THE CASSINI MAIN ENGINE ASSEMBLY COVER MECHANISM

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ABSTRACT

This paper describes a micrometeroid protection system for the main engines of the Cassini spacecraft. The engine Cover Assembly is a deployable/restorable half sphere of multilayer insulation mounted to an articulatable frame over 2 meters (7 feet) in diameter. The Cover folds into a compact wedge only 25 cm (1 O inches) at its maximum thickness. The micrometeroid environment and typical protection methods are described as well as the design details and development problems of the Cover Mechanism Assembly.

INTRODUCTION

Cassini is thought by many to be one of the last immense interplanetary spacecraft the United States will produce. Standing over 6.7 meters tall and weighing over 5,600 kilograms, Cassini will orbit Saturn for a four year tour of the rings and moons (Figure 1), Cassini is an international collaboration between the National Aeronautics and Space Administration (NASA, supplying the Orbiter), Agenzia Spaziale Italiana (ASI, supplying the High Gain Antenna) and the European Space Agency (ESA, supplying the Huygens Titan Probe). In all, the scientists and engineers on the Cassini team come from 16 European countries and 33 states of the U.S. Using its main engines and numerous gravity assists, each orbit around the planet will be unique to allow Cassini to perform close-up observations of many of Saturn's satellites. Due to the criticality of the main engines for this mission, the Main Engine Assembly (MEA) is block redundant, consisting of dual engines, gimbals and linear actuators (Figure 2). Late in the development of the spacecraft, it was determined that these high performance engines were particularly sensitive to damage from the micrometeroid environment of interplanetary space. Thus, a protection scheme had to be implemented for the MEA after the entire Cassini spacecraft had been designed and built. This paper presents the design of the MEA Cover Mechanism Assembly and some of the more interesting problems encountered during its development.

THE MICROMETEROID ENVIRONMENT

Interplanetary spacecraft have been designed with micrometeroid protection features since the early 1970s. When Galileo was built in the early 1980s, protection was incorporated according to a micrometeroid fluence model based on data collected by Pioneer 10 and 11, Helios 1 and flux measurements near Earth (spacecraft and lunar craters). For Cassini, the model had been updated with data from the Galileo and Ulysses spacecraft. The interplanetary environment is considered to have a nearly omnidirectional flux of micrometeroids ranging in mass from 10-13 grams up to 10-4

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grams, with an average impact velocity of 20 kilometers per second (44,700 mph). The concentration of these high speed particles is highest at low heliocentric distances and falls off rapidly outside of 2 astronomical units (AU). Essentially, protection from micrometeroids is required between the Sun and the asteroid belt, and becomes less important as you approach the outer planets (Reference 1).

The protection methods are usually quite simple and fall into two classes. Provisions must be made either for a single, thick *first surface protection* or for a thinner *second surface protection* which in itself is shielded at a distance by a first surface. This latter method is the preferred design because it is a lower mass solution. The key is providing the spacing between the first and second surfaces. A typical example is a mission critical electronics bay, exposed to space. The electronics box enclosure must have a wall thickness of 5.8 mm (.23 inch) of aluminum if it is a single first surface protection for micrometeroids. However, the aluminum enclosure can be a more reasonable 1.6 mm (.06 inch) thick second surface if it is shielded by multilayer insulation (MLI) spaced off at least 6.1 cm (2.4 inch) from the first surface. The idea here is to break up these small high velocity particles with the first surface into even smaller particles that are stopped by the second surface. A volume of the first surface and the micrometeroids are partially vaporized, and the “shotgun blast” is sprayed against the second protecting surface. The density of the shielding materials, the field of view to space and the sensitivity of an item to damage are the primary parameters that determine the required thicknesses. MLI thermal blankets, usually already required for temperature control, are spaced off with thin wall Mylar or fiberglass spacers and fastened by Kapton tape. Most of the spacecraft is provided adequate micrometeroid protection by the thermal blanketing alone.

The sensitivity of Cassini's main engines was uncovered late in the program, The engine combustion chamber is thick walled and is protected by its limited exposure to space and the density of its construction. However, the engine nozzles are thin walled columbium with a crucial .08 mm (.003 inch) thick disilicide coating. This coating protects the base columbium material from oxidation. It is this coating that is sensitive to micrometeroid damage, since a single small damaged area in a strategic location along the nozzle can result in a “burn through” when the engine is fired. The micrometeroid protection of these nozzles was complicated by the fact that cooling is provided by the field of view to space Simply covering the nozzle exterior with a thermal blanket wouldn’t be acceptable. A deployable shield was required that could protect the entire nozzle yet provide a full view of space for cooling and plume exhaust.

PRIMARY DESIGN REQUIREMENTS

One of the most fundamental design requirements of the micrometeroid protection mechanism for Cassini's main engines was that the retrofit could not adversely impact the integration of the Radioisotope Thermoelectric Generators (RTGs) and the Linear Separation Assembly’s electric Detonators. These items are the last components to be integrated to the spacecraft, and they are installed on the otherwise completely assembled stack on the launch pad. Furthermore, integration of the three RTGs is a extremely sensitive activity that must operate like clockwork, since each technician performing the installation can only be exposed for a few minutes to close proximity of these radioactive components. No holdups, interferences, or delays could be tolerated.

The Cover Assembly must be deployable (Cover closed to protect the MEA) after separation from the launch vehicle, with a life requirement of 25 cycles to restow (Cover open) for engine firings during the cruise to Saturn. Just prior to Saturn orbit insertion, the MEA Cover will be re-stowed (opened) permanently. Due to the mission criticality of this device blocking the main engines, it was deemed that the Cover Mechanism should be fail-safed to an open position. This was implemented by designing the Cover to be ejectable in the event of a failure of the motor drive, bearings or gears.

The Cover had to provide 100% area protection of the Main Engine Assembly with multiple layer insulation (MLI) spaced a minimum of 20 cm (8 inch) away from the engine nozzles. The Cover must also restow (open) sufficiently to provide an adequate field of view of the engine nozzles to space for thermal cooling. In addition, it was highly desirable to be able to verify the Cover Mechanism Assembly at the spacecraft system level solar thermal vacuum test. Thus, a design goal was to provide an operational capability under Earth gravity without external assistance.

DESIGN IMPLEMENTATION

The overall design concept was to provide a collapsible half sphere that pivots at the equator to accomplish the deployable protection. For simplicity, the Cover Assembly would be driven from one end pivot only, with an idler pivot on the other end that would be driven by the Cover structure. Therefore, the Cover Assembly would consist of two half circle *Bows* pivoting at one end on a *Drive Mechanism* and at the other end on an *Idler Mechanism,* with a MLI envelope that folds between the Bows. By configuring the Cover to mount outside of the Launch Vehicle Adapter (LVA), only JPL designed hardware would have to be modified, no volume restrictions existed except for the RTG and Detonator integration issues, and the resulting deployment would then consist of a simple rotation about a single pivot axis. After separation of the spacecraft from the launch vehicle, the Cover would then be deployed over the MEA. The size of the Cover was determined by the volume needed for the Drive Mechanism to clear the LVA. Selection of JPL's largest predesigned redundant actuator for the Drive Mechanism resulted in a Cover that was over 2.1 meters (7 feet) in diameter.

The addition of the Cover changed the spacecraft thermal design, which must now meet the allowable flight temperature ranges *with or without* the Cover deployed. Deploying (closing) the Cover over the bottom of the spacecraft, in addition to enclosing the main engines, also blocks the Propulsion Module's view to space. Therefore, a Thermal Skirt was required to provide a space-facing radiator for the Propulsion Module as well as to close out the open annulus of the Cover for micrometeoroid protection. The Cover Assembly must also be supported through the launch dynamics environment. This was accomplished by the addition of two Launch Restraint Assemblies (LRAs) to support the Cover Bows and protect the Cover MLI envelope from damage. The LRA spring capsules each provide a 935 Newton (210 lb) preload through the Cover from the LVA, through each Bow, across the separation plane to the spacecraft structure. The required stroke of .5 cm (.2 inch) is to compensate for the potential deflection of the spacecraft during launch vibration and keeps the LRA ball-and-socket interfaces together. When spacecraft/launch vehicle separation occurs, the LRA interfaces are self-releasing as the two bodies part. Small kickoff springs within each joint insure proper release (Figure 5). Each LRA is spring loaded to rotate away from the Cover Bows, enabling an unencumbered deployment.

Cover Design

The Cover structure consists of a leading *Drive Bow* made from 3.2 cm (1 .25 inch) aluminum tubing and a *Fixed* Bow made from 2.5 cm (1 .00 inch) tubing of the same material, each formed into a half circle. These Bows pivot at their ends at a *Hub*. The two Cover Bows together result in the full equator of a sphere. Intermediate *Full Stays,* sliding within a groove in each hollow Hub, then assemble to make up the Cover frame. The Hub design consists of the intermediate Cover stiffeners (the Stays) sliding in a slot of the hollow Hub, being retained by the “lollipop” head of the end fitting. The Stay fittings allow efficient compaction because they can translate axially (due to the flexibility of the Cover envelope) within the Hub interior volume to stack flatly when stowed (Figure 6). Each Stay is made from 3.2 mm (.13 inch) thick graphite-epoxy sheet cut into the shape of an arc. The “lollipop” shaped end fittings, machined out of 15-5 stainless steel, are bonded on the ends of the Stays. Graphite-epoxy was used for the Stays rather than a metallic rod to insure that if hit by a micrometeroid, metal particles could not spray out like shrapnel. The flat shape of the Full Stay end fittings allow the Cover frame to collapse to a relatively compact stowed position. The MLI deployable half sphere micrometeroid shield is attached to the Bows, and eight Full Stays slide into pockets sewn within the envelope. Seven *Partial Stays* (that do not extend all the way to the Hubs) are sewn into pockets within the envelope between each set of Full Stays to further support the shape of the half sphere.

The Cover envelope is fabricated from two layers of .20 mm (.008 inch) thick Beta cloth (Teflon-impregnated glass cloth) sandwiched between two layers of .03 mm (.001 inch) carbon-filled Kapton. The spherical shape is a result of cutting the envelope pattern out of gore-shaped pieces and sewing them together. (Imagine peeling a globe). In this way, the Cover becomes a half sphere accordion, collapsing into a relatively flat package. Applying torque to the Drive Bow allows it to pull the Cover envelope to the spherical shape, or push it back to the stowed position. Because Cassini must have an isopotential exterior surface to meet its science requirements, the Cover must be grounded to the spacecraft structure. The carbon-filled Kapton outer surface is sufficiently conductive, but it must be provided a electrically conductive path to the spacecraft to bleed off any potential that develops. This is accomplished through the Bows and the Full Stays, and the materials and finishes within the Hubs were selected to meet this requirement. The Stay fittings were left as bare polished stainless steel sliding within titanium Hubs coated with Nedox SF-2 (product of General Magnaplate, Ventura CA). This Nedox is a hard nickel coating filled with Teflon, and was selected because it is one of the few dry lube coatings that is electrically conductive.

Cover Development

Although JPL has developed a number of deployable sunshades and similar structures, consisting mainly of stiffened multilayer insulation blankets, this was the first time that a complete half sphere had to be deployed. It was felt that key to the success of the development was insuring proper articulation of the frame, particularly in the area of the Hub. Design of the Cover envelope was originally assumed to be the major challenge, but the person responsible for design and fabrication of Cassini's thermal blankets was unperturbed. Essentially "eyeballing" the shape of the individual gore segments, he fabricated a half scale mockup within days of receiving an articulatable frame to build upon. This half scale mockup was built as a proof-of-concept for the Hub design and the Cover envelope fabrication techniques. The compacting material between the Stays is prevented from interfering with the stowage by removing a 20 cm (8 inch) radius of Cover envelope about each pivot point. Small semicircular fixed shields close out these openings in the Cover.

After the original half scale mockup proved the task attainable, a full size prototype was fabricated with the expectation that new surprises would reveal themselves. Eventually, two engineering model units had to be made in addition to the prototype. The progression from using the MLI materials available from stock to current flight-approved materials caused a number of problems. The first and primary problem became evident with the initial engineering model. The flight materials were significantly stiffer than the materials used for the full size prototype. The inner and outer layer of Kapton was stiff and brittle, buckling into large lumpy wrinkles when stowing. As the Cover folds (like an accordion) while stowing, the radius of the inner edge of each fold gets smaller, but the material is cut to the larger radius of the deployed sphere. Therefore, the material must buckle along the direction of the folded edge. The material of the earlier prototype was more forgiving, resulting in distributed small wrinkles and a maximum stowed height of about 15 cm (6 inch). The new Beta cloth was similarly much stiffer, such that the stowed height due to the buckling increased to between 28 to 38 cm (1 1 to 15 inches). It was attempted to produce shaped cuts in the internal Beta cloth, and different methods of suspending the Stays to the envelope were tried. One of the causes of the problem was that the original Kapton materials were manufactured with a rip-stop scrim that didn't stiffen the material excessively, whereas the new materials were manufactured with a different scrim. While old material stocks were pursued for the task, eventually all new materials were used. The completed flight unit substituted carbon-filled Kapton for the original aluminized Kapton. Although the carbon-filled Kapton was twice the thickness of the aluminized material, it was more flexible due to the use of different scrim and didn't exhibit some cracking seen in test. While the final design never did perform as well as the prototype, its stowage improved to a maximum height of about 25 cm (1 O inches). The Launch Restraint Assemblies (LRAs) had to be re-manufactured to accommodate the larger stow envelope.

Drive Mechanism Design

Ejection of the Cover was made possible by coupling the drive elements to the Hub with a single spur gear set, configured in such a way that the Cover could "drop away" from the mesh (Figures 3 & 7). The drive gear is simply supported between the bearings internal to the motor actuator and a needle bearing in the Drive Mechanism housing, The driven gear is cantilevered from the Hub to complete the gear mesh. To minimize the size of the assembly, the gears are designed with high strength (and high toughness) Maraging steel C300. The gear blanks were machined, heat treated to 200 MPa (290 KSI) and then fabricated to final form by wire electrostatic discharge machining (Reference 2). The drive actuator used is a size 20 JPL Dual Drive Actuator (DDA). This is a fully block redundant actuator, combining two paths of 20:1 dual stage spur gears into coaxially mounted 605:1 pancake Harmonic output gears with a common drive output (Figure 9). Two independent, electronically commutated DC brushless motors provide the drive torque (Reference 3). This Dual Drive Actuator is the one of the same units that were built for the Shuttle Imaging Radar foldable antenna. The SIR-C antenna design was changed mid-project and these actuators were not needed. Ten years later, the existence of fully machined and kitted components for Cassini's usage really helped in getting the MEA Cover development performed in time. The bearings and gears are lubricated with Bray 600 or 601, appropriate to the corrosion resistance of the materials. JPL has always used the "off-the-shelf" nodular cast iron Harmonic gears with 52100 bearings for its DDAs rather than the custom stainless steel units used elsewhere in the industry. No ill effects have resulted in this practice, and delivery schedule problems are alleviated. The drive/driven gears were only grease-plated with Bray 600.

The rotation of the Cover is limited by adjustable stops within the Drive Mechanism. To reduce the loads on the actuator bearings, a balanced stop design is used. A lever is allowed to pivot about the actuator bearing centerline and contacts equidistant stops for the stow and deploy positions. This lever also contacts the microswitch actuating levers to provide stow and deploy telemetry indications (Figure 10). One subtle detail that was required to be incorporated in the design was to accommodate the deflection of the Drive Bow during the launch dynamics. The actuator is non-backdrivable, so to prevent the output gears from being overtorqued due to the applied loads from the Drive Bow, sufficient backlash had to be incorporated in the load path from the Bow to the actuator. The drive/driven gear mesh provided approximately half of the total required 3.3 degrees of backlash, while the spline connection from the driven gear to the Hub was fabricated to provide the other half. Only a small amount of backlash was available within the Harmonic gear mesh. The required amount of backlash was maintained at the lowest practical level by the use of a larger diameter tubing for the Drive Bow. For launch, the Cover's position must be rotated to the center of the backlash region, 1.7 degrees towards the deploy direction from the stow hardstop.

Idler Mechanism Design

The Idler Mechanism consists of a simple pivot using the same high load bearing configuration of the Drive Mechanism primarily to simplify the design and procurement needs (Figure 4). Similarly, the same mounting interface as the Drive Mechanism was used. Lightweight Delrin gears transfer the rotary motion of the Cover Bow to a potentiometer. The pot was a flight spare 5000 ohm Cermet unit (manufactured by Beckman Industrial) left over from the Mars Pathfinder project. To replicate the resistance range of a temperature transducer circuit, a series/parallel set of resistors was packaged in a box next to the pot.

Ejection Mechanism Design

Each Cover Hub is fastened to the mechanism's structure with a 6.3 mm (.25 inch) bolt passing through the throat of a Bolt Cutter (manufactured by Special Devices, Incorporated, Newhall Ca). The Hub interface retains the shear loads by the use of 30 degree male/female wedges, and is hardcoated with Nedox dry lubrication to insure release while maintaining electrical conductivity for Cover grounding purposes. A pair of linear spring capsules in both the Drive and Idler Mechanism provide kickoff energy to the Hubs (Figures 7 & 8). An "ejected state" microswitch is also installed at both Mechanisms to provide telemetry. Extensive modeling using ADAMS kinematics analysis was performed to insure that the ejected Cover could not recontact the spacecraft. The Cover was modeled as 9 rigid bodies connected by revolute joints with stiffness and friction forces incorporated. The initial center of gravity of the deployed Cover resides directly in line with the Hubs, while a large offset exists in the stowed condition. While there would be no reason to eject a Cover failed in the stowed position, the partially stowed position equivalent to a 30° wedge was determined to be the maximum limit of a "safely stowed" position. Any further deployment (closing) of the Cover would adversely affect the thermal environment of the engines when ignited. By analysis it was determined that additional linear spring capsules located at the Launch Restraint Assemblies, each providing 40 Newtons (9 lb) of force, could push off the Fixed Bow and solve the problem. The ADAMS analyses confirmed that the Cover would tumble off the spacecraft as before, but the additional LRA pushoff springs decreased the tumble rate while increasing the translation rate, thereby allowing the Cover to clear the main engine nozzles with adequate safe clearance.

TESTING AND INTEGRATION

The verification of the design for acceptable flight performance began with the prototype Cover Assembly, in which flex cycle testing was performed in the ambient environment to see if any unexpected wear occurred in the envelope materials and the Stays sliding within the Hubs (Figure 12). The engineering model (EM) Cover and Mechanisms were then fabricated, and a test fixture was produced that allowed the Assembly to be tested with the pivot axis horizontal or vertical. In this way the operating performance could be characterized for both the spacecraft system test orientation as well as in a partial gravity-compensated configuration. The integrated Assembly was placed with the pivot axis in a vertical orientation in JPL's 10 Foot Space Simulator (over 3 meter diameter test chamber) for thermal vacuum qualification. The Mechanisms were controlled with separate heat exchangers to maintain their test temperatures between -50°C to +65°C, while the chamber shroud controlled the temperature of the Cover to -1 15°C. The Assembly was calibrated for torque versus deployment angle, and a 100 cycle life test was performed (4 times the life requirement). Full stowage to the launch position was not achievable although the Cover was positioned within the "safely stowed" angle. At this juncture it was decided that the drive actuator should be redesigned to provide more output torque.

The EM actuator for the MEA Cover Mechanism was originally assembled with a 363:1 Harmonic gear ratio, which was predicted to provide an output torque of 76 N•m (670 in•lb). The actuator was fully temperature/atmosphere dyno tested, which consists of generating clockwise and counterclockwise speed/torque/current plots for each motor and both motors operating conditions. The EM unit's output torque was not only lower than the prediction, but the performance was severely degraded under cold operating conditions. While predicting the performance of Harmonic gear systems can be an inexact science, this shortfall in output torque was unprecedented at JPL (See "Problems and Lessons Learned" below). A flight unit with 605:1 Harmonic gears was assembled, flight acceptance temperature/atmosphere dyno tested and retrofitted to the Drive Mechanism. The flight Mechanisms and EM Cover were integrated on the lower portion of the spacecraft structure with all the new modifications and was subjected to flight acceptance dynamics testing (Figure 11). Afterward, the hardware was delivered to Assembly, Test and Launch Operations (ATLO) for flight spacecraft integration in preparation for the system testing. New materials for the flight Cover had still not been delivered.

The flight Cover was eventually completed I-he final mass of the deployable Cover as delivered was 18.3 kg, which included the LRA spring capsules. Assembly level thermal vacuum characterization testing was performed in the 10 Foot Space Simulator, including a 25 cycle life test. The test actuator, retrofitted with the 605:1 ratio, was used to operate the flight Cover and for the torque versus angle characterization (Figure 16). The flight Cover was inspected, and delivered to ATLO in time for system dynamics and solar thermal vacuum testing. In all, the mass increase to Cassini for the retrofit of the MEA Cover Mechanism Assembly, support structure and associated thermal structure totaled over 41 kg.

PROBLEMS AND L. ESSON LEARNED

Two significant problems occurred during the development of the Cover Mechanisms, both involving the Dual Drive Actuators. The first episode appeared when the DDA's Harmonic gears were received and the actuator was dyno tested. The output torque was far below predictions. Although the torque requirement was expected to be less than 12 N•m (100 in•lb), the "desirement" for the capability to operate in the system level testing drove the design for the production of the most torque possible. The 363:1 output ratio was predicted to produce approximately 76 N•m (670 in•lb) at room temperature, Its actual output peaked at 65 N•m (575 in•lb) but dropped off substantially at cold temperatures. The unit would stall as low as 28 N•m (250 in•lb) at -50°C. These torques were insufficient to fully stow the engineering model Cover during the thermal vacuum qualification tests. At cold temperatures the torque requirement to stow the Cover, made from the stiffer flight materials, to an acceptable position turned out to be almost 34 N*m (300 in•lb) (Figure 15). While grasping for an explanation it was concluded that the higher ratio 605:1 gears from the SIR-C program must be installed to obtain more torque.

A torque limiting clutch was originally mounted to the actuator in the SIR-C program to insure that the output torque would not exceed the maximum torque rating of the gears, and to protect the actuator during ground testing of that very large deployable. The original qualification unit from SIR-C was tested to motor stall with the torque limiting clutch removed to ascertain whether the Harmonic gears would ratchet. The maximum torque developed was 79 N•m (700 in•lb), safely below the ratchet limit of 96

N•m (850 in•lb). Another actuator was assembled for Cassini using the 605:1 gears, and flight acceptance dyno testing commenced. Again, the output torque was lower than expected! Something unexplained was occurring.

While not ever fully explained, one primary cause of the anomaly was attributed to the Bray grease. The formulation of Bray grease was changed since the SIR-C actuators were originally tested in 1987. Although it is still identical chemically to the original, the grease is now formulated without the use of ozone depleting chemicals. In the past the base oil and Teflon particles were mixed in Freon, and the volatiles were evaporated off, until the usual buttery consistency resulted. Now the base oil and Teflon are mixed without any liquifier or solvent. A number of times in the last couple of years assembly technicians have noticed that some tubes of the Bray grease have exhibited lumps. The original SIR-C qualification unit was then fully dyno tested, cleaned and relubricated with the new grease, and re-dyno tested. Figure 13 shows a significant degradation of performance of the size 20 DDA with 605:1 Harmonic gears at -50°C, apparently due only to the new grease formulation. Finally, with side-by-side testing of an actuator showing differences in performance, the grease came under more focused scrutiny.

Samples of the old Bray 601 EP were compared to the new Bray 601 EF under a scanning electron microscope. While the old grease displayed consistent-sized spherical particles ranging from 3 to 5 microns, the new grease showed inconsistent, jagged edged particles ranging as large as 30 microns diameter. Although testing at other NASA facilities have indicated no performance differences between the old and newly formulated Bray grease, including cold temperature operation, JPL's experience with this one A-to-B comparison showed otherwise. However, while no other explanation was found, testing was limited and schedule restrictions dictated that the investigation be halted. The manufacturer of Bray grease (Castrol of North America) was contacted and the problem explained. It is believed that Castrol has initiated better mixing procedures for a more consistent product. Interestingly, the biggest mystery involved the cold temperature performance of the first actuator built for Cassini, using the 363:1 Harmonic gears. While the actuator with the 605:1 gears demonstrated a speed reduction from room temperature to -50°C ranging from 19% (old grease) to 30% (new grease), the 363:1 gears exhibited a whopping 71% reduction in no-load speed. Figure 14 shows how this speed difference at cold temperatures was remarkably high compared to the 605:1 unit. Also compared are the speed differences due only to the change to the new grease formulation. No further explanations were pursued due to the workload in completing the MEA Cover Mechanism.

A second problem that emerged after delivery of the MEA Cover Mechanism to Cassini involved the electronic packaging of the DC brushless motors. A motor in another assembly failed to operate properly after dynamics testing. Disassembly revealed that one of the drive transistors had broken free from the circuit board and the lead wires disconnected. Further inspection revealed that 4 of the 6 transistors were debonded. The vibration levels were very nearly the same as the levels these flight spare motors were originally qualified to in 1982. The conclusion of the investigation was that the conformal encapsulant used to bond the T05 cans to the circuit board had aged, and did not have adequate bonding strength. After closely scrutinizing the electronics packaging, it was observed that workmanship standards have improved vastly in the last 14 years. It was decided to remanufacture the entire electronics packaging for all of Cassini's DDA motors. The MEA Cover DDA was removed from the spacecraft after system test and the motors replaced. Each motor had been reworked and re-dyno tested over its qualification temperature range. After retrofit to the DDA, actuator dyno and vibration testing was repeated prior to reassembly into the Drive Mechanism and redelivery to Cassini.

Another late change related to the Cover ejection occurred. The original ejection kinematics analysis did not provide any energy to the separation from the Bolt Cutters. During the lot acceptance testing of the Bolt Cutter, fixturing was included that allowed a 7 kg mass to be swung as a pendulum and the velocity was recorded. Upon cutting of the bolt, this mass was propelled over 5 cm (2 inch) in 110 milliseconds! Incorporation of this imparted energy in the ADAMS analysis determined that the Cover would contact the engine nozzles under some ejection conditions. Fortunately, merely deleting the four Hub kickoff spring capsules accomplished safe ejection under all conditions.

In terms of the design and development of collapsible MLI envelopes, it appears that there is no magic involved. However, one should plan on the fabrication of a number of prototypes because so many parameters are derived empirically. Compensation for seam thickness in the blanket pattern varies not only with the thickness of the material but also in the stiffness of the material, which affects the way it is sewed. One can also vary the number of rigid supports to find the optimum number and spacing. One feature that was incorporated between the fabrication of the last engineering model and the flight unit was that the radius of the Stays were made 13 mm (.5 inch) or so smaller than the radius dictated by the Bows. The Cover folded **more effectively** because the Stays would stack within the radius of the Bows, resulting in a smaller overall stow thickness.

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ACKNOWLEDGMENTS •

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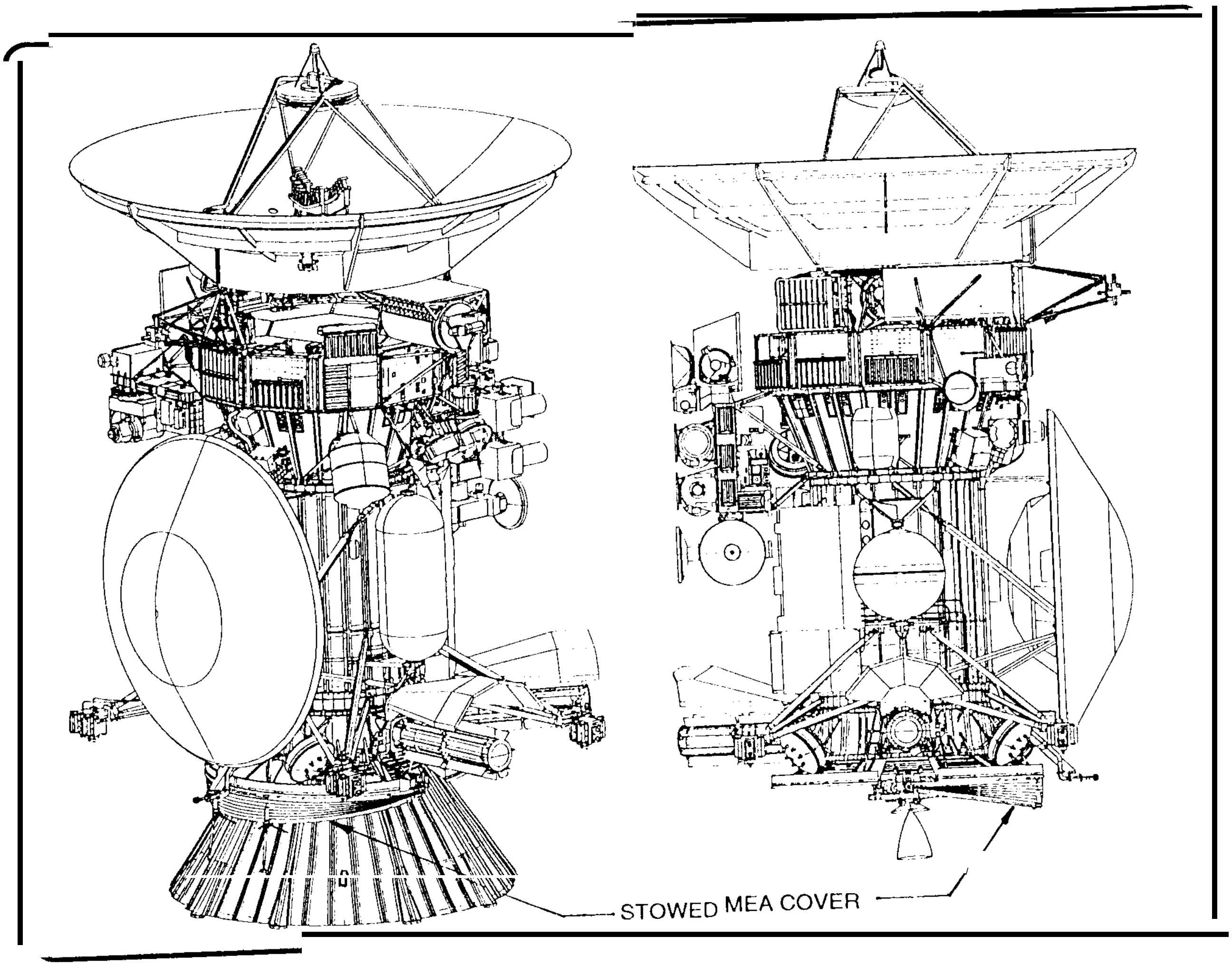

FIGURE 1 -CASSINI, BEFORE AND AFTER SEPARATION FROM LAUNCH VEHICLE ADAPTER

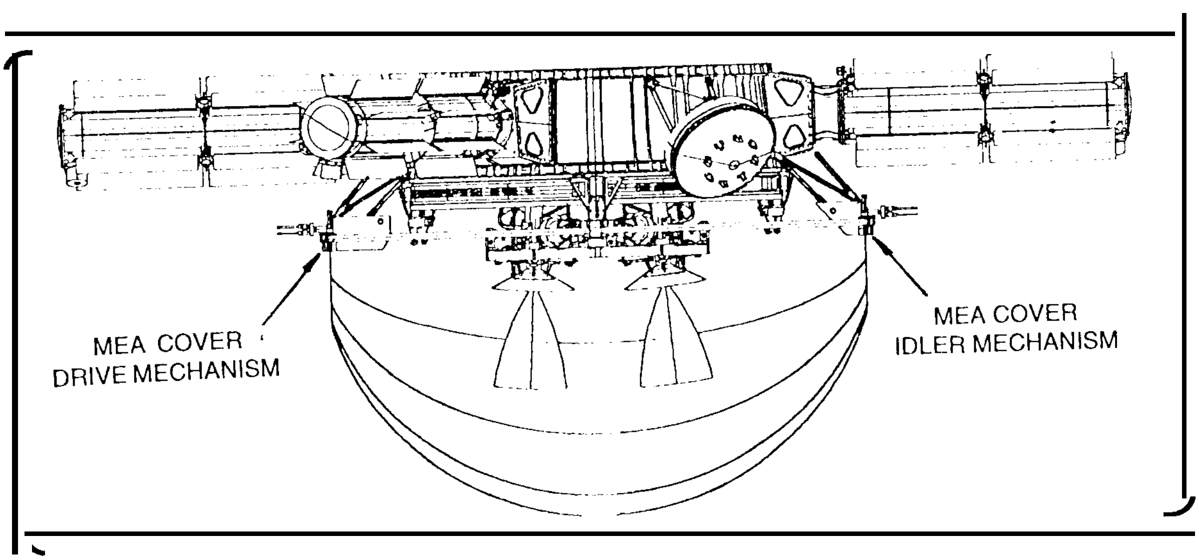

FIGURE 2- VIEW OF CASSINI'S REDUNDANT MAIN ENGINES

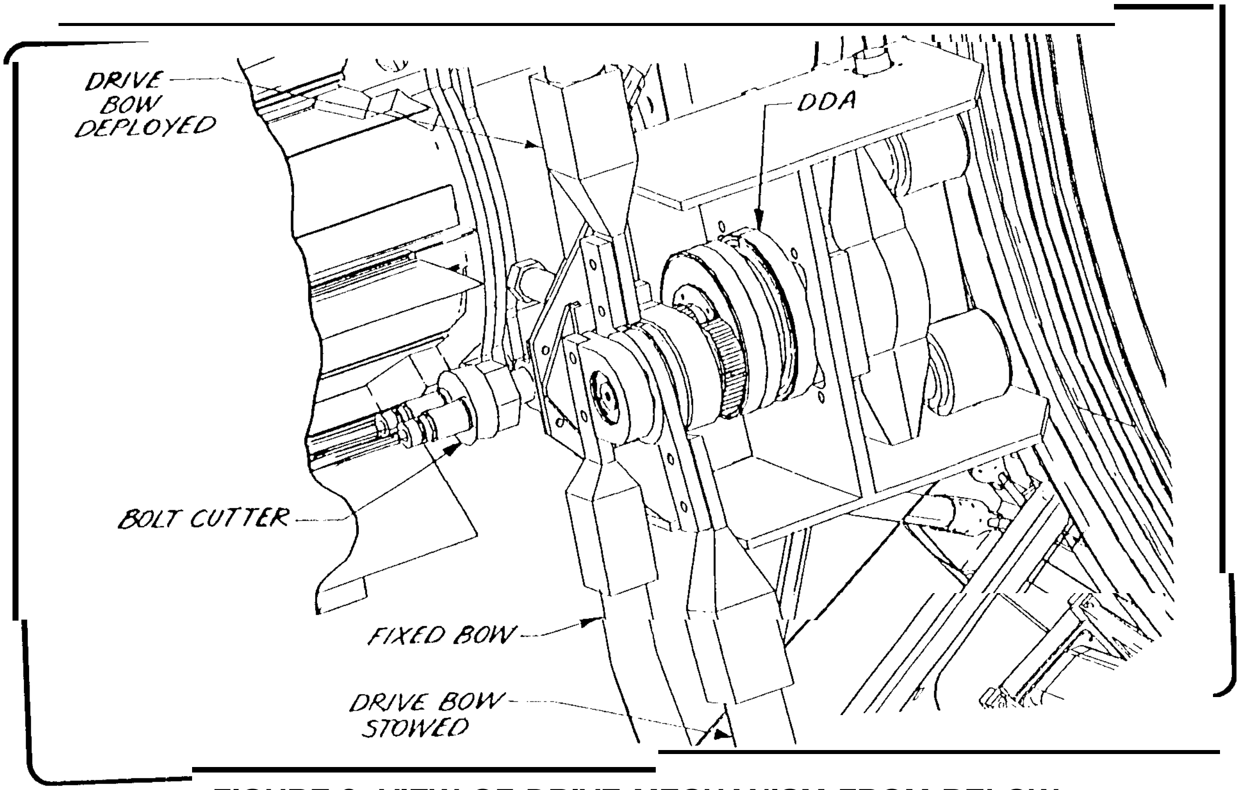

FIGURE 3- VIEW OF DRIVE MECHANISM FROM BELOW
(SOME COVERS REMOVED)

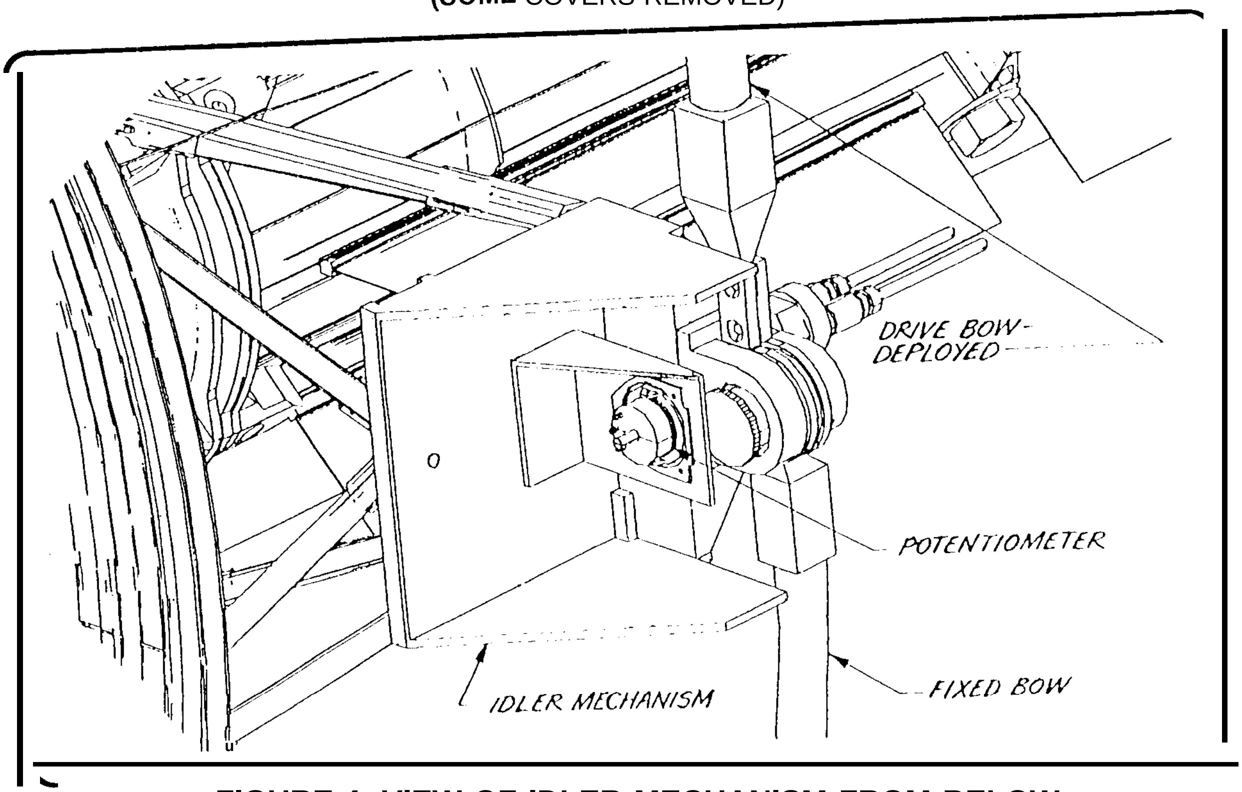

FIGURE 4- VIEW OF IDLER MECHANISM FROM BELOW
(SOME COVERS REMOVED)

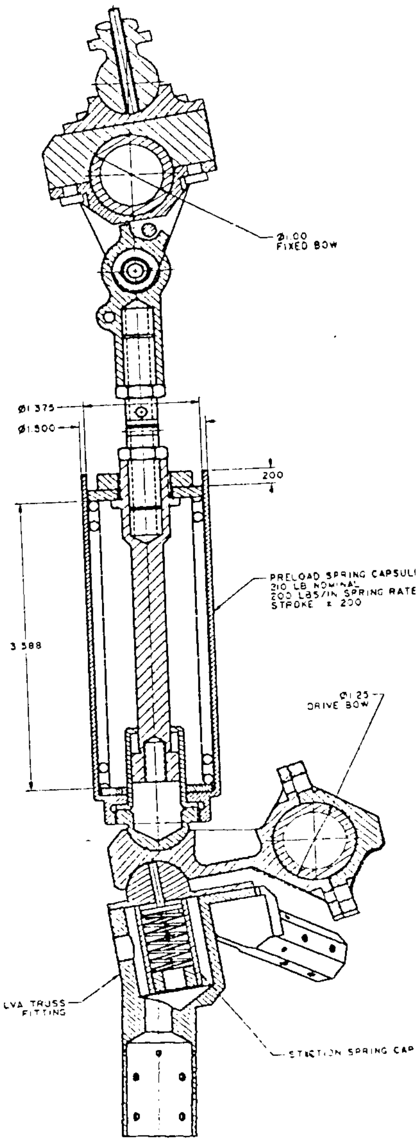

FIGURE 5- DETAIL OF LAUNCH RESTRAINT ASSY

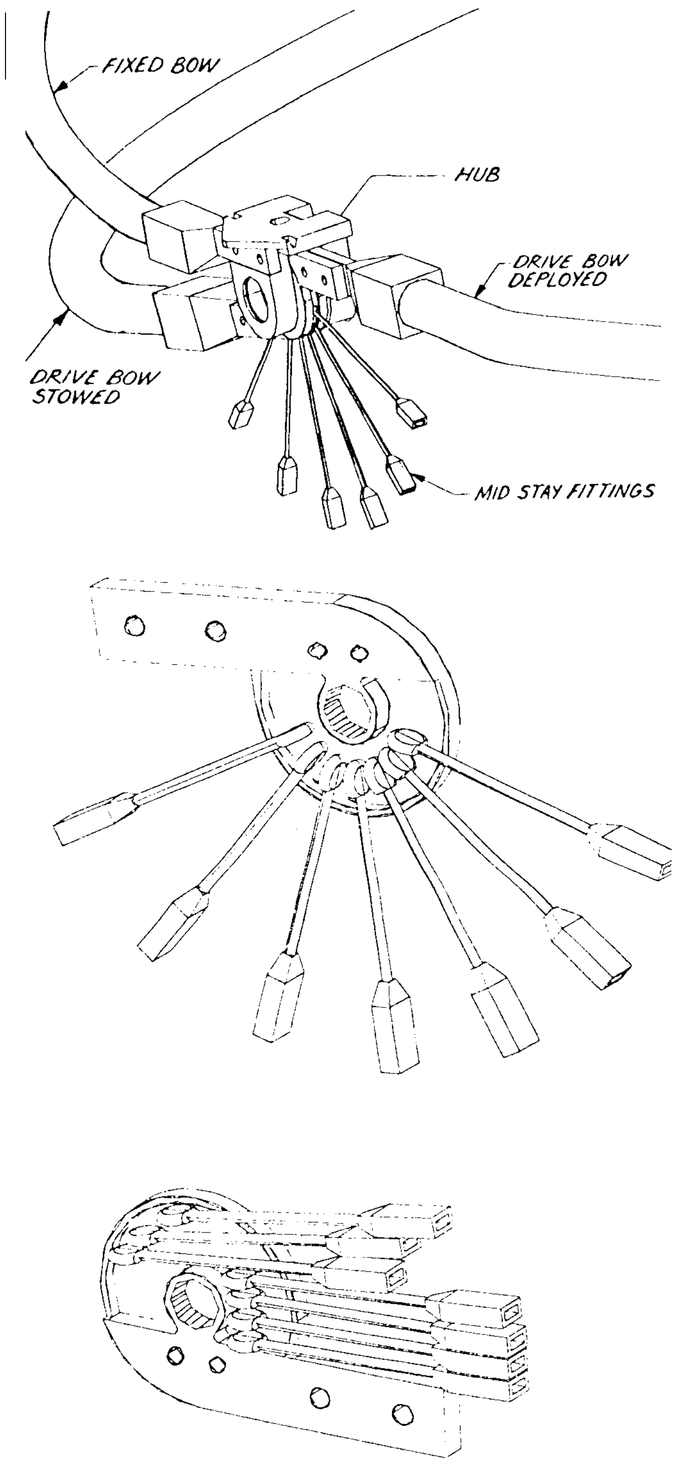

FIGURE 6- DETAILS OF COVER HUB

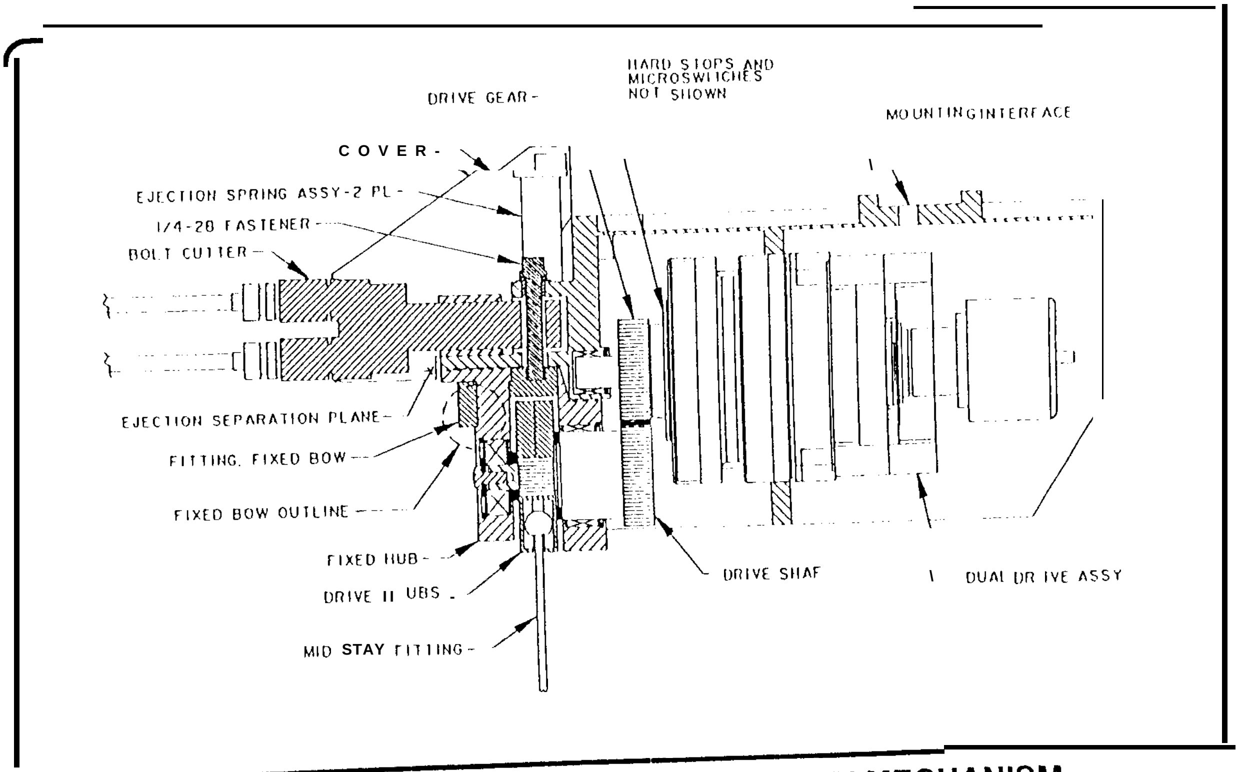

FIGURE 7 - CROSSECTION VIEW OF DRIVE MECHANISM

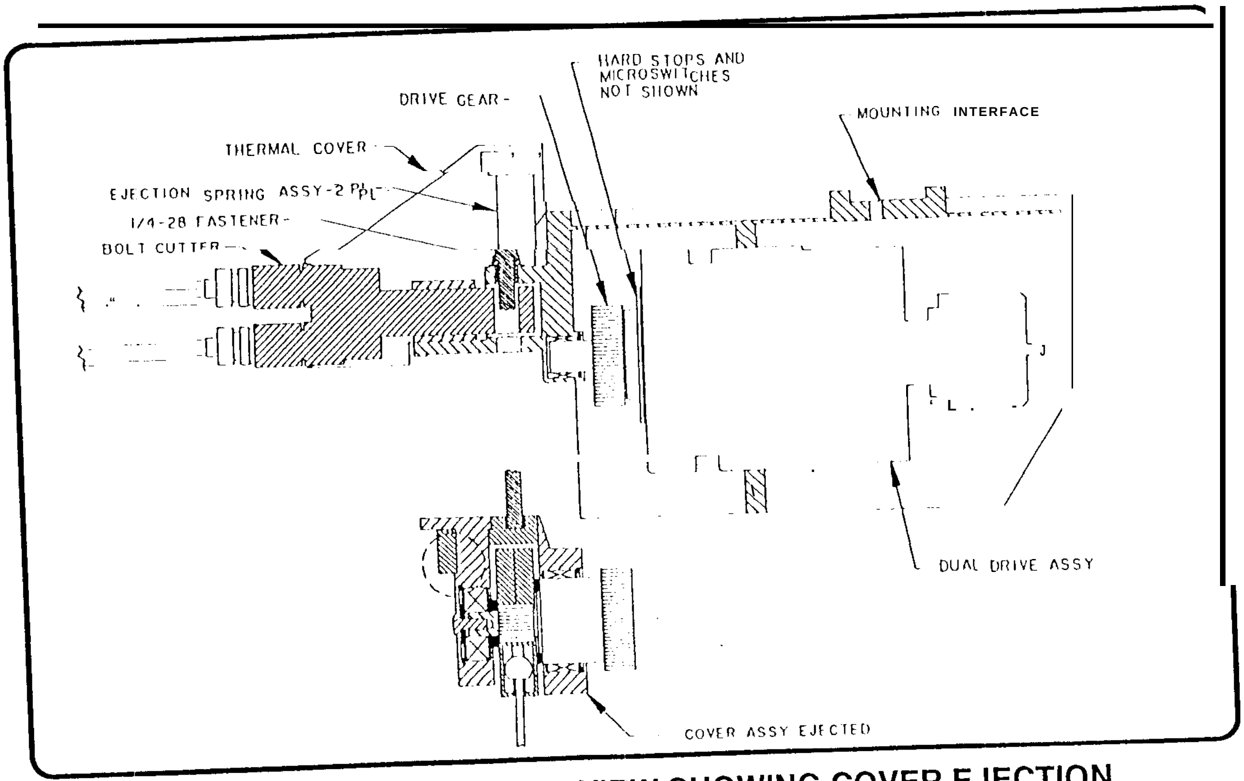

FIGURE 8 - CROSSECTION VIEW SHOWING COVER EJECTION

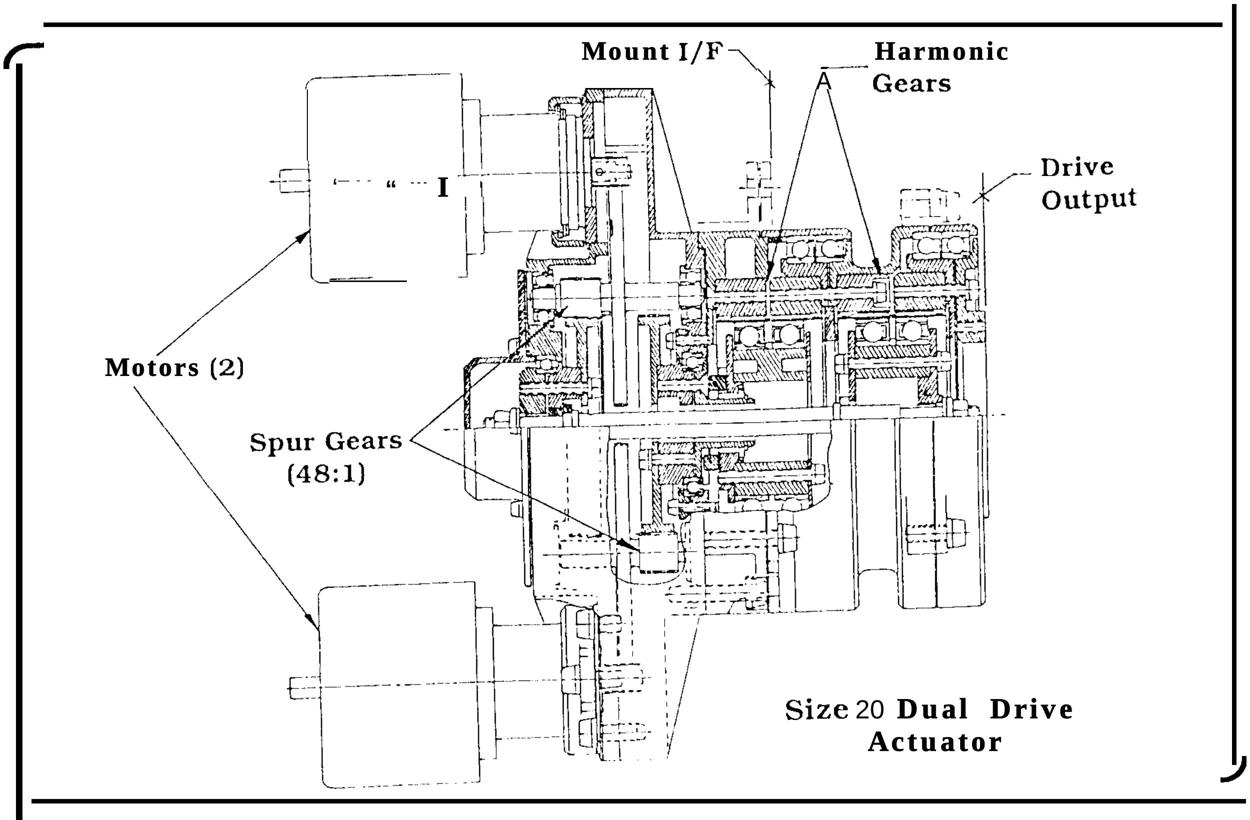

FIGURE 9- CROSSECTION VIEW OF MEA COVER DDA

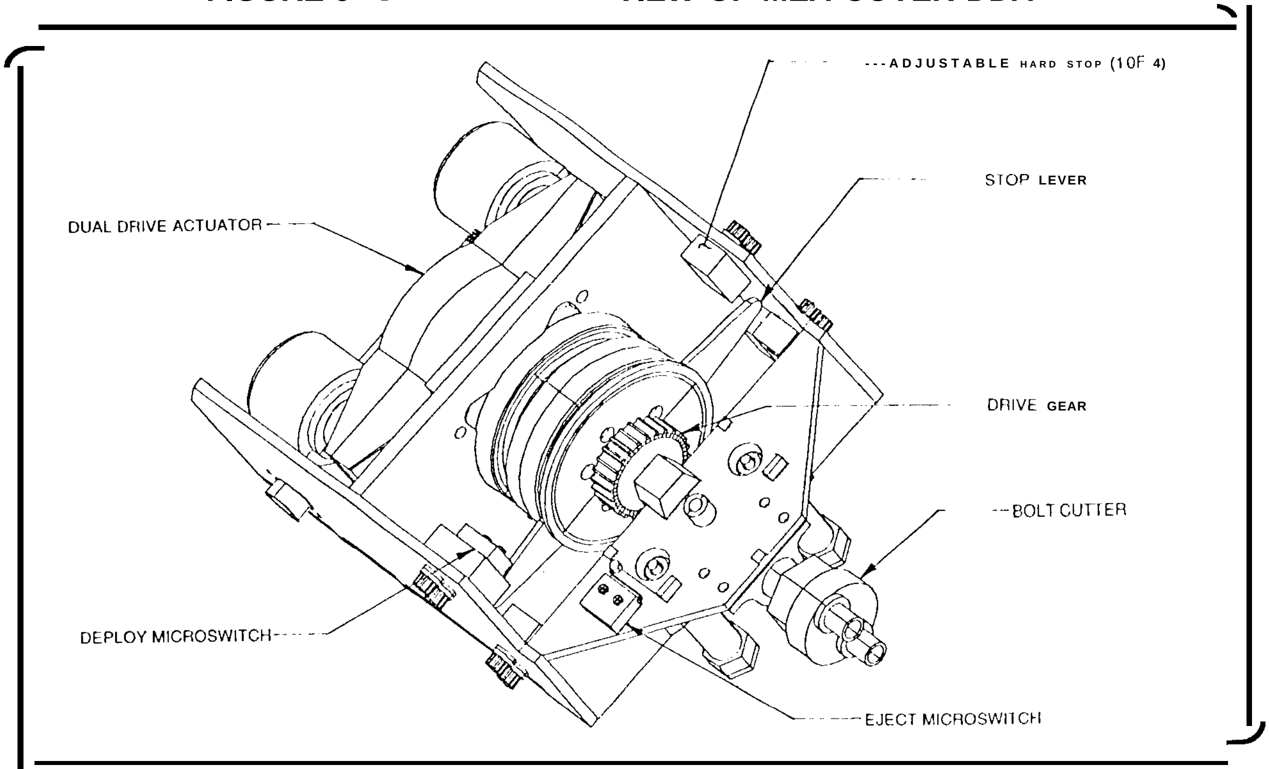

FIGURE 10- VIEW OF DRIVE MECHANISM SHOWING STOP LEVER

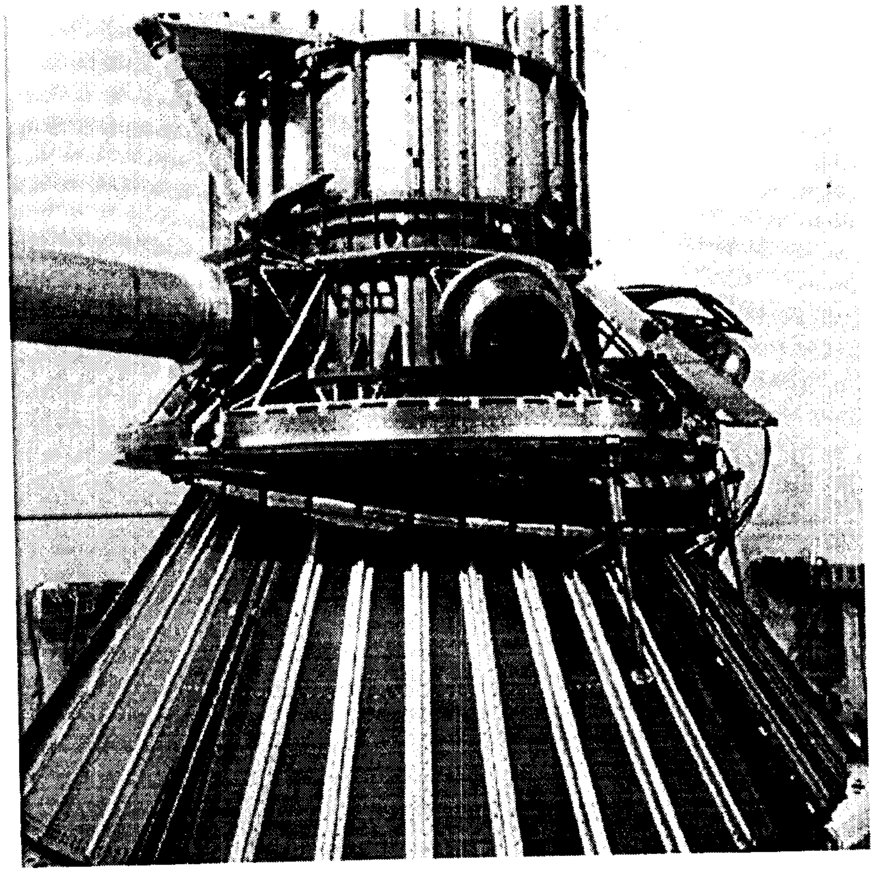

FIGURE 11 - EM COVER ASSEMBLY, DYNAMICS TEST CONFIGURATION

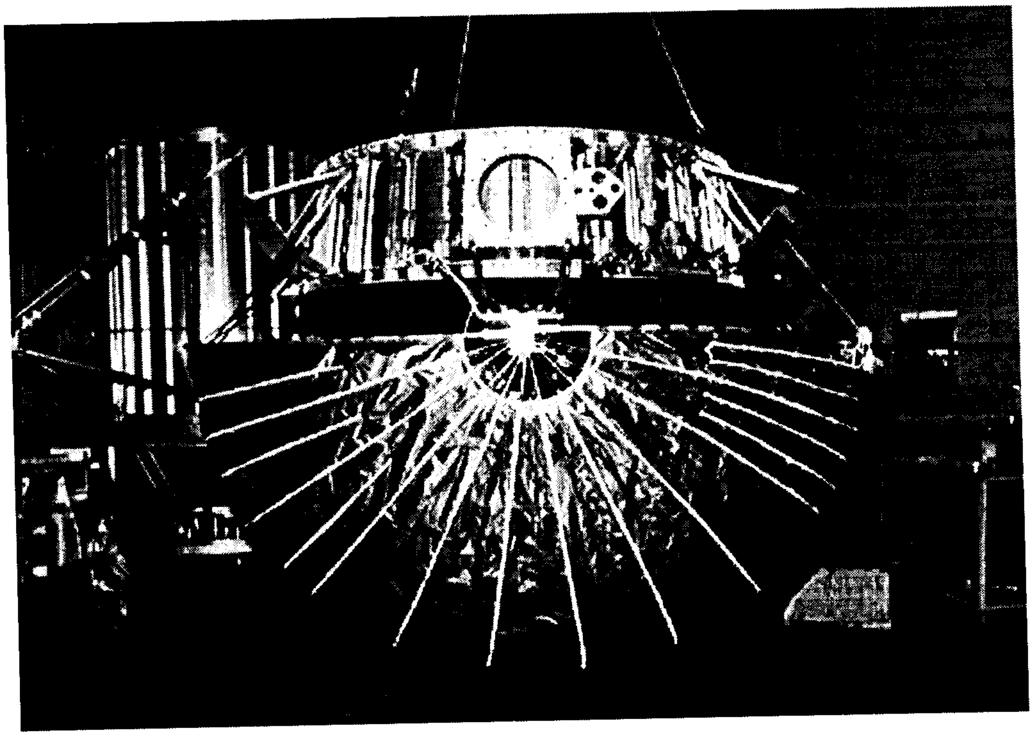

FIGURE 12 - PROTOTYPE COVER ASSEMBLY TESTING

FIGURE 13 - BRAY GREASE COMPARISON TEST (605:1, 23° C to -50° C)

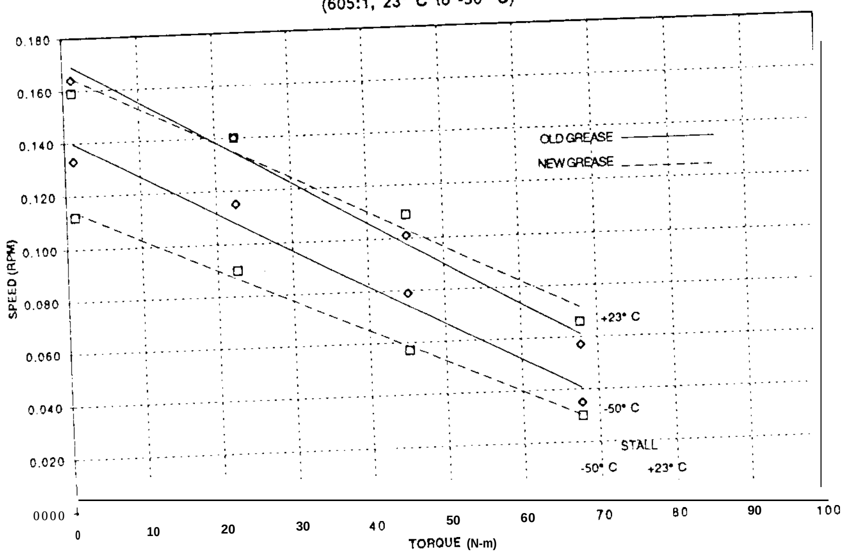

FIGURE 14- SPEED REDUCTION DUE TO TEMPERATURE DROP (23° C to -50° C)

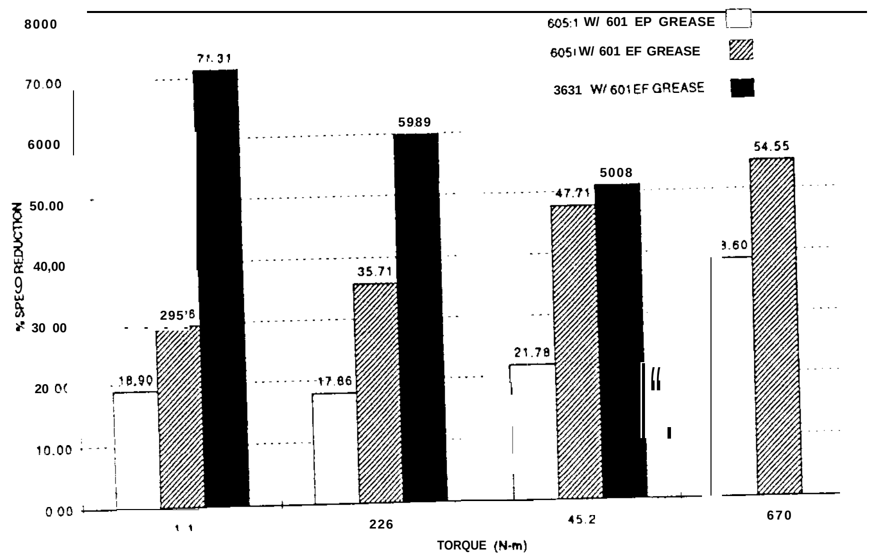

FIGURE 15- ENGINEERING MODEL MEA COVER TORQUE TO STOW VS. ANGLE

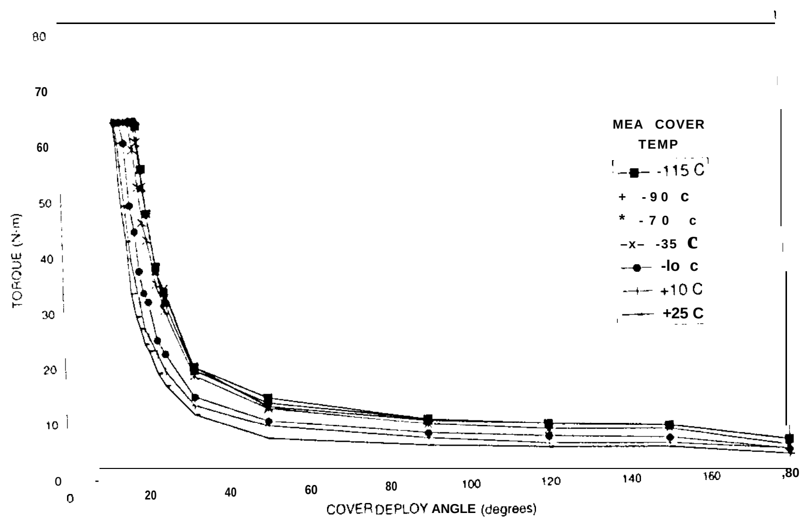

FIGURE 16- FLIGHT MODEL MEA COVER TORQUE TO STOW VS. ANGLE