations and tailor the design to final design quickly, with final flight design drawings being released 5 months after HDRM evolution kick-off. It was agreed that the full-retraction variant was most attractive from a mechanism-reliability perspective; this could also be done without the need for the undesirable rigid retraction cap. Instead, the natural shoulder available on the separation nut spigot could be used to constrain the spigot in the cone to prevent full withdrawal. In this way, the HDRM retained the low-profile stowed volume of the non-retracting variant.

An O-ring on the underside of the cone is used to dampen the impact of the spigot shoulder, to permit the spigot to be reused to refurbishment after hard-deployments. In addition, the arrangement of the spigot, stud and retraction spring mean that if the retraction spring were to fail, the assembly allows the spigot to retain degrees of freedom, which have been shown to provide successful deployment – giving additional confidence to the design. The final design is shown in Figure 14 and Figure 15.

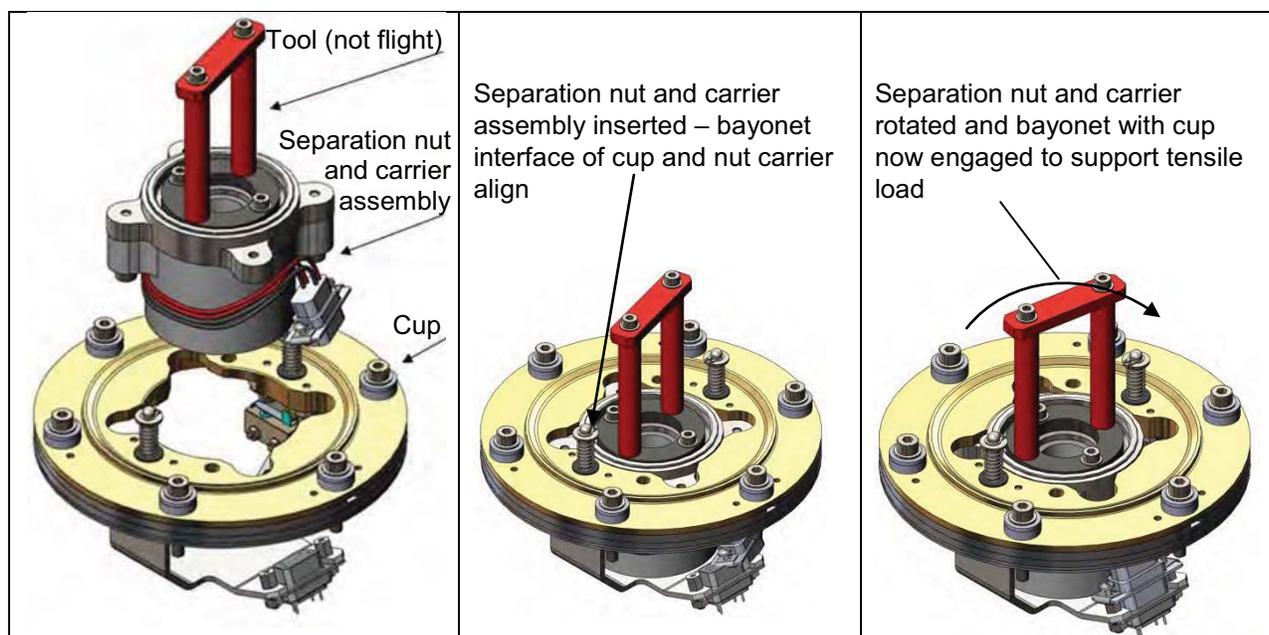


**Figure 14. Sep-Nut-HDRM: Left – CAD, Right – partially-built hardware (cup assembly and cone)**

In order to remove and replace the separation nut on the spacecraft for hard deployments during AIT, the actuator must be inserted from the ‘front’ of the cup only. In this configuration, the stress analysis of the design showed that the fasteners initially used to hold the actuator body to the cup was overstressed in tension, and enough larger fasteners to cope with the stressing could not be accommodated. Therefore a novel solution was incorporated to offload the fasteners by using a bayonet interface between the actuator carrier and the cup as shown in Figure 16. In this way, 2 small fasteners are merely used to stop the actuator falling out upon assembly – they do not experience any load, which instead is transferred directly through the cup body.



**Figure 15. Sep-Nut-HDRM Retraction Mechanism Overview**



**Figure 16. Separation nut carrier and cup bayonet load transfer**

### Conclusions

The revised Sep-Nut-HDRM assembly weighs <1 kg and supports up to 33-kN preload. The original Frangi-HDRM design weighed 750 g and could only support 7-kN preload. The significant increase in load capacity far outweighs the small increase in mass; and allows for a very generic HDRM design for future applications such as larger/multi-panel arrays or deployable boom/antenna. The QM Sep-Nut-HDRM has been built and is about to undergo hard-deployment testing. The decision has been taken that vibration testing is not necessary since the QM HDRM design load capacity is far larger than the intended use, and thus the assembly does not experience any live load above its own preload. Instead, only thermal hard deployment testing shall be undertaken on the QM HDRM to prove the design is fit for flight while also covering the torque scatter possibility to avoid instrumentation and reduce recurring flight build costs. Subsequently, 4 FM HDRMs shall be built and integrated onto the craft during summer 2014.