THE DEVELOPMENT OF HIGH-TEMPERATURE COMPOSITE SOLAR ARRAY SUBSTRATE PANELS FOR THE MESSENGER SPACECRAFT

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ABSTRACT

The MESSENGER (MErcury Surface, Space, ENvironment, GEochemistry, and Ranging) spacecraft will be the first spacecraft to orbit the planet Mercury. Designed and built by The Johns Hopkins University Applied Physics Laboratory (JHU/APL), the spacecraft will orbit the planet for one year. In order to reduce cost and schedule of this NASA Discovery Mission, the solar arrays were required to be constructed of conventional, space-qualified materials. System thermal, mass, and stiffness requirements dictated that the panel facings be fabricated from a high thermal-conductivity and stiffness pitch-fiber composite material capable of withstanding short-term temperatures as high as 270°C. A toughened, 177ºC-curing cyanate ester composite material resin system with extensive flight heritage was chosen, with a post-cure used to extend the glass-transition temperature closer to the maximum predicted temperature. A lengthy development program was conducted at JHU/APL to provide assurance that the materials and processes chosen were capable of performing under such a demanding thermal environment. The results of this program will be applicable to other high-temperature spacecraft applications of advanced pitch-fiber cyanate ester composite structures.

KEYWORDS: Sandwich Construction, Solar Array, Spacecraft.

INTRODUCTION

The solar array substrate panels for the MESSENGER spacecraft are required to survive extreme levels of solar heating during the mission to Mercury. During normal spacecraft operation around the planet, the solar arrays will experience at least 278 eclipse cycles, during 28 of which the temperature will vary from –100°C to +150°C at a rate of 70°C/minute. During an attitude anomaly, the structural portion of the panel could reach 270ºC in the event the arrays were directly pointed at the Sun near Mercury, where insolation is up to eleven times greater than that at Earth[1]. The structural laminate must survive these conditions, which are beyond the glass-transition temperature of the resin system.

The solar array for MESSENGER consists of two rotating wings, one side of which is populated with a 30/70 ratio of 5.5-mil-thick gallium arsenide (GaAs/Ge) cells and Optical Solar Reflectors (OSRs) (Figure 1). The heat absorbed by the cells is conducted through the substrate skins to the OSRs, where it is re-radiated. In order to provide sufficient conduction in the preferred direction, an asymmetric laminate configuration was selected for the facings. System requirements dictated that the panel facings be fabricated from an advanced pitch-fiber composite material with very high thermal-conductivity and stiffness-to-weight ratio. Aluminum honeycomb core was chosen to minimize through-thickness temperature gradients in the sandwich panel. Between the cells and the forward substrate skin is an electrically isolating layer of Kapton film. Vapor Deposited Aluminum (VDA) Kapton film was chosen for the back-facing outside surface, bonded with the aluminum side to the facings to provide the proper thermal properties. Solid composite inserts were chosen at all mounting locations to minimize coefficient of thermal expansion (CTE) mismatch between the inserts and the skin.

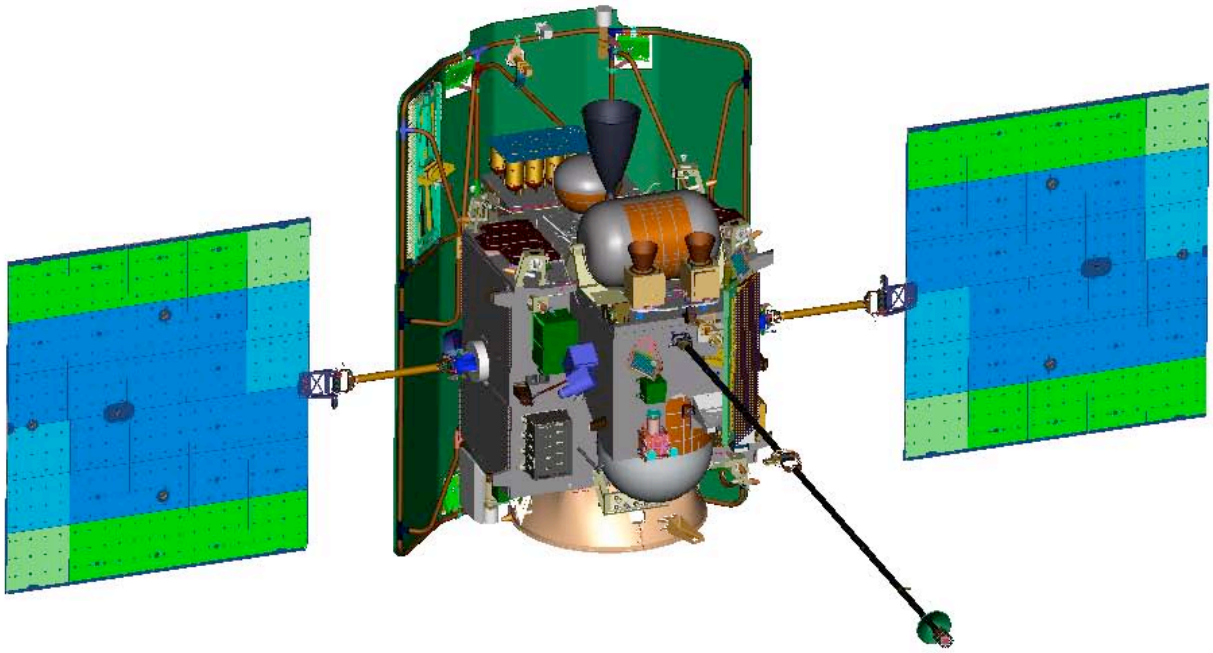

Figure 1. MESSENGER spacecraft showing solar array panels deployed. The VDA Kapton tiles shown on the backside of the panels provide thermal control.

The substrate development was determined to be a critical mission risk and schedule driver. The requirements and an initial concept were formulated; thermal and stress analyses were performed; materials were selected; representative test panels were fabricated in-house and tested as part of the process development; materials and process specifications were written; and vendors were solicited for representative test panels.

Extensive problems experienced with test panels supplied by vendors early in the program focused attention on materials and process issues. After thermal vacuum testing, these test panels exhibited porosity, extremely low flatwise tensile strength, and delamination of the Kapton and VDA Kapton outer layers. An in-house program was initiated to develop a robust

substrate laminate by optimizing the materials and processes used. Processing trials conducted at JHU/APL were able to reduce risk and provide improved test panels for population with cells for environmental testing[1]. This knowledge was then transferred to the vendors chosen to compete for the fabrication of the flight substrates. Several scale-up problems, not foreseen during the fabrication of the smaller test panels, were encountered during the fabrication of the flight substrates.

APPROACH

Design and Analysis

Figure 2 shows the spacecraft in the launch configuration with the solar arrays stowed. Each panel interfaces to the spacecraft body through three ball and socket joints. The panel is preloaded against each joint by a single release located roughly at the centroid of the joints. The preload provides a 'pop' from strain energy stored in the panel at the initiation of release. The titanium wing arm is also held with a pyrotechnic release, and the arm is sequenced to be released after the panel has fully deployed and settled. This two-step process avoids the need to provide synchronization of panel and arm if a single release point is used.

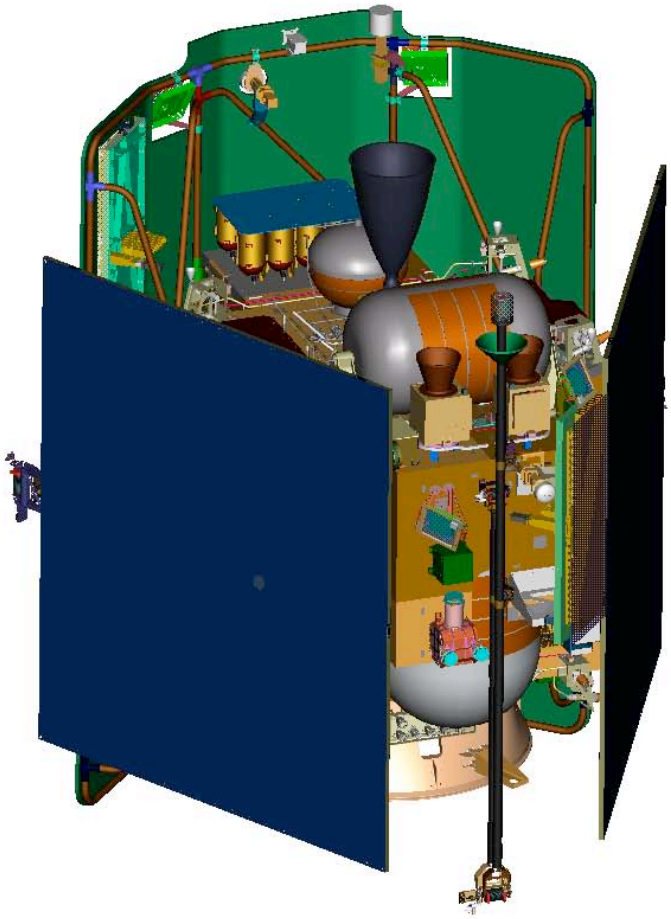

Figure 2. MESSENGER spacecraft showing solar array panels in the stowed position.

The finite element model (FEM) of a deployed array wing shown in Figure 3 clearly reveals the grouping of the four interface points on one-half of the panel. When stowed against the spacecraft, the hinged half of the panel is cantilevered with no support. This large cantilever, combined with the relatively low compressive and in-plane shear strengths of the laminate, required rather thick panel faces with doublers. The shape of the doublers in Figure 3 shows the location of the high bending stresses in the panel when stowed.

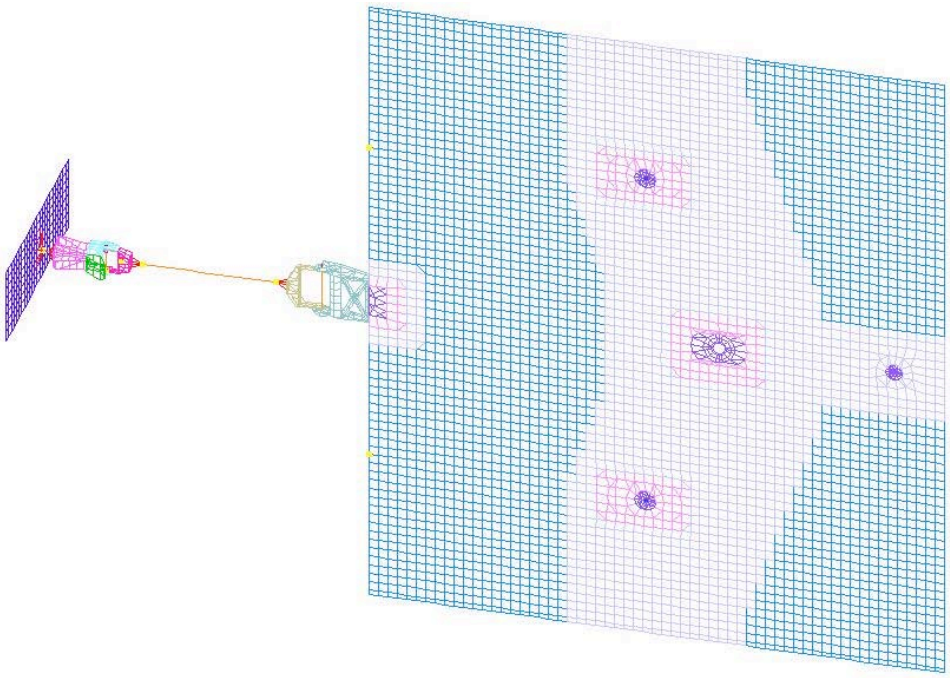

Figure 3. Finite element model of deployed solar array. The doublers follow the stress distribution in the sandwich faces due to bending of the cantilevered panel in the stowed position.

Initial trial laminate configurations were evaluated using the CompositePro laminate analysis code and hand loads analysis. After the asymmetric facing lay-up (Table 1) was shown to perform well thermally, a FEM of the design was constructed, which evolved into the model presented. Ply-by-ply stress analysis was performed using the laminate analysis capability of NASTRAN™. The doublers were tailored for shape by plotting both stress and the Tsai-Wu failure indices using the pre- and post-processor FEMAP™. Figure 4 shows a stress distribution plot from FEMAP™ for a typical load case.

Table 1. Substrate Panel Ply Table

Ply No.	Type of Ply	Ply Angle	Ply Location	Ply Thickness (mm)
1	Kapton, 200HPP-ST	N/A	Full	0.051
2	RS-4A, Unsupported	N/A	Full	0.102
3	K13C/RS-3, 85 FAW	0	Full	0.064
4	K13C/RS-3, 85 FAW	22.5	Full	0.064
5	K13C/RS-3, 85 FAW	45	Full	0.064
6	K13C/RS-3, 85 FAW	67.5	Full	0.064
7	K13C/RS-3, 85 FAW	90	Full	0.064
8	K13C/RS-3, 85 FAW	112.5	Full	0.064
9	K13C/RS-3, 85 FAW	135	Full	0.064
10	K13C/RS-3, 85 FAW	157.5	Full	0.064
11	K13C/RS-3, 85 FAW	0	Full	0.064
12	RS-4A, Supported	N/A	Full	0.051
13	Doubler B	0	Doubler B	0.508
14	RS-4A, Supported	N/A	Full	0.051
15	Doubler A	0	Doubler A	0.508
16	RS-4A, Supported	N/A	Full	0.051
	Aluminum Honeycomb		Full	14.00
17	RS-4A, Supported	N/A	Full	0.051
18	Doubler A	0	Doubler A	0.508
19	RS-4A, Supported	N/A	Full	0.051
20	Doubler B	0	Doubler B	0.508
21	RS-4A, Supported	N/A	Full	0.051
22	K13C/RS-3, 85 FAW	0	Full	0.064
23	K13C/RS-3, 85 FAW	157.5	Full	0.064
24	K13C/RS-3, 85 FAW	135	Full	0.064
25	K13C/RS-3, 85 FAW	112.5	Full	0.064
26	K13C/RS-3, 85 FAW	90	Full	0.064
27	K13C/RS-3, 85 FAW	67.5	Full	0.064
28	K13C/RS-3, 85 FAW	45	Full	0.064
29	K13C/RS-3, 85 FAW	22.5	Full	0.064
30	K13C/RS-3, 85 FAW	0	Full	0.064
31	RS-3C, Unsupported	N/A	Full	0.102
32	Kapton, VDA	N/A	Full	0.051

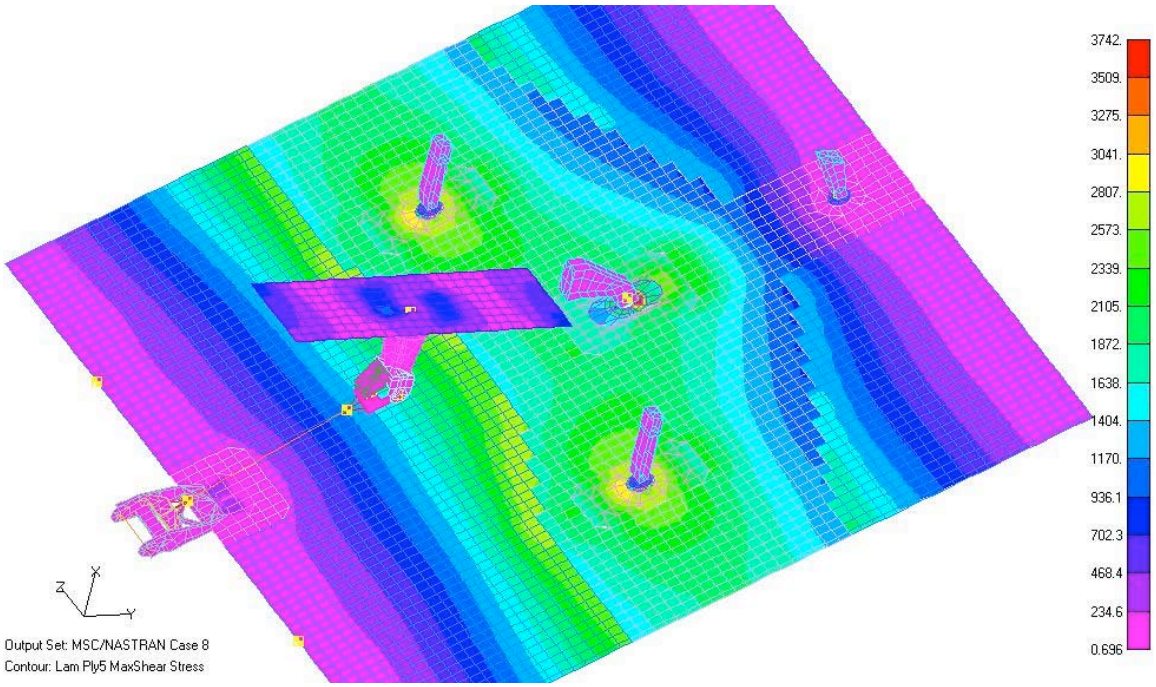

Figure 4. A typical plot of the finite element model stress distribution in the sandwich faces due to bending of the cantilevered panel when in the stowed position.

Materials Selection

The solar array substrate panel facings are made from K13C2U/RS-3 uni-directional carbon-fiber cyanate ester pre-preg supplied by YLA, Inc., at a fiber areal weight of 85 gm/m^2, which equates to a thickness of 0.06 mm (0.0025 inch) per ply. The K13C pitch-based carbon fiber was selected for its high tensile modulus and high thermal conductivity. YLA's RS-3 cyanate ester resin was selected for its extensive flight heritage, toughness, low moisture absorption, low outgassing, and high glass-transition temperature. The panels incorporate 1/8 inch cell, 3.1 lb, 5056 vented aluminum honeycomb core selected for its high shear-stiffness-to-weight ratio and thermal conductivity.

The facings are co-cured to the aluminum honeycomb core with a layer of 209 gm/m^2 RS-4A supported cyanate ester film adhesive, also from YLA. The VDA and plain Kapton film outer layers are co-cured to the facings with layers of 209 gm/m^2 RS-3C and RS-4A unsupported cyanate ester film adhesive, respectively. The materials evaluated during development testing and their locations are shown in Figure 5.

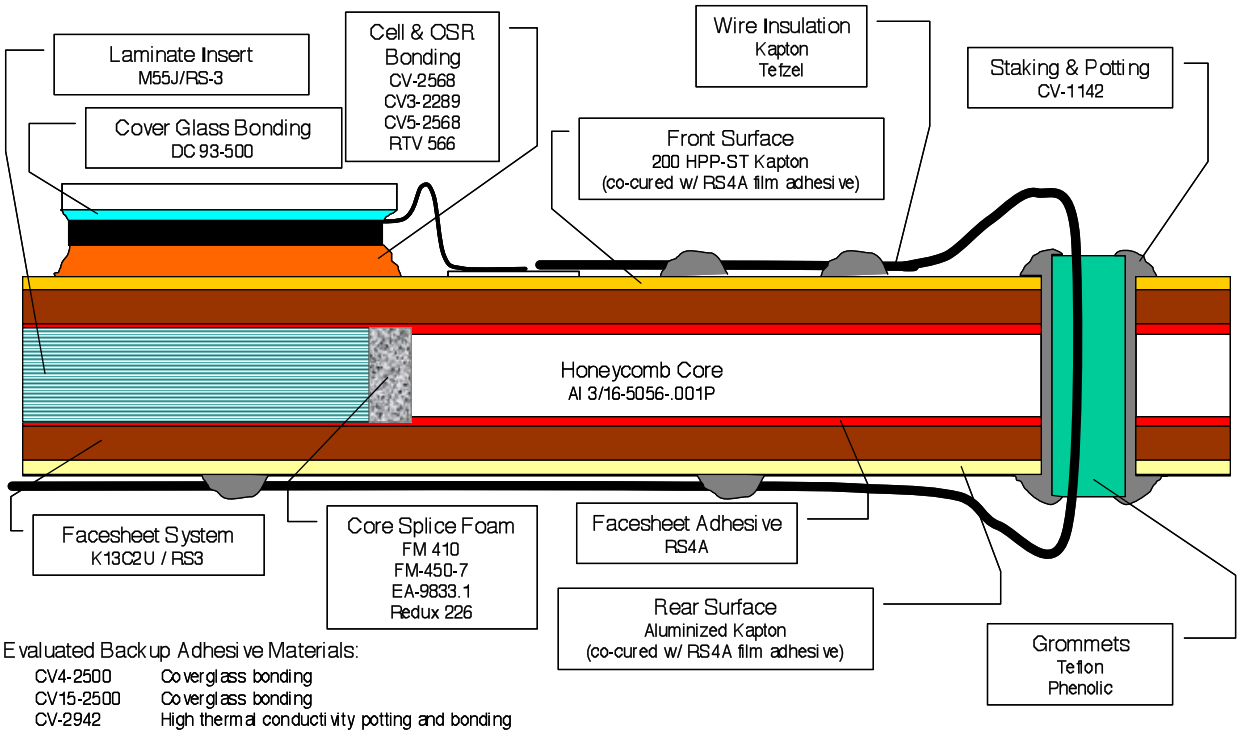

Figure 5. Candidate solar array component materials and their configuration.

The pre-cured doublers and triplers are bonded into pockets machined into the honeycomb core. Solid graphite/cyanate ester laminate disks with threaded titanium inserts are bonded into holes in the core using Cytec FM-450-7 bismaleimide foaming adhesive. The latter was selected for its high glass-transition temperature and ability to be co-cured with cyanate ester resin. The doublers and triplers were made from K13C1U/RS-3 uni-directional carbon fiber cyanate ester pre-preg supplied by YLA at a fiber areal weight of 47 gm/m^2, which equates to a thickness of 0.04 mm (0.0015 inch) per ply. The upper surface consists of a layer of 200HPP-ST Kapton film, which is the high-temperature version that has been treated for better adhesive properties. It is required as an electrical insulation layer on which the solar cells are mounted. The lower surface consists of a layer of VDA Kapton film with its aluminized surface bonded to the panel. It was selected for thermal emissivity properties.

Initially, the process development panels were cured in a 276-kPa (40-psi) autoclave cure cycle that included a ramp to 110°C with a two-hour hold, a ramp to 177°C with a two-hour hold, and a ramp to 60°C. Later, to eliminate porosity, panels were cured with a ramp to 88°C with a four-hour hold, a ramp to 177°C with a two-hour hold, and a ramp to 60°C. They were then post-cured to 232°C in a separate cycle under full vacuum. The panels were cured and post-cured on a large flat 25-mm- (1-inch) thick steel tool. A thin (1-mm or 0.04-inch) steel caul plate and slip sheet were used directly above and beneath the panel in both processes to ensure a smooth outer surface.

Mechanical and Environmental Testing

The mechanical and physical properties of the composite materials were measured by the supplier, YLA, in order to qualify the material to the materials specification. These properties were those of a solid laminate and are not representative of the co-cured laminate, since the co-curing process is known to result in reduced values due to the non-uniform pressure applied to the facing laminate by the honeycomb core. Therefore, mechanical testing of sandwich panel coupons, some of which were exposed to expected mission temperature extremes, was performed during the development phase of the program to evaluate the quality of vendor-produced panels, develop actual allowables, and assess the effect of elevated temperature exposure. Later, similar test panels were fabricated as travelers along with each flight substrate. They, too, were conditioned at the full range of temperatures and mechanically tested. The data were compared with the test data for each material lot and used as a means for acceptance of the panels.

The transverse tensile strength of the sandwich laminate was measured using the ASTM C 297-61 flatwise tension tests[2]. This test evaluates the adhesive bonds of the Kapton to the facings and the facings to the core, as well as the adhesion between the plies within the facings. Sandwich panel laminate samples were bonded to aluminum end blocks in the configuration specified by the test specification and tested in tension. The results, while indicative of the quality of the panel, are relatively qualitative and cannot be used to verify the design allowables.

The ASTM C 393-62 sandwich flexure bending test[3] was used to evaluate the bending strength and stiffness of the panel laminate. Load and deflection data are used to calculate tensile/compressive properties of facings. The specimens were sized so the facing would fail in compression.

Elevated temperature exposure of the test coupons was conducted by exposing them to different levels of simulated solar radiation heating through the use of a specially designed infrared thermal testing apparatus known as the JHU/APL "E-box." The box consists of a copper enclosure with independent temperature control installed inside a 9-m^3 vacuum chamber. The large chamber provides vacuum, electrical, and LN_2 interfaces, but it is not temperature controlled for the tests. This provides repeatable and uniform sample temperatures between -180° and 300°C at full vacuum. Test panels and sets of mechanical test coupons were exposed to thermal vacuum cycles to 175°, 250°, 270°, and 290°C in the E-box. The purpose of exposing coupons to 290ºC was to assess the margin in the design, since this was beyond the predicted maximum exposure temperature of 270°C.

The test coupons supplied by several of JHU/APL's preferred vendors early in the program exhibited extremely low flatwise tensile strength (Figure 6). This problem was the catalyst behind the program undertaken at JHU/APL to develop a solar array substrate that would meet all program requirements.

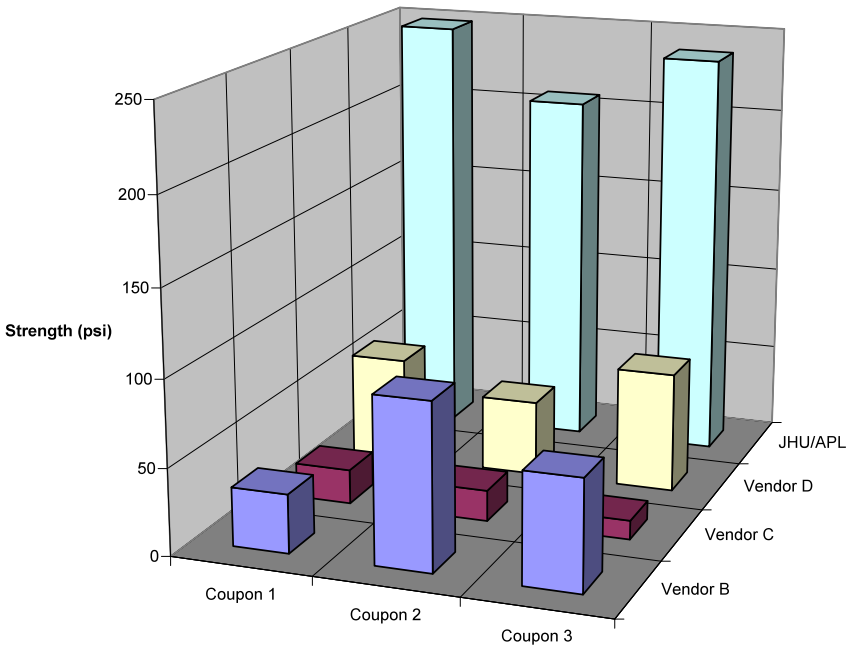

Figure 6. Early test data for flatwise tensile strength of test coupons from vendors and JHU/APL (1000 psi = 6.9 MPa)

Process Development and Qualification

As mentioned above, extensive problems experienced with test panels supplied by vendors early in the program focused attention on materials and process issues. These test panels exhibited delamination of the Kapton and VDA Kapton outer layers during thermal vacuum testing, porosity, and extremely low flatwise tensile strength. In order to improve the quality of the panels, a project was undertaken at JHU/APL to develop a robust substrate laminate by optimizing the materials and processes used.

An asymmetric laminate configuration was chosen for the substrate panel facings, as mentioned above (Table 1). Since excessive strains are generated in curing a free-standing asymmetric laminate, a co-curing approach was selected, where the facing laminate was cured and bonded to the honeycomb core in one cure cycle. The alternate approach, which is frequently used for asymmetric solar array substrate laminates, is to pre-cure the facings, allow them to roll up, then flatten them out to machine them and to bond them to the core. This technique is usually used on thin pan fiber laminates and was perceived as having too much risk for the relatively thick and brittle pitch fiber laminates required here. The risk with co-curing the facings is that the non-uniform pressure applied to the facing laminate will result in voids and reduced allowables.

In an early test panel, the use of a traditional high-temperature phenolic core splice foam material caused extensive blistering of the laminate adjacent to the insert. This behavior was traced to carbamate formation in facings due to the moisture evolution of the phenolic during the cure cycle. The use of the FM-450-7 bismaleimide core splice foam solved the problem.

The cause of the low values of flatwise tensile strength exhibited by coupons supplied by vendors early in the program was never identified. The problem was solved later in the program when the vendors used the revised JHU/APL process to fabricate qualification test coupons. The vendor qualification test coupons showed much better results (Figures 7-8). This improved performance was likely due to uniform use of the materials and process specifications supplied to the vendors, which incorporated the lessons learned in the development program. Critical process steps include careful cleaning and drying of the Kapton films prior to lay-up and long debulking cycles to eliminate air bubbles between the Kapton, film adhesive, and facing laminate. While there was still some variation between the vendor and JHU/APL-built coupons, no extremely low values were seen in the flatwise tensile and sandwich flexure test results. More importantly, the test results show little degradation in mechanical properties after exposure to elevated temperatures up to 290°C in a vacuum.

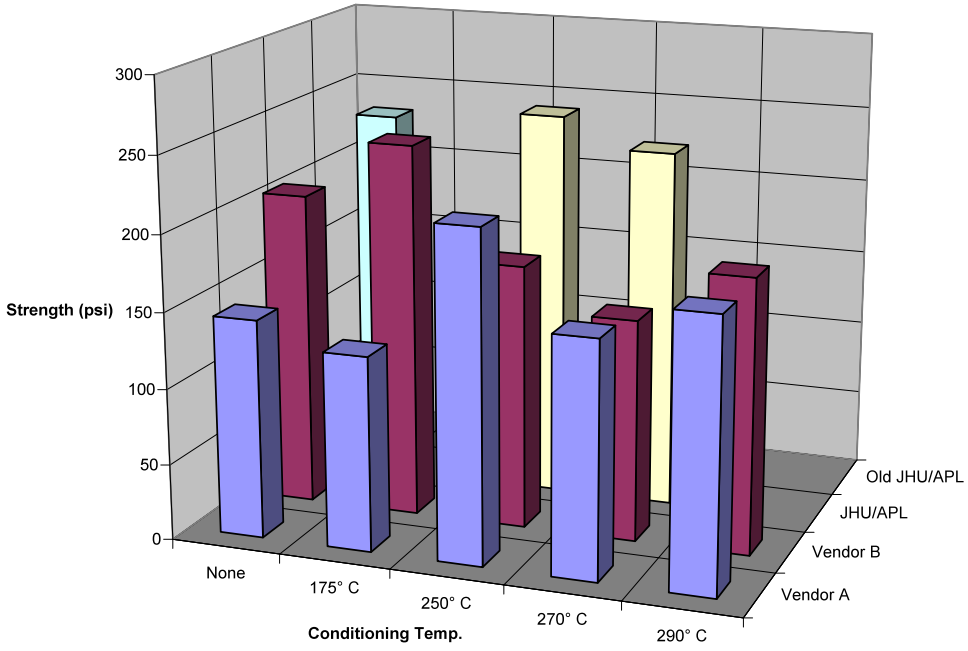

Figure 7. Development panel test data for average flatwise tensile strength vs. exposure temperature when fabricated using optimized process (1000 psi = 6.9 MPa)

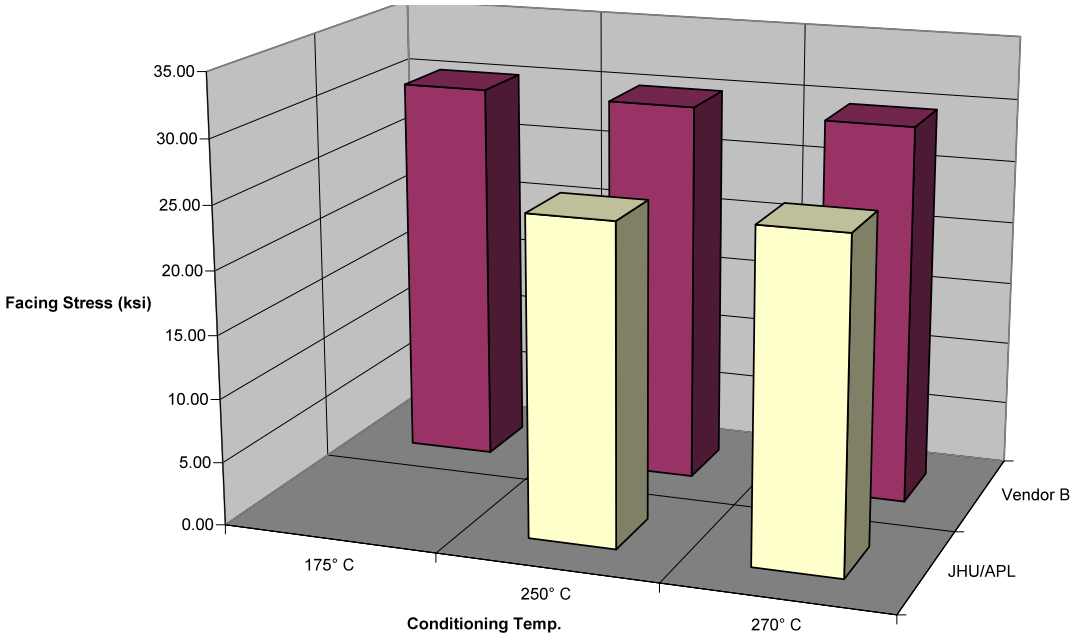

Figure 8. Development panel test data for average sandwich flexure strength (3-point bending) vs. exposure temperature when fabricated using optimized process (1 ksi = 6.9 MPa)

Early test panels used a layer of Cytec BR-127 epoxy primer on the Kapton and VDA Kapton to promote adhesion. While this approach has proven effective on other solar array substrate panel applications, the high-temperature exposure required in this application will not allow the use of the primer since the temperature exposure is beyond its limit. In another set of test panels, only the laminate resin was used to bond the Kapton to the facing. In later applications a discrete layer of unsupported film adhesive was used; nevertheless, these changes significantly reduced, but did not eliminate, the Kapton delamination problem. Since all flatwise tension coupons tested failed within the laminate, the bond between the Kapton films and the facings was assumed to be adequate.

Traveler test panels were fabricated along with each flight substrate (Figures 9-10). They, too, were conditioned at the full range of temperatures and tested. The data were compared with the qualification test data and used as a means for flight panel acceptance.

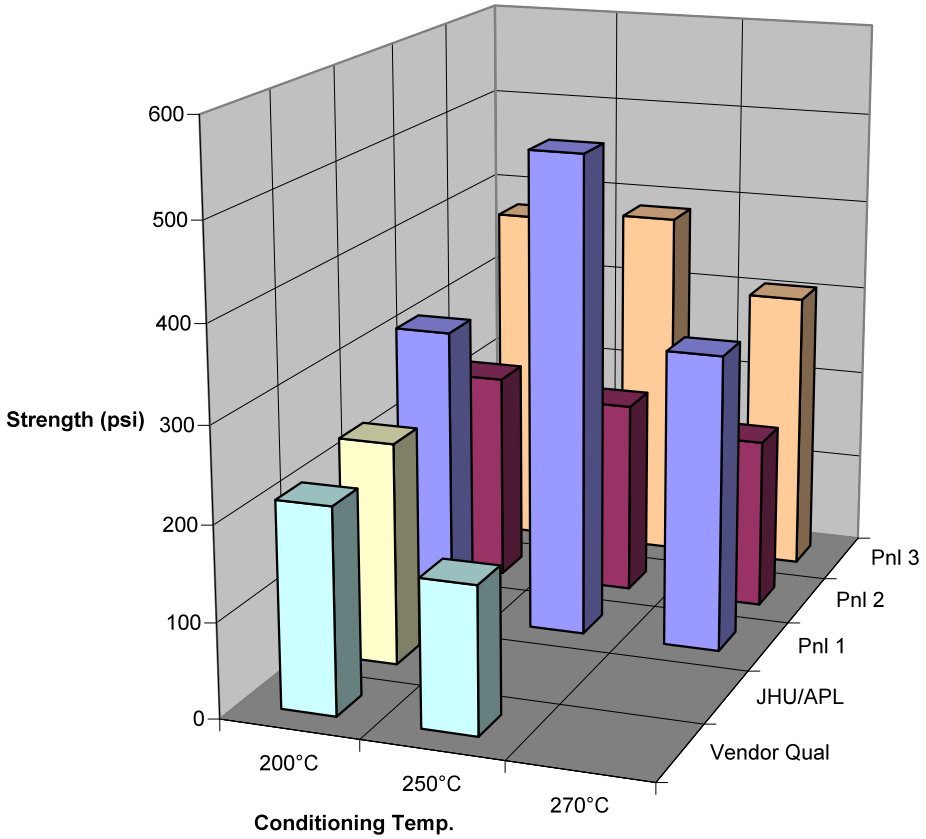

Figure 9. Flight panel test data, average flatwise tensile strength vs. exposure temperature (1000 psi = 6.9 MPa)

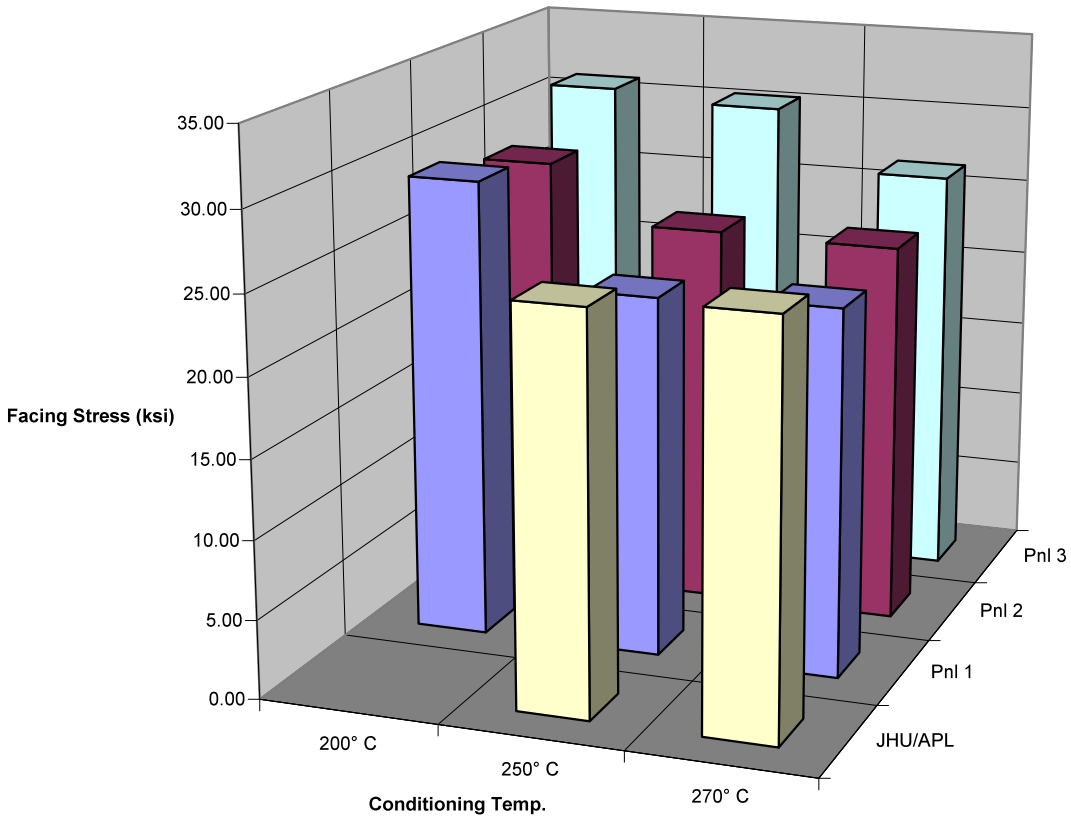

Figure 10. Flight panel test data, flexural strength (3-point bending) vs. exposure temperature (1 ksi = 6.9 MPa)

A microscopy study was performed on samples of several early test panels. The purpose was to investigate the cause of both the VDA Kapton delamination and the porosity that was suspected to exist in the laminate. A study performed by The Aerospace Corporation revealed significant porosity in panels fabricated by one of the vendors. The investigators suspected that the porosity was due to carbamate formation during the laminate cure. Alternately, however, poor consolidation pressure, which is an inherent problem with laminates co-cured directly to honeycomb core, was also thought to be the cause. Representative samples were cut from several panels fabricated at JHU/APL. These were mounted in epoxy, ground, and polished using standard metallographic techniques. Care was taken to grind the sample down far enough to ensure that sample cutting and preparation did not affect the analyzed areas. The samples were observed and photographed using an inverted microscope with a digital microscope camera connected to a data acquisition computer.

Extensive inter-ply porosity, primarily between honeycomb core cell walls, was observed in the co-cured laminates fabricated early in the program (Figure 11). Some micro-cracking was also observed. This effect is very similar to the porosity observed by researchers at The Aerospace Corporation in similar panels from another program. While they also observed fiber breakage at the cell walls in the vendor-fabricated panels, this type of damage does not appear in any of the early panels processed at JHU/APL.

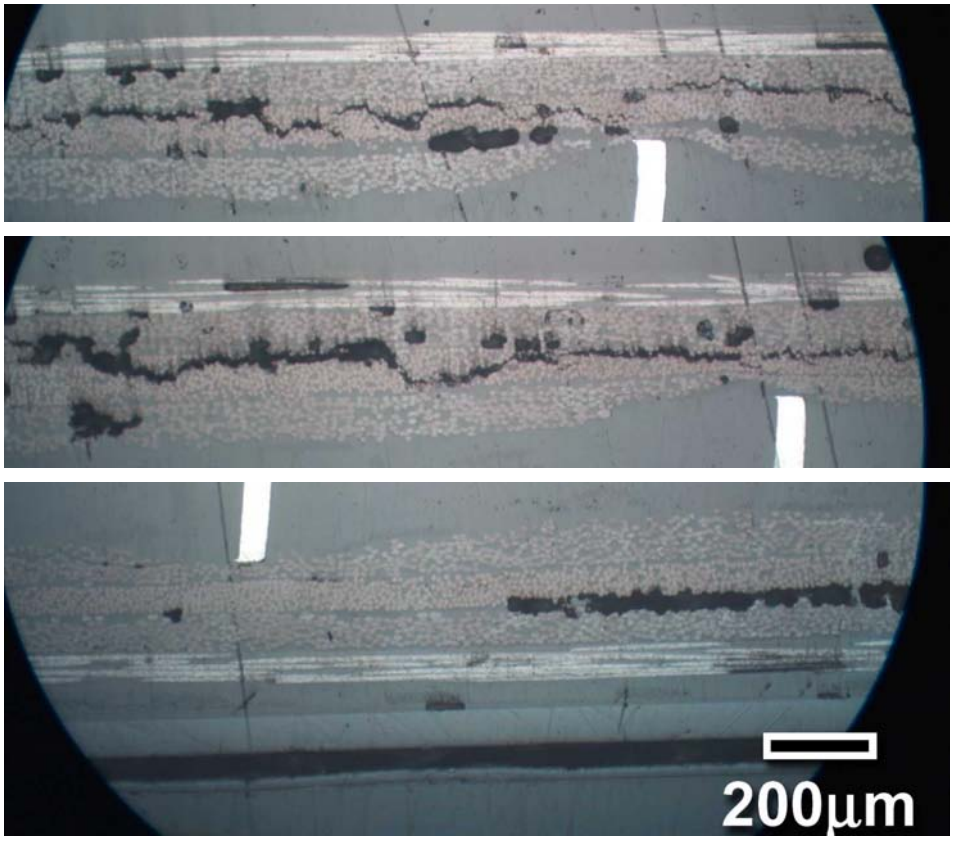

Figure 11. Cross-sections of co-cured composite laminate from early test panels fabricated at JHU/APL, showing inter-ply porosity primarily between the cell walls.

No porosity is observed in the pre-cured insert, doublers, or the co-cured laminate in the region adjacent to the insert and doubler in Figure 10. The areas adjacent to the insert and doublers are well supported during the cure. This finding implies that the observed porosity is due to poor consolidation pressure, which is inherent in a laminate co-cured directly to honeycomb core. If carbamate formation were the cause of the porosity, it would be evident throughout the co-cured laminate and in other laminates fabricated using the same process. Also, since the VDA Kapton delamination occurred in the area of the doublers, and this area should have had no porosity, then the delamination is probably not related to the porosity.

Pronounced intra-ply cracking is observed within one ply of the doublers that were co-cured into an early test panel (Figure 12). It is not evident in sections of the doubler that have not been co-cured into a panel. Therefore, the cracking occurs during the co-cure cycle or in the subsequent post-cure of the panel. Its exact cause is unknown, but it is likely to be the result of residual thermal stresses.

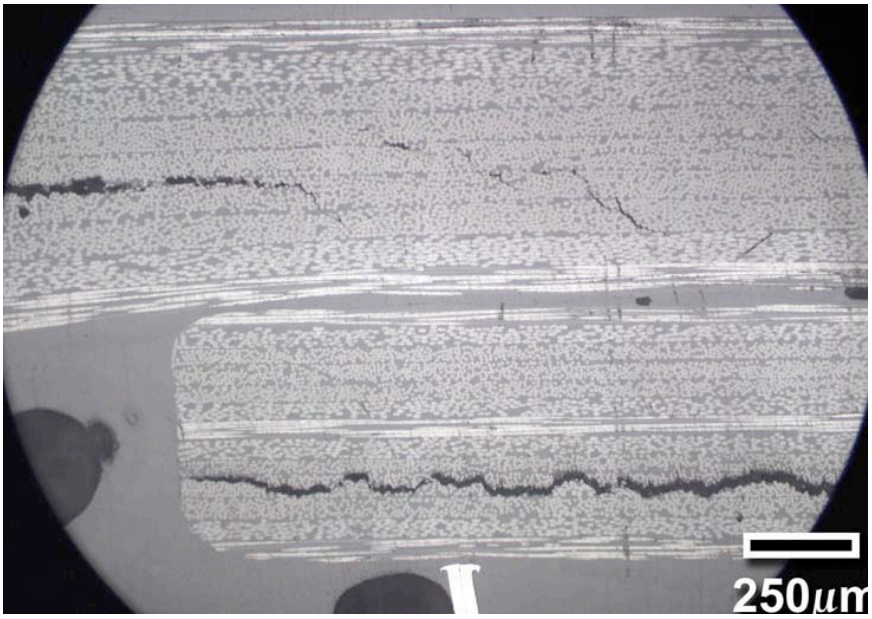

Figure 12. Cross-section of pre-cured doubler and co-cured composite laminate from an early test panel fabricated at JHU/APL. Inter-ply porosity is evident in the area not above the doubler, while there is no porosity in the laminate above the doubler. Cracking in the doubler is also evident.

The voids observed in the early laminates prompted discussions with the pre-preg vendor, YLA. They suggested adding a four-hour dwell at 88°C to the cure cycle in place of the two-hour 110°C dwell originally used for the purpose of removing any water vapor from the laminate and bagging materials (in order to prevent carbamate formation). The new approach would have the same effect of removing water vapor, while allowing the resin viscosity to increase prior to ramping the temperature up to the 177°C cure temperature, with the goal of preventing void formation. A panel with no doublers, triplers, or insert was made using that approach, and a sample was cut from that panel for microscopic evaluation. As can be seen in Figure 13, the panel had no porosity, verifying that the cure cycle revision solved the porosity problem.

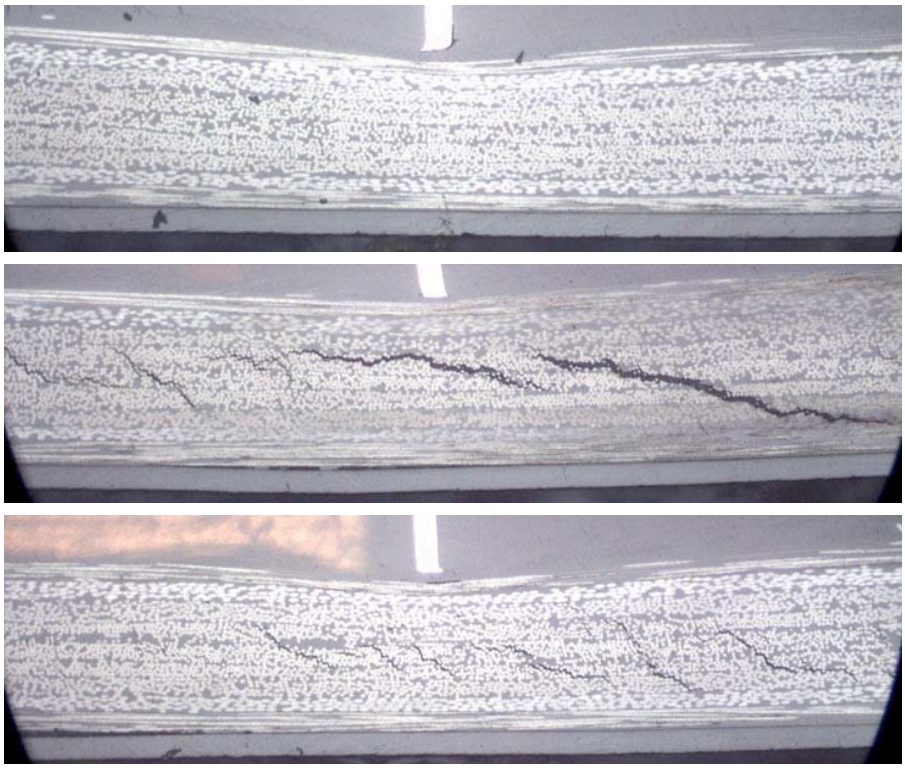

Figure 13. Cross-sections of development panel co-cured composite laminate fabricated with the optimized cure cycle, showing inter-ply porosity has been eliminated. Some micro-cracking remains.

Since the porosity evident in the earlier panels was eliminated through the modification of the cure cycle, the porosity was a consolidation problem, and not due to carbamate formation. Some micro-cracking still occurs, as well as VDA Kapton delamination in small areas. As noted above, the VDA Kapton delamination occurred in areas where there was no porosity and in panels with no porosity, so it is not related to the porosity.

Since the delamination occurs primarily beneath the VDA Kapton, not the plain Kapton, it is suspected that it is due to the impervious nature of the aluminum coating on the VDA Kapton and its resulting inability to allow outgassing of the facing laminate. The limited strength of the resin at the elevated thermal vacuum testing temperature of 270°C probably adds to the problem.

Extensive micro-cracking was observed in all laminates, including those fabricated with the modified cure cycle. This behavior is considered to be fairly common with co-cured pitch fiber laminates that have seen large thermal cycles. Also, pronounced intra-ply cracking is observed within one ply of the doublers, after they have been co-cured into a panel and post-cured. The exact cause is unknown, but it is likely to be the result of residual thermal stresses.

Similar microscopy was performed on test panels made along with each flight substrate (Figure 14). The micrographs were compared with those of the qualification test data and used as a means for acceptance of the panels.

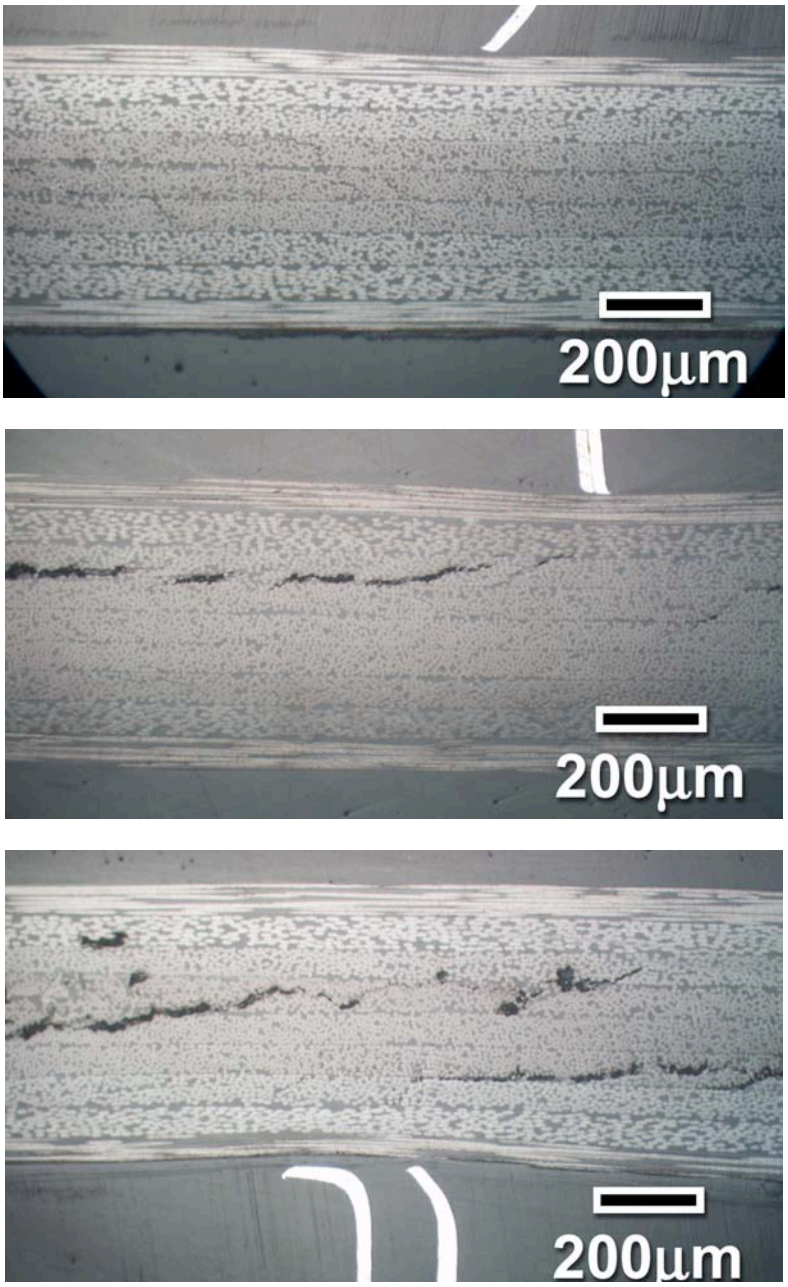

Figure 14. Cross-sections of flight substrate panel laminate.

Flight Panel Fabrication

Materials and process specifications incorporating all lessons learned were written and supplied to the vendors as part of the drawing package so that all processing details that were found to affect performance were incorporated into the flight panel fabrication process. Several scale-up problems, not encountered during the fabrication of the smaller test panels, occurred during the fabrication of the flight substrates.

For the flight panel, debulking film adhesive to the Kapton and VDA Kapton film required 16 hours since the pieces were much larger than those used for the qualification panels as opposed

to the six hours called for in the process specification based on experience from the development program, which used much smaller pieces of Kapton. The larger pieces used on the flight substrates required more time since trapped air had to travel to farther to reach the edge to escape. The vendor observed that after six hours under vacuum there was still a significant amount of porosity between the Kapton and the film adhesive, so it was left under vacuum overnight and ply cut the next day. In addition, the vendor found that areas of the film adhesive with buckled release paper need to be avoided, as the buckled paper does not allow the film adhesive to stick properly to the Kapton and VDA Kapton.

A serious problem occurred during the fabrication of one of the flight panels. Extensive cracking was observed in the doubler on the cell side of the second flight substrate panel. The crack extended along two sides of the center tripler. The crack occurred during an interim autoclave cure cycle used to bond accurately the solid inserts, triplers, and doubler to the honeycomb core and to each other.

The crack was attributed to thermal strain generated by the caul plates in the cure cycle. Graphite/epoxy composite caul plates were used against both sides of the core assembly to reduce the thermal strain in the laminate. The components were held in place with pins penetrating the caul plates. The caul plates were constructed from eight plies of intermediate modulus (T-300) carbon fiber/epoxy fabric pre-preg, totaling 0.300 cm (0.118 inch) in thickness. The doubler consists of 13 plies of K13C1U/RS-3 unidirectional tape pre-preg totaling 0.051 cm (0.020 inch) in thickness, in a nearly quasi-isotropic laminate configuration. While the CTE of the caul plate is significantly lower than that of aluminum or steel tooling, it is higher than that of the thin doubler laminate. The CTE of graphite/epoxy caul plate is 2.3 to 3.1 ppm/ºC, while the CTE of doubler is -0.49 ppm/ºC in the 0º direction and -0.65 ppm/ºC in the 90º direction. This mismatch, along with the brittle nature of pitch-fiber composite laminates, was undoubtedly the cause of the failure.

Laminate analysis software was used to predict the stresses in the doubler laminate during the cool-down phase of the cure cycle. While normally used to predict stresses in a laminate with an external load applied, a thermal load can also be used. Material properties for the caul plate and the doubler were used as input to the software, along with the orientation and thickness of each ply. The analysis assumed that there was intimate contact between the caul plate and the doubler, which would be a good approximation of the condition where the autoclave pressure induces a high level of friction between the two components. The results predicted compressive failure in the doubler laminate. Obviously, in order for the doubler to survive it must be allowed to slip relative to the caul plate as it cools.

The problem was solved in the fabrication of later core assemblies by venting the autoclave pressure and the vacuum on the part for the cool-down of the part from 149ºC. This resulted in reducing the friction between the parts, allowing them to slip relative to each other as they cooled and contracted at different rates, greatly reducing the compressive strain in the composite laminate.

Cracks also occurred in the facings of one flight panel during cool down from the post-cure temperature. While a composite caul plate and slip sheet with a CTE close to that of the facing was used for the initial cure cycle, a steel caul plate and slip sheet were used for the post-cure

cycle, since the composite caul plate and slip sheet would not withstand the post-cure temperature. The contraction of the steel during cooling caused compression failures in the facing laminate adjacent to the doubler. The panel was repaired and successfully static load tested but held as a flight spare. In successive panels, additional layers of release film were used between the panel and the slip sheet. The cooling rate was reduced from 2.8 to 1.7ºC/minute. The autoclave pressure and part vacuum were released during cool-down at 149ºC to allow the parts to slip relative to each other. The problem did not recur.

Extensive localized delamination was observed between the face sheet and doubler on one of the flight substrate qualification panels during thermal vacuum testing. The appearance of the failure surfaces indicated that the film adhesive had cured prior to application of pressure to the laminate. The cure cycle was again modified such that the full 276 kPa (40 psi) autoclave pressure was applied at the beginning of the cure cycle instead of after the four-hour 88ºC dwell in order to prevent the adhesive from curing prior to achieving good contact with the bond surfaces.

CONCLUSIONS

Conventional design and stress analysis techniques were successfully used to predict the macroscopic behavior of the solar array substrate**.** The micro-cracking and porosity observed in the test samples was not predicted and illustrate the importance that a strong testing program has on the successful application of composites in a critical new application.

Processing variations between vendors can have disastrous results for these types of materials where the materials are stressed to (and possibly beyond) their performance limits. Careful attention to processing details is required to achieve predicted materials properties. This is best conveyed through detailed material and process specifications that reflect the details of successful process development efforts.

Cyanate ester resin systems cured at 177°C and post-cured at 232°C can be effectively utilized for applications that require short-term exposures to 270°C, or possibly even 290°C, in a vacuum environment. Mechanical testing showed little degradation in properties after exposure to elevated temperatures, so it can be assumed that the original properties returned after cooling from those temperatures. Adjustments to the cure cycle such as the extended 88°C dwell used here effectively eliminate the porosity that typically results from co-curing a facing laminate over honeycomb core. Careful handling, cleaning, and drying of Kapton film materials, as well as extended debulking to the film adhesive and laminate, are required to eliminate porosity beneath the films and most delaminations during thermal vacuum exposure. Scale-up problems, not foreseen in the qualification phase, occurred during the fabrication of the flight substrates. These problems were solved, and substrate panels meeting all requirements were successfully fabricated (Figures 15 and 16).

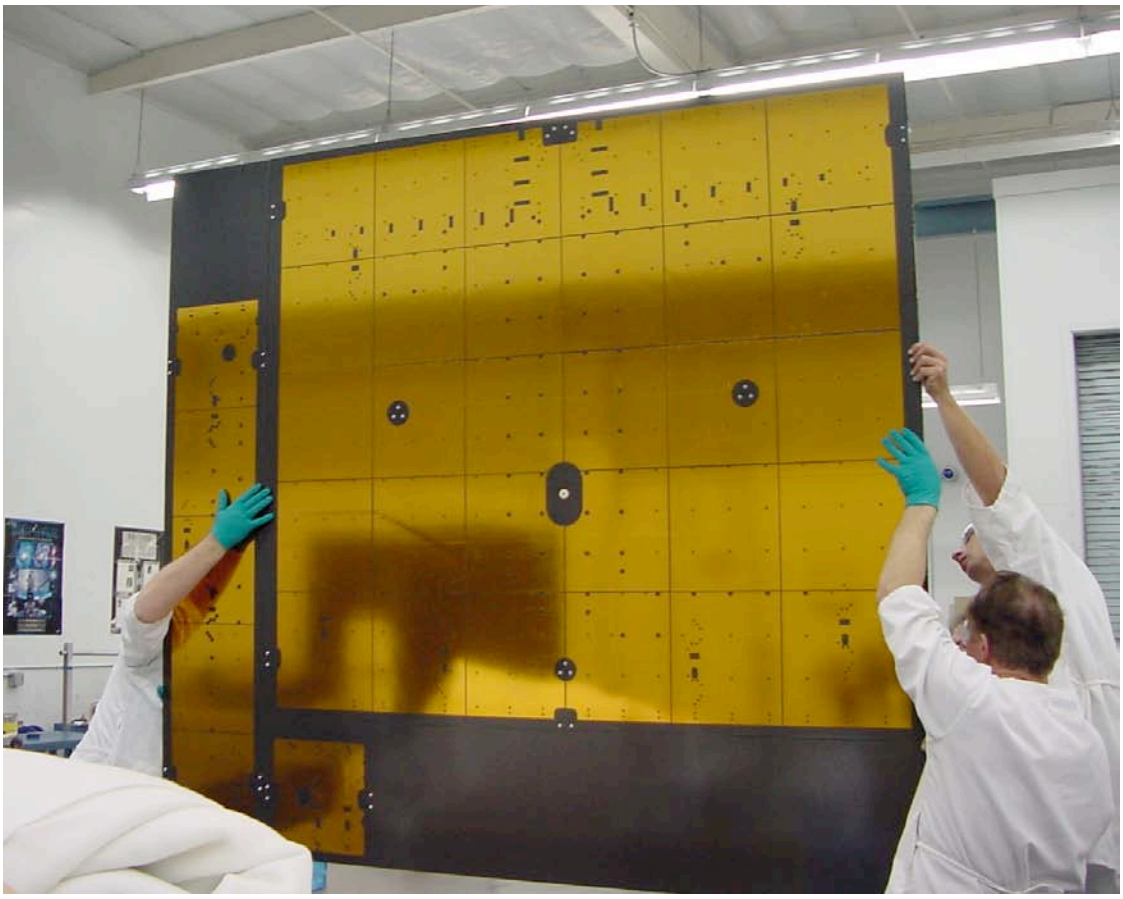

Figure 15. Photograph of as-cured flight substrate panel and qualification panels.

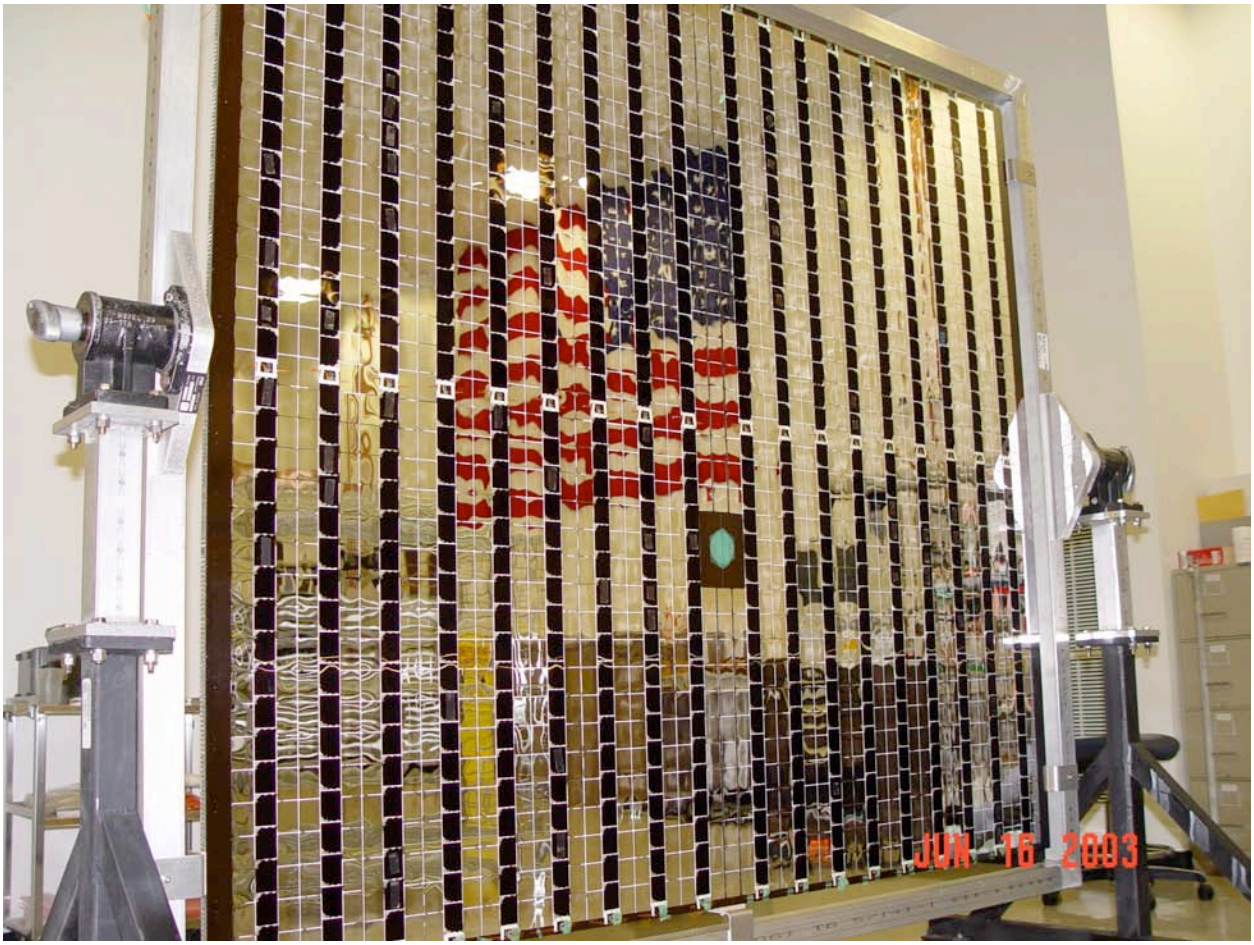

Figure 16. Photograph of populated flight substrate panel.

These findings are applicable to other high-temperature spacecraft applications of advanced pitch fiber cyanate ester composite structures, such as for other missions near the Sun or for those that require aero-braking into the atmospheres of Mars or Venus.

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