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HIGH-PERFORMANCE, FLEXIBLE, DEPLOYABLE ARRAY DEVELOPMENT FOR SPACE APPLICATIONS[1]

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SUMMARY

Flexible, deployable arrays are an attractive alternative to conventional solar arrays for near-term and future space power applications, particularly due to their potential for high specific power and low stowage volume. Combined with low-cost flexible thin-film photovoltaics, these arrays have the potential to become an enabling or an enhancing technology for many missions. In order to expedite the acceptance of thin-film photovoltaics for space applications, however, parallel development of flexible photovoltaics and the corresponding deployable structure is essential. Many innovative technologies must be incorporated in these arrays to ensure a significant performance increase over conventional technologies. For example, innovative mechanisms which employ shape memory alloys for stowage latches, deployment mechanisms, and array positioning gimbals can be incorporated into flexible array design with significant improvement in the areas of cost, weight, and reliability.

This paper discusses recent activities at Martin Marietta regarding the development of flexible, deployable solar array technology. Particular emphasis is placed on the novel use of shape memory alloys for lightweight deployment elements to improve the overall specific power of the array. Array performance projections with flexible thin-film copper-indium-diselenide (CIS) are presented, and Government-sponsored solar array programs recently initiated at Martin Marietta through NASA and Air Force Phillips Laboratory are discussed.

INTRODUCTION

It is evident that a strong trend towards smaller, lighter spacecraft launched on smaller launch vehicles has developed, and presently all spacecraft subsystem components are being scrutinized for improved performance. Along these lines, the power subsystem, and more specifically, the solar array and associated structure, has been receiving significant attention, particularly in terms of higher specific power and reduced stowed volume. The reason the solar array plays such a significant role in this activity is that solar array structures are often physically the largest subsystem, and any reduction of weight on an extended boom can have significant impact on spacecraft operations. To achieve overall specific power levels greater than 100 W/kg, and stowage volumes less than 0.10 m^3 for arrays as large as 1 kW, a complete system approach must be taken. In most cases, aggressive goals such as these cannot be met with evolutionary technologies. Rather, revolutionary advancement of technologies must be accomplished to meet or exceed these goals.

Both reduced array weight and higher photovoltaic conversion efficiency have an effect on spacecraft design in terms of increased payload size/weight. Dynamics of arrays in the kilowatt range can adversely affect spacecraft performance. Presently, PV arrays are nominally 20 W/kg to 40 W/kg, where most of the array weight is associated with the rigid substrate upon which the cells are mounted. As a result, a great deal of effort has been placed on improving efficiency to reduce array size, but low manufacturing yield and increased complexity drives device and installation cost significantly. Other efforts to improve specific power are related to lightweight structures, including fold-up and roll-up technologies. An additional advantage of these lightweight substrate approaches is the promise of smaller stowage volume.

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The following paragraphs introduce the incorporation of revolutionary technologies currently under development at Martin Marietta in both thin-film flexible photovoltaics and in lightweight, reliable deployment/structural elements.

Stowage and Deployment of Flexible Arrays - While use of composite materials in array structures to improve specific power for reduced weight is becoming more commonplace, they cannot easily accommodate small stowage volume, particularly for large arrays. One of the ideal configurations is a flexible array, although implementation of such technology has been limited due to the lack of a commercially-available flexible solar cell design. Some flexible deployable arrays have been flown, most notably on the Hubble Space Telescope and the Solar Array Flight Experiment (SAFE) on the Space Shuttle. In both cases, the fragile nature of the solar cells (Si and GaAs) demanded significant attention to structural support. As a result, no array which has flown was truly flexible; at best, the array consisted of numerous array segments housing the rigid cells hinged together in flexible joints.

Due to the advent of flexible solar cells and modules, many flexible array concepts are now possible. For example, issues regarding the stowage and deployment of a rollup array similar to that used in the Hubble Telescope can be more easily addressed. Difficulty in rolling up rigid, fragile solar cells in a compact stowage volume is eliminated by completely flexible solar arrays and photovoltaics. Because the blanket is completely flexible, it is possible to integrate the deployment mechanism and the blanket for improved rigidity, thereby eliminating the possibility of buckling noted in the latest Hubble repair mission.

Martin Marietta is investigating a variety of techniques to deploy flexible PV blankets. One promising concept utilizes shape memory alloy (SMA) elements to achieve both weight and performance advantages over other technologies, including self-rigidizing inflatable technologies. Furthermore, because the shape memory effect can be activated both electrically and by passive solar heating, structural deformations caused by unanticipated spacecraft maneuvers exceeding design parameters can be autonomously repaired in orbit. This key advantage over every deployment scheme can ensure array repair and continued high-level performance over a significant array lifetime.

In addition to the blanket and deployment schemes, a key aspect in the overall specific power goals is the pointing subsystem. Martin Marietta IR&D funding has developed a shape memory gimbal mechanism capable of two-axis tracking with a weight savings of approximately 50% compared to conventional technology.

Photovoltaics - An attractive alternative to conventional crystalline and amorphous silicon PV is polycrystalline thin-film devices. Polycrystalline thin-film PV, such as copper-indium-diselenide (CIS), offer an alternative to Si for most space power applications. These devices have the highest tolerance for radiation damage of any crystalline PV material in proton and electron environments because their short diffusion length (compared to conventional Si and GaAs) makes them less susceptible to damage caused by radiation. Furthermore, polycrystalline thin-film PV have proven themselves far more stable than single-junction a-Si [1,2]. Polycrystalline thin-film devices, which nominally do not exceed 5-8 μm thickness (excluding substrate), have been reported with efficiencies as high as 15.5% (NREL - 1993) in Air Mass (AM) 1.5 insolation. Also in a joint Martin Marietta/NREL effort to supply a PV test article, an active area efficiency exceeding 13.5% in AM0 was observed. The CIS Technology is discussed in an accompanying paper: "Advances in Polycrystalline Thin-Film Photovoltaics for Space Applications".

LIGHTWEIGHT ARRAY TECHNOLOGIES

Integral Deployment / Structural Elements - The achievement of high overall specific power places a premium on subsystem mass allocations. Current deployable solar arrays generally use motor driven mechanisms to extend rigid panels or deployable structural support elements (Astromasts, STEMs, telescoping booms, etc.). While structurally efficient, the mass of the deployment drives and mechanisms penalizes the overall solar array specific power. For non-retractable applications, inflatable deployment systems are one approach to eliminating the motors and mechanisms, but must include a pressurization system and an envelope or collapsed structure that holds pressure during deployment. Inflatables generate reliability concerns from the standpoint of assuring that the integrity of the pressure envelope is not lost during final stowage prior to the mission.

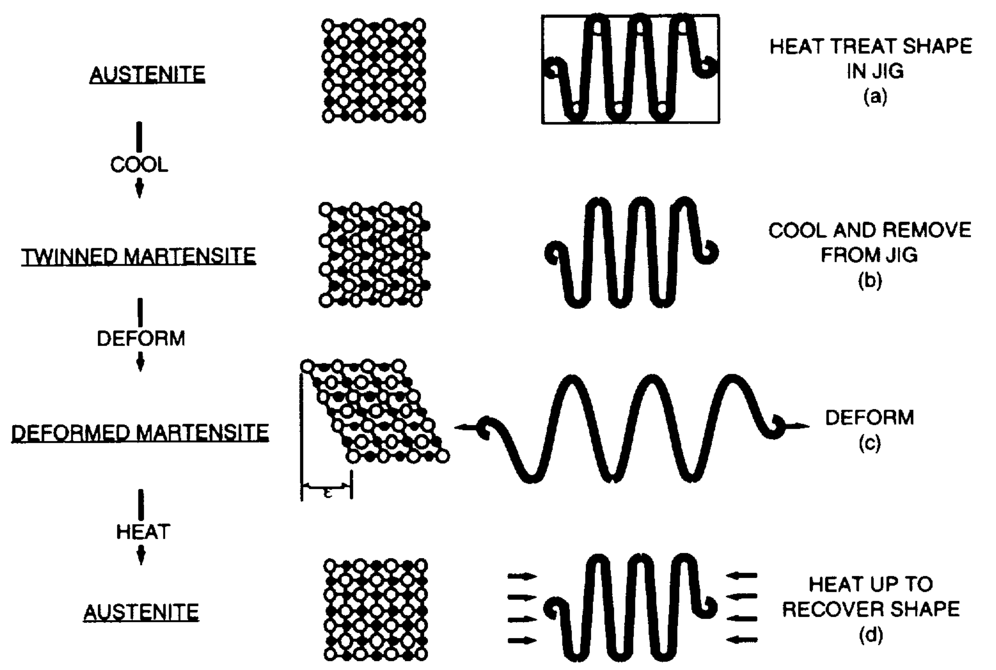
In order to achieve the most reliable, lowest mass system, Martin Marietta has developed a concept which combines deployment and structural elements of the solar array into an integral system. We are developing integral deployment / structural support elements composed either entirely of SMA, or SMA-composite laminates. Deployment will be accomplished by one of two methods; 1) heating of the SMA member, or 2) using heated SMA to control deployment of an elastic member. The integral deployment / structural elements, combined with a fully flexible array blanket will allow the attainment of array specific power levels greater than 100 W/kg.

Design Using Shape Memory Alloys - Shape memory alloys such as Nitinol (a Nickel-Titanium intermetallic discovered by the Naval Ordinance Laboratory in 1963) provide attractive engineering properties for use in lightweight actuator designs and adaptive composite structures. SMAs undergo a reversible crystalline phase transformation that is the basis of the "shape memory effect". The low temperature phase is a twinned, martensitic structure which is capable of large strain deformation (in excess of 10% in some alloys) with relatively little stress (approx. 70 MPa). The high temperature phase is a cubic based, austenitic structure with mechanical behavior more similar to conventional metals. When the martensite is deformed, and then heated, the original heat-treated shape is recovered. However, if the deformed martensite is constrained during heating, high recovery stresses evolve (>690 MPa is possible in some alloys). A combination of the two effects allows SMAs to produce mechanical work with the application of heat. Using wire as an example, Figure 1 demonstrates the shape memory effect.

Despite their attractive capabilities, the utility of SMAs in the past has been limited due to a lack of understanding of their very interdependent force-length-temperature response and associated non-linear and hysteretic behavior, as well as the effects of creep, fatigue, and material property drift which results from transformational cycling. These effects have been under study at Martin Marietta to provide the basis for effective alloy processing and "training" before incorporation in applications. Moreover, recent development of analytic modeling theory has made possible effective engineering of optimized mechanisms and devices based on experimentally derived parameters from property-stabilized SMA material.

Shape memory alloys are an ideal solution to the deployment of a flexible lightweight solar array. Because the array has an extremely low mass and is deployed in a 0-G environment, the load carrying demands placed on the shape memory deployment system will be very manageable. In addition, since the system requires one mission cycle (and 5 to 10 test cycles) SMA stability and non-linearity is not a major issue. However, several design challenges remain, including the use of the SMA material as structural elements in a martensitic state, and forming the required support geometries. As shown in Figure 2, the yield strength for a typical NiTi SMA is much lower for martensite than for austenite. As a result, SMAs are seldom used in their martensitic state as structural members. However, because lightweight arrays experience minimal on-orbit loads, using the SMA elements in their martensitic state is feasible. Benefits of reduced mechanism mass and associated interfacing hardware must be traded against the increased structural mass associated with the shape memory elements. Our studies indicate that an integral SMA structure provides the lowest mass deployment/support system.

Structural and Deployment Concepts - Our deployment concept employs a novel expansive roll-out approach. In the stowed configuration the array blanket and support elements are rolled-up together in a bundle approximately 30 cm in diameter. Activation of the primary deployment elements causes the bundle to expand and un-roll. The fully flexible solar blanket (including flexible solar cells) allows a smooth deployment with very little resistance applied to the deployment elements. When the deployment elements fully straighten at their full activation, this stage of deployment is complete.



Demonstration of the shape memory effect -- First the wire is heat treated into the desired shape using a restraining jig (a) at an annealing temperature well above the alloy's transformation temperature. After the austenitic wire cools, it transforms into martensite with no shape change (b). The twinned martensite is easily deformed (c) generating the strain shown. When the spring is heated above its transformation temperature, the crystal structure reverts back to austenite, whereupon the original shape is recovered (d). Steps c and d may then be repeated. Because this transformation proceeds with the speed of sound, the application of heat to a shape memory element is usually the rate limiting step.

Figure 1 Description of the Shape Memory Effect

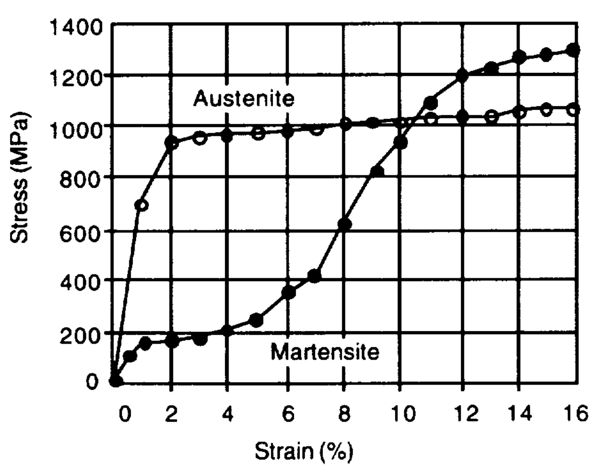


Figure 2 NiTi Stress/Strain Curve

The expansive roll-out deployment concept has several inherent advantages over other deployment approaches. High reliability can be achieved since there are no moving parts, drive mechanisms, or latches (except for launch restraint). Also, the absence of these components and associated interface hardware leads to the lowest mass deployment and structural support system achievable. Further, unlike systems using deployable masts or other elaborate deployable truss works, the low part count lends itself to rapid and efficient production. Finally, the concept allows manual restowage for repeated deployments using the actual mission hardware. This feature further improves the reliability of the system by allowing functional ground testing of all hardware.

In order to obtain the greatest structural efficiency, yet maintain a single spacecraft attachment point, the array configuration should be approximately square rather than rectangular. However, to achieve reasonable maximum dimensions in the stowed configuration, a nearly square array must use a two stage deployment sequence. While a two-stage deployment adds some complexity and risk, use of SMA hinges for the secondary deployment makes the additional deployment step simple and reliable. The overall deployment sequence is shown in Figure 3.

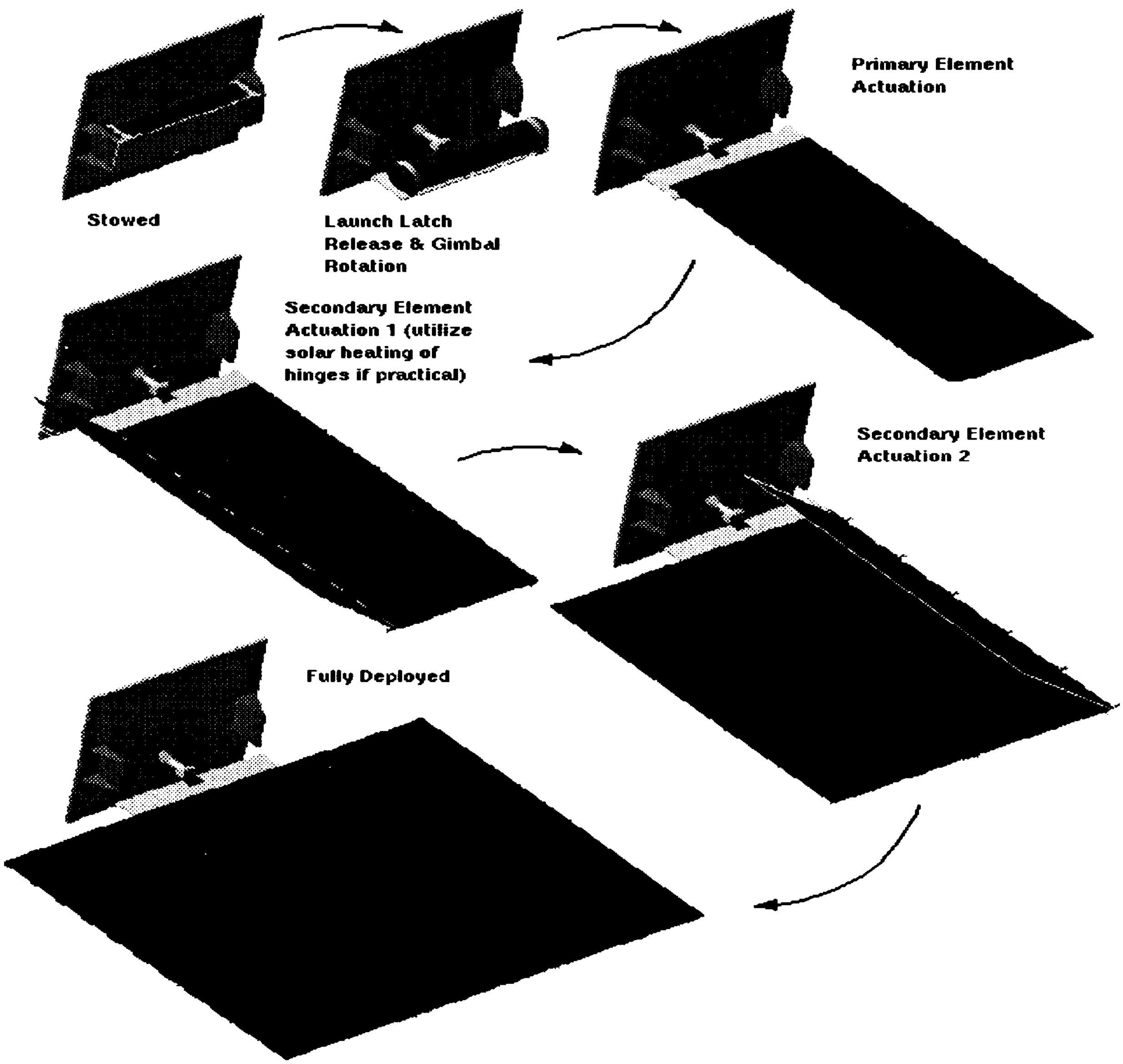


Figure 3 Array Development Sequence

A variety of primary support element geometries are under development and evaluation. These include the deployable element cross-sections shown in Figure 4. Initial analysis of potential cross-sections suggests that while the SMA tube configuration is most structurally efficient, it may be difficult to package repeatedly. Alternatively, while the arc cross-section is most easily fabricated and packaged, lateral / torsional buckling of arc cross section makes a suitable design difficult to achieve unless additional lateral support is provided. The array blanket interface could provide the necessary lateral support to make use of the arc cross-section viable. The two-part, joined cross section is also a candidate, but requires attention to the joint design and reliability, as well as requiring more assembly time.

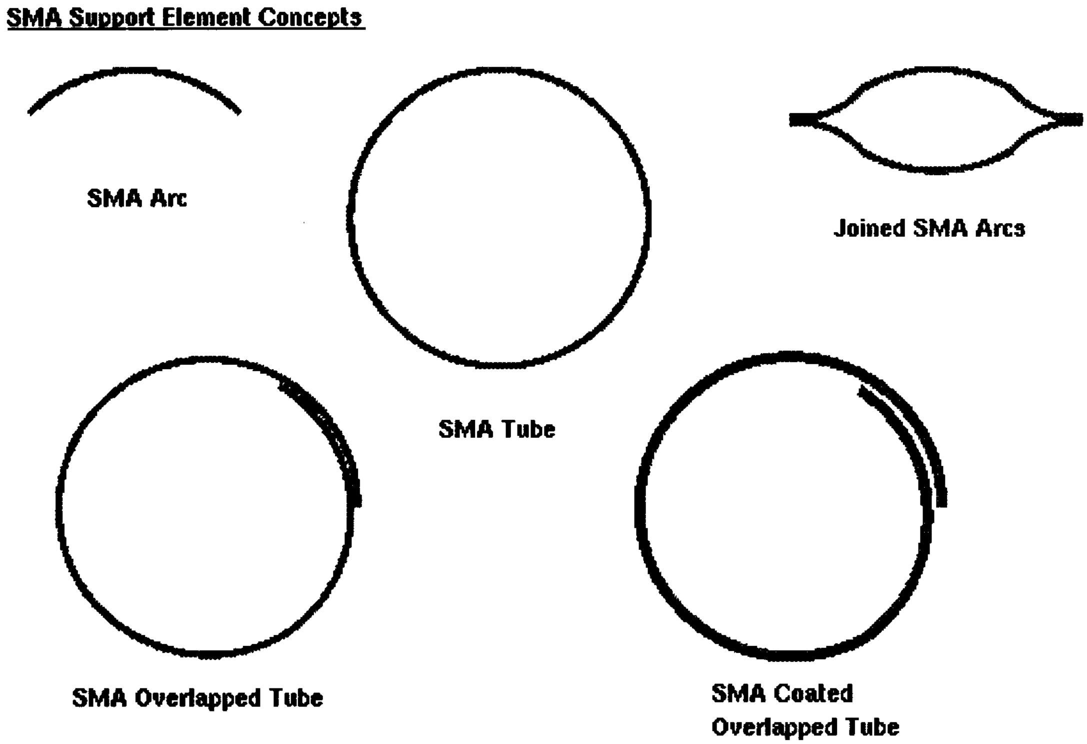


Figure 4 Primary Support Member Concepts

If deployed loading and natural frequency requirements are greater than about 0.2 G and 0.25 Hz respectively, an SMA coated STEM-type element must be implemented. In this concept, the deployment force is generated by elastic energy stored in the STEM. An SMA foil layer sputtered on the STEM serves to control the deployment. Without an SMA control layer, the STEM would spring out in a violent and unpredictable fashion, thereby disturbing the spacecraft and possibly damaging the array blanket. Prior to blanket release, the SMA would be heated to hold the STEM rolled-up. With gradual cooling, the STEM would overcome the force generated by the SMA coating and deploy in a smooth, predictable fashion. In order to evaluate the viability of this concept, the required SMA coating thickness and thermal control to avoid post deployment deflection due to solar heating must be determined.

Secondary deployment can be accomplished using SMA activated flexural hinges. These hinges may be all SMA or a coated beryllium-copper flexure (Fig. 5). This approach enjoys the same advantages as those discussed for the primary deployment elements. A potential advantage of the SMA hinges is the capability to achieve passive deployment from solar heating. This would further simplify the system but requires a better knowledge of the thermal environment.

SMA actuated deployment is accomplished by inducing a temperature change in the “as stowed” shape memory elements. Increasing temperature causes the element to experience a phase transformation

SMA HINGE

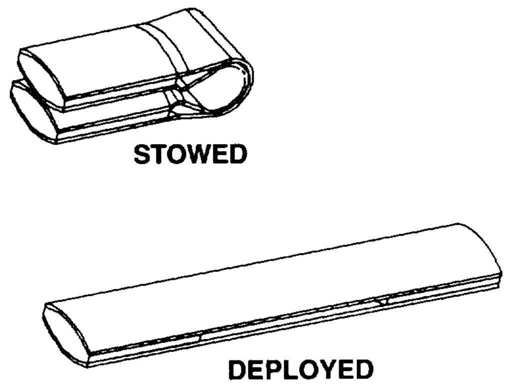


Figure 5 SMA Flexure Hinge

from a fully martensitic condition to an austenitic state generating the necessary deployment force and displacement. The change in temperature can be generated by passively exposing the array to solar radiation or using a control system which regulates current flow to a thin film heater placed in intimate thermal contact within the actuation element

Lightweight Gimbal Technology - The burgeoning demand for lightweight spacecraft and solar array technology requires development of an ultra-lightweight two-axis gimbal drive for solar array positioning. In response to this requirement, Martin Marietta's Mechanical Research & Technology Group has developed a new design for a lightweight gimbal to meet small spacecraft solar array pointing requirements.

The basic requirement for the SMA gimbal was to position a small rigid panel with a slow and smooth motion during a fixed orbit with a two-axis rotation of +/- 90° in elevation and +/- 175° in azimuth. Owing to small orbital loads, the torque requirement was determined to be 1 N-m; primarily due to the bearings and cable management system. In order to keep a 2:1 torque margin the cable management system had to be smooth and consistent. As always, weight was an important consideration and a 1 kg maximum was set as a design goal. Power and drive efficiency was also considered and limited to 3.0 watts average and 5.0 watts peak. The gimbal was required to maintain position and stiffness with the power off.

The gimbal design that was developed to meet these requirements uses shape memory alloy springs in a unique way to obtain a smooth rotary positioning. Figure 6 shows a photograph of the gimbal engineering development unit. A proprietary SMA drive technique, used for both the azimuth and elevation drives, results in significant reduction of the control electronics associated with brushless motor drives.

Another requirement that must be addressed is gimbal stiffness and holding torque. We have incorporated a brake into the design which passively releases when the SMA drive is activated, and then engages after power is removed. There are two brakes per axis and they act as a "no-backdrive" mechanism which stabilizes the gimbal position.

During the evolution of this design, many methods to reduce structural weight were implemented. The gimbal housings are constructed using graphite polycyanate composite material combined with the high performance composite plastic, Torlon. These two materials are used together, and in some cases co-cured, to produce a unique hybrid housing that is strong and very lightweight. This technique allows housing design without complex composite lay-ups or inefficient weight reduction machining.

Flexible Blanket Design - The photovoltaic blanket subsystem for our solar array concepts will be fully flexible. Blanket layup consists of a laminate printed circuit, adhesive, and monolithically-integrated CIS

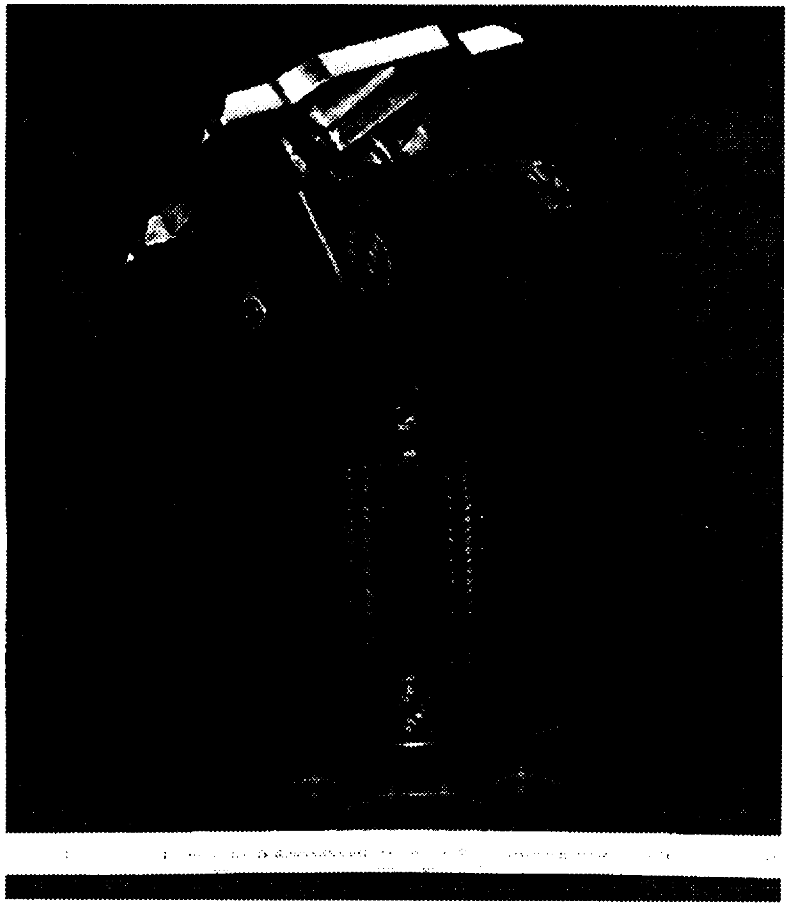


Figure 6 SMA Gimbal Engineering Development Unit

modules which already include thin-film replacement for conventional coverglass. Because the large size of the modules (nominally 30 x 30 cm) allows for passing circuitry underneath while still maintaining electrical connections, it is possible to lay-up the printed circuit in a single layer. Electrical interconnects are made from the module to the interconnect pads via soldered flexible jumpers parallel to the rolling axis. The printed circuits laminated into the blanket are used to convey array power to a flexible bus connector at the edge of each blanket section

ARRAY FUNCTIONAL AND ENVIRONMENTS TESTING

Extensive functional and operational testing of flexible arrays is planned. Tests include power generation, deployment, thermal vacuum, and random vibration. The following paragraphs summarize key aspects of the planned testing.

Power Generation - While array mass can easily be measured, solar simulation requires specialized equipment. Individual development modules up to 15 cm x 15 cm (6.0 in. x 6.0 in.) will be tested in a computer-controlled facility capable of testing these modules in AM0 and AM1.5 insolation while monitoring temperature as the modules and array are heated/cooled. I-V and quantum efficiency characteristics will be measured by sweeping in both directions, as well as standard test methods developed for space arrays by NASA Lewis Research Center. Standard solar cells will also be measured to provide a reference signal. Modules up to 30.5 cm x 122 cm, as well as the complete array, can be tested in Martin Marietta's large space chamber/space simulator laboratory. Continuous solar spot size of 4.9 m (16 ft) diameter from 0.35 to 1.4 sun insolation can be realized. In addition, because of the uncertainty in timedependent phenomenon observed in some polycrystalline materials, modules and array components will also be tested in the Large Area Pulsed Solar Simulator (LAPSS) which can accommodate 5.2 m (17 ft) spot diameter and the data will be used to develop a standard comparison between the test methods.

Deployment Testing - A critical aspect of the solar array system is reliable deployment. In order to assure high reliability, deployment must be demonstrated using the actual hardware destined for the mission. To this end, we plan to design and conduct deployment tests of the assembled solar array. An advantage of our integral SMA structure is the capability to re-stow (manually) the array and repeat the deployment sequence multiple times. However, due to the highly flexible nature of our array concept, all aspects of the deployment cannot be tested simultaneously. We therefore plan to use a relatively simple gravity off-load suspension to test the primary and secondary deployment using separate test setups. This will require a test approach utilizing long (> 35 m), lightweight suspension wires supporting the array in two alternate test configurations; a primary deployment test, and a secondary deployment test.

Thermal Vacuum Testing - Thermo-structural array response is of critical importance to successful operation in the space environment. The low mass and stiffness of the structure makes it susceptible to thermal distortion and buckling. Our planned testing includes subjecting flexible arrays and components (in deployed configuration) to anticipated low and high temperature extremes (as shown in Figure 7, typical extremes are -200°C to 80°C, although actual values will depend upon the optical characteristics of the modules) and measuring the thermo-structural response and power output.

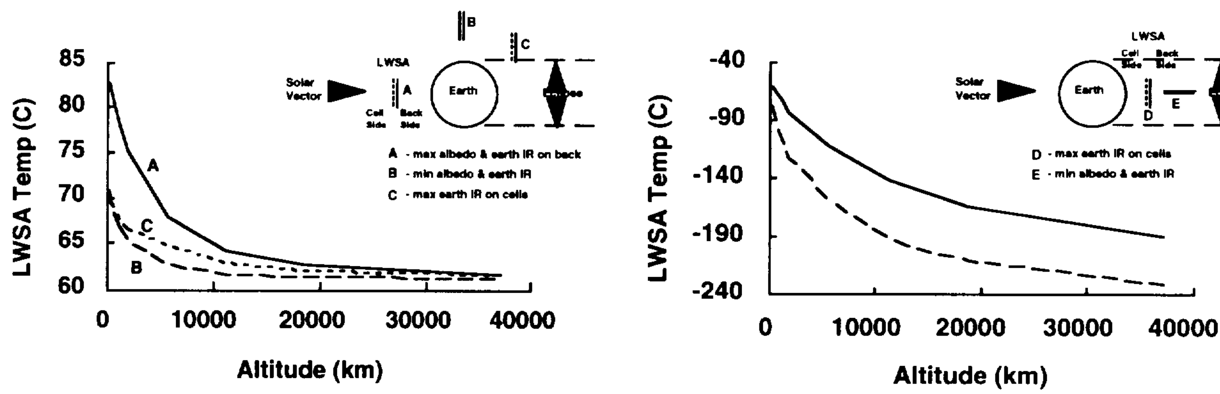


Figure 7 Array Temperature Versus Altitude

Launch Vibration Testing - Vibration and quasi-static loading during launch will produce the highest load levels experienced by the solar array in its stowed configuration. Typically, stresses due to random vibration response far exceed those caused by quasi-static accelerations. Therefore, we plan to conduct a random vibration test on the stowed solar array system to proto-flight levels using a Delta or Pegasus input spectrum. Analyses will be performed to predict peak load levels and verify that combined launch quasi-static and random loads are enveloped.

PLANNED ARRAY DEVELOPMENT

Martin Marietta has recently been awarded two Government sponsored solar array programs; a Phase A INSTEP study (NASA) to define a thin-film, deployable array flight experiment, and an Air Force Phillips Lab contract to develop a prototype lightweight, deployable solar array.

Our INSTEP program will focus on the thermo-structural and electrical performance of flexible thin-film solar arrays. The experiment, as proposed , is summarized in Figure 8.

The Phillips Lab / Martin Marietta LWSA program will combine the technologies discussed above to meet

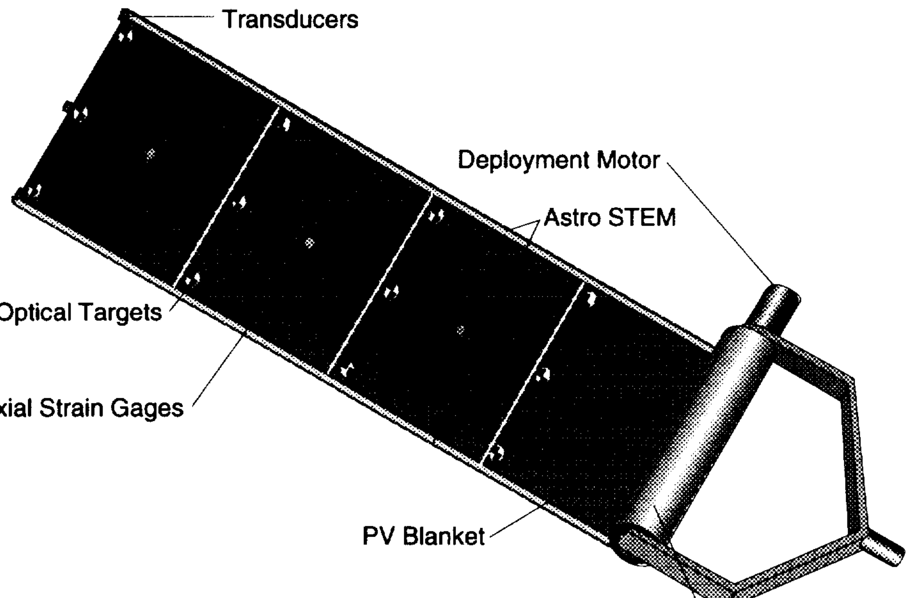


Figure 8 INSTEP Flexible Thin-Film Solar Array Experiment

the needs of future Air Force spacecraft. We have developed an initial configuration for our concept that employs integral deployment/structural elements, an ultra-lightweight gimbal, and a fully flexible blanket using flexible CIS solar cells. This configuration, shown in Figure 9, is sized for at least 750 W EOL power generation. The integral deployment/structural elements are sized for 0.1 G load in the deployed configuration. Use of the structural innovations previously described, together with the lightweight and flexible blanket, allow our concept to achieve a specific power of 160 W/kg EOL under nominal LEO operating conditions.

Estimated total system mass for achieving 500 W, 750 W, and 1000 W power levels is listed in Table I for the LEO and GEO environments at both 28°C and 60°C operating temperatures. These mass estimates were derived from our 750 W baseline concept. Note that the EOL specific power output exceeds 150 W/kg after 7 years in the LEO environment and can approach that level after 10 years in the GEO environment if operating temperatures can be kept relatively low.

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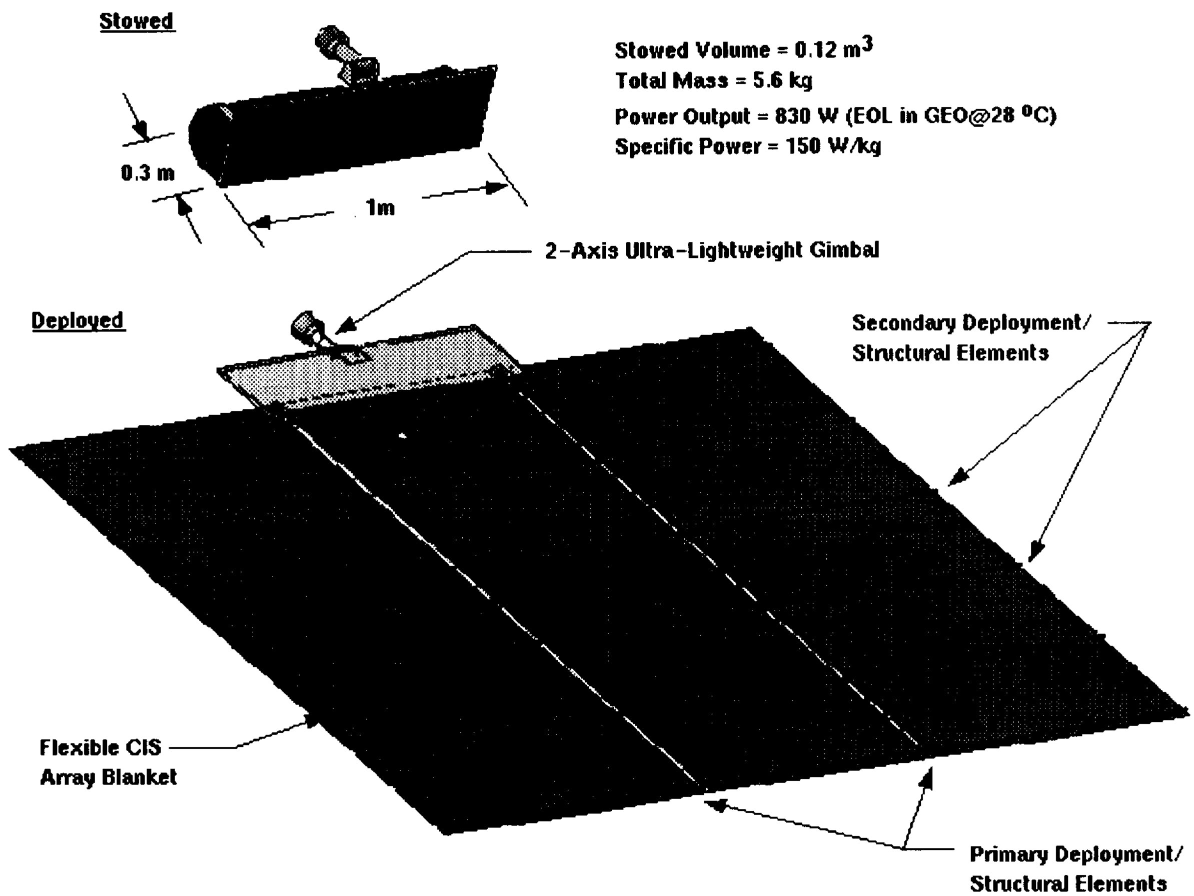

Figure 9 LWSA Baseline Design Concept

Table 1 EOL Specific Power Estimates for LEO and GEO Mission Profiles

		Low Earth Orbit (7 yrs, 1000 km)						Geosynchronous Orbit (10 yrs, 36000 km)					
Power (W)	Temp (° C)	Area (m2)	Blanket (kg)	Structure (kg)	Gimbal (kg)	Total (kg)	**Specific Power (W/kg)**	Area (m2)	Blanket (kg)	Structure (kg)	Gimbal (kg)	Total (kg)	**Specific Power (W/kg)**
500	28	4.0	1.6	0.5	1.0	3.1	**160**	4.6	1.8	0.7	1.0	3.5	**140**
	60	4.4	1.8	0.6	1.0	3.4	**150**	5.1	2.0	0.8	1.0	3.8	**130**
750	28	6.0	2.2	1.2	1.0	4.4	**170**	6.8	2.6	1.5	1.0	5.1	**150**
	60	6.7	2.3	1.4	1.0	4.7	**160**	7.5	2.8	1.8	1.0	5.6	**130**
1000	28	8.1	3.0	2.1	1.0	6.1	**160**	9.1	3.4	2.6	1.0	7.0	**140**
	60	8.9	3.3	2.5	1.0	6.8	**150**	10.1	3.7	3.3	1.0	8.0	**130**