THE EFFECTS OF LONG-DURATION SPACE EXPOSURE ON THE MECHANICAL PROPERTIES OF SOME CARBON-REINFORCED RESIN MATRIX COMPOSITES

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INTRODUCTION

LDEF Experiment A0175 involved the non-instrumented exposure of seven carbon-fiber-reinforced resin-matrix advanced composite panels contained in two trays - A7 and A1. These two trays were located, respectively, on the leading and trailing faces of LDEF, obliquely oriented to the RAM (Row 9) and WAKE (Row 3) directions, as shown in Figure 1. This figure also shows:

(a) The identity and location of the seven panels, which consisted of six flat laminates of the following material systems: carbon/epoxy (T300/934), carbon/bismaleimide (T300/F178), and carbon/polyimide (C6000/LARC-160 and C6000/PMR-15), plus one bonded honeycomb sandwich panel (T300/934 facesheets and Nomex core) patterned after the Space Shuttle payload bay door construction. These material systems were selected to represent a range of then-available matrix resins which, by virtue of their differing polymer chemistry, could conceivably exhibit differing susceptibility to the low-earth orbit (LEO) environment.

(b) The principal exposure conditions of the LDEF environment at these tray locations. Noteworthy to some of the observations to be discussed herein is the four-orders-of-magnitude difference in the atomic oxygen (AO) fluence, [1] which made a shallow incidence angle (~22°) to Tray A7, while Tray A1 on the trailing face was essentially shielded from AO exposure.

This evaluation focused on determining the individual and relative suitability of a variety of resin-matrix composite systems for long-term space structural applications. This was accomplished primarily by measuring and comparing a range of engineering mechanical properties on over 300 test coupons sectioned from the flight panels and from identical control panels, and tested at ambient and elevated temperatures. This testing was supported by limited physical characterization, involving visual examination of flight panel surface features, measurements of weight loss and warpage, and examination for changes in internal integrity (microcracking, delamination) by ultrasonic c-scan and polished cross-sections.

RESULTS

Visual Observations

Detailed results of a survey performed by The Meteroid and Debris Special Investigation Group (M&D SIG) at Kennedy Space Center on the number and size of micrometeroid and debris impact sites have been published previously [4] and are summarized briefly here. For this experiment, the survey found that the number of impact sites, both above and below a 0.5 mm threshold size, was roughly twice as high on Tray A7 on the leading face of LDEF as on Tray A1 on the trailing face. Seven of the largest impact features, including the largest site (2.3 x 0.7 mm) located on the PMR-15 laminate, were protected against contamination after the survey, subsequently excised from the panels, and submitted for further evaluation (in progress) by the M&D SIG.

A second visual observation was of a fine white powdery residue over the surfaces of all three panels from Tray A7, except for a narrow band on each adjacent to the aluminum retainer strip along the "leading edge" of this tray. This residue was assumed to be a product of the reaction of AO with the surface layer of matrix resin; the narrow band was merely a region which had been shadowed from the full AO flux by the retainer strip. No loose fibers were detected on the superficially eroded surfaces.

A final visual observation involved the differing appearance of yellow epoxy ink markings which had been applied to identify both flight and control panels. These markings on the trailing-face panels were noticeably discolored to a brownish tint, whereas those on the leading-face panels as well as on laboratory control panels retained their original bright yellow color. Similar observations have been reported by other LDEF investigators and interpreted [5] as being due to competing effects of ultraviolet radiation and atomic oxygen on the epoxy ink. The discoloration is attributed to the effect of ultraviolet radiation which was of similar intensity on both trays; however, the atomic oxygen flux, which was four orders of magnitude more intense on Tray A7, resulted in a continual erosion of the UV- discolored surface of the ink, thus maintaining its near-original coloration.

Weight Loss Determination

Following disassembly of the panels from their trays and a drying cycle to remove any moisture absorbed during the disassembly period (7-12 days at 250F, identical to that performed prior to preflight weighing), the panels were reweighed. As shown in Table 1, the laminates from the leading-face tray lost 14-to-17 grams, equivalent to approximately 1%, whereas those from the trailing-face lost 0-to-3 grams (0-to-0.4%). [The sandwich panel lost 12 grams which, after subtracting the weight of the 76 metallic fasteners, corresponds to 0.5%.]

The difference in weight change between leading- and trailing-face laminates is attributable to the observed AO erosion of the former. If all of the weight loss is assumed to be due to AO erosion of matrix resin (realistically, some fraction may be due to outgassing of volatile constituents), this is equivalent to an erosion depth of 40 microns.

Warpage Measurements

It is commonly observed in composites that even balanced, symmetric laminates cured on a flat surface exhibit some small degree of nonflatness which is a manifestation of the state of residual stress within the laminate. It was originally thought that any changes in this physical characteristic might be relatable to the exposure conditions, such as thermal cycling, or to other potential changes in the laminates due to exposure such as, perhaps, microcracking or one-sided surface attrition. Consequently, the flatness of the exposed panels was measured before and after exposure.

The panels were placed on a surface table, weighted down along one edge with the exposed surfaces up, and the deflection measured along the opposite edge at the midpoint and both corners. In the preflight measurements, all of the BMI and PI panels were concave upward; in the postflight measurements, they were still concave upward, although generally to a much lesser degree. The carbon/epoxy laminate and sandwich panel were both flat, both before and after exposure. As shown in Table 2, the remaining laminates exhibited a marked reduction in warpage following exposure, with the single exception of the cocured* BMI laminate which exhibited, on average, a slight increase in warpage.

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*For a description of the terms "precured" and "cocured" , see the section on Mechanical Properties.

Evaluation of Laminate Quality

All flight panels were re-inspected following exposure by ultrasonic c-scan, which revealed no defects - such as delaminations or disbonds - which could be attributable to the exposure. Both the epoxy laminate and sandwich panel were completely free of any ultrasonic indications; the BMI laminates each exhibited overall sound quality with one or two small areas of porosity from the original as-cured condition. The PI laminates, however, exhibited substantial areas of porosity - both in the control and flight laminates - which must be attributed to non-optimized curing conditions for these materials. Accordingly, additional measurements of resin content, fiber volume, and void volume were performed which yielded results (Table 3) consistent with the c-scan observations, i.e., higher porosity levels in the PI laminates. More importantly, these measurements indicated lower fiber volume values than the customary 60% (nominal) level. These results raise a caution in comparing the mechanical properties results reported herein with those published elsewhere in the literature for these PI materials.

Cross-Sectional Examination of Panel Integrity

Small samples were sectioned from control and exposed panels and polished for examination of laminate integrity. A typical section through the epoxy sandwich panel is shown in Figure 2 which contains the exposed facesheet of the flight article and one or two Nomex core cell walls with adhesive fillets along the bondline. No differences were observed between pre- and postflight articles; there was no evidence of microcracking.

The postflight condition of the carbon/epoxy laminate is illustrated in Figure 3 which shows uniform microcracking through the thickness. In the single sample examined, the density of microcracks in the 8-ply stacks of unidirectional tape was identical on both sides of the laminate midplane (15 per inch) and the individual spacing varied from 0.04-to-0.11 inch. The original control laminate for this material was not available for examination; however, a newer laminate prepared by identical processing with the same layup sequence showed no evidence of microcracking in the as-cured condition.

Cross-sections of the BMI laminates in both pre- and postflight conditions are shown in Figure 4 (precured) and 5 (cocured). Both cure conditions exhibited similar levels of microcracking in the preflight condition and a notably higher density of cracks in the postflight condition. The results of measurement of the microcrack density at various layers through the thickness on a number of such specimens are summarized in Table 4. The results from specimen to specimen were quite consistent. It is noted that the midplane plies, which showed a low density relative to the surface plies in the preflight condition, exhibited a much higher density than the surface plies in the postflight condition, while the surface density also increased. This pattern held true for both cure conditions. No interpretation may be offered for this pattern of microcracking.

Cross-sections of the PI laminates in both pre- and postflight conditions are shown in Figure 6 (LARC 160) and 7 (PMR-15). As with the BMI samples, both materials exhibited similar microcrack densities in the preflight condition; however, there was no obvious increase in microcracking in the single postflight sample examined for each PI material (refer to Table 4).

During this examination it was also discovered that one 90° ply had been inadvertently omitted from the layup in the PMR-15 control, as noted in Figure 7; this fact comes into play in the interpretation of mechanical properties results discussed below. Also shown here in the postflight laminate is a localized area of porosity, which had been detected previously by ultrasonic inspection.

The development of, or increase in, microcracking in the flight panels is attributable to the thermal cycling experienced by all the panels during their exposure, which amounted to more than

30,000 cycles (orbits) over a maximum temperature range of 200 to -40°F. Also, an increase in microcrack density can provide an explanation for the reduction in warpage described above, since the cracking would tend to relieve cured-in residual stresses.

Mechanical Properties

All of the mechanical property test results are presented in Tables 5 and 6 and Figures 8 through 10. Each data point represents the high-average-low value of 5 replicas, unless otherwise noted in the table.

For the epoxy sandwich panel, flatwise tension and beam shear tests were used to evaluate the effect of exposure on, principally, the honeycomb core. In these tests, failure is expected to occur in the core (rather than in the adhesive bondline or the facesheet), and this was, in fact, observed for both the control and flight specimens. In general, wider scatter bands were observed for the flight specimens, although it should be noted that the control scatter bands were uncharacteristically small (a variation of $\pm$5% is not uncommon for these tests).

With regard to the flatwise tension test, there was essentially no difference noted in the room temperature results (less than 2% difference in average value), while the 350°F results showed a 17% lower value for the flight specimens. The beam shear test yielded just the reverse pattern of behavior, i.e., no difference at 350°F and a 6% lower value for the flight specimens at room temperature. Taken together, these tests are regarded as showing no unambiguous effect of the LDEF exposure on the honeycomb core strength.

The beam shear specimens were also instrumented with strain gages on the compression side, and the modulus values so determined are included in Table 5 . A 17% lower value was measured for the flight specimens at both temperatures. In itself, this result might be taken as evidence for some exposure-induced degradation in stiffness which could conceivably have been caused by thermal-cycling-induced breakdown of the fiber-matrix bond, for example. However, this possible interpretation must be tempered by other observed differences in modulus (discussed below) in which other flight specimens exhibited higher -- as well as lower -- moduli values.

Results of tension, compression, and rail shear testing are shown in Figure 9 for the BMI laminates which represented two different cure conditions. The "precured" laminate had been autoclave-cured at 350°F and 85 psi (per the prepreg manufacturer's specification) against a flat metal surface, whereas the "cocured" laminate was autoclave-cured against a layer of honeycomb core at a reduced pressure of 45 psi. This latter simulated a cure condition commonly employed in the production of sandwich structures, and it typically yields slightly reduced values for certain matrix-dependent mechanical properties due to the reduced consolidation pressure and the dimple-pattern which is transferred to the laminate from the honeycomb cell structure.

With regard to the 0°-tension and 90°-compression strength measurements, the differences between the mean strength values of control versus exposed materials are less than the individual scatter bands associated with these mean values. In some cases, the exposed material exhibits even a slightly higher strength value than the control, this despite the existence of a significantly higher density of microcracking in the former, as discussed previously.

With respect to moduli, the 0°-tensile moduli of all the exposed samples were curiously higher than those of the controls by anywhere from 1 to 22 percent, while the 90°-compressive moduli at 75°F were somewhat lower by 6 to 16 percent respectively for the precured and cocured materials. (Note: Compressive moduli were not measured at 450°F in the exposed samples due to the premature failure of elevated-temperature compression specimens of the control material at the point of contact with compressometer knife edges).

The lack of a consistent pattern of behavior in these property measurements precludes any inference as to the effect of microcracking, or any other exposure-related mechanism, on these properties.

The rail shear results are even more inconsistent and difficult to understand, inasmuch as the strength values measured for the exposed material are 40 to 60 percent higher than control values (measured in 1979), while the moduli values are comparable. Both sets of specimens - control and exposed - were examined to check for any apparent differences in failure mode, but all specimens exhibited valid and similar failures through the gage area. The comparable moduli values discount the possibility of incorrect chart-scale settings. The orientation of the specimens relative to the laminate 0° direction was confirmed to be the same (although in this regard, it is noted that rail shear strength should be insensitive to such an orientation mix-up). Similar standardized specimen preparation procedures, test fixtures, and test methods were used in both series of tests. There is simply no plausible explanation for this wide disparity in strength values.

With respect to the polyimide laminates, it is recalled that the fiber volume values for all the PI laminates were slightly-to-well below the generally targeted range of 60-65 percent (Table 3). For this reason, the mechanical properties measured for both PI materials in this study should not be considered to be representative of these material systems. Furthermore, the wide variation from laminate-to-laminate among the LARC-160 panels, in particular, makes any attempt to interpret differences in mechanical properties between control and exposed specimens rather meaningless. At least with the PMR-15 laminates, the physical properties and ultrasonic quality are not too dissimilar, so comparison of exposed and control values may be valid.

For the PMR-15 material, the mechanical properties of the control specimens were generally slightly higher than those of the exposed material by no more than 12 percent, which is not a large difference, especially considering the small sample populations (five replicates) being compared here. This trend is consistent with the somewhat better quality of the control laminate, as indicated by ultrasonic C-scans and the fiber volume measurements.

In a few cases, the control material exhibited lower values, namely 90° compression strength (7 percent) and modulus (38 percent) at 75°F and rail shear strength (6 percent) at 75°F. The former may be attributed in large part to the fact that the control laminate was missing one of the four 90° plies as noted previously, which would be expected to reduce its compression strength and modulus in this 90° direction relative to the exposed laminate. However, cursory analysis has indicated that the absence of one 90° ply should result in a slightly higher* rail shear strength (4 percent), rather than the measured 6 percent lower value, although again, this difference may not be significant.

It is further recalled that cross-sectional examination showed comparable levels of microcracking in both control and exposed PMR-15 samples and, moreover, that in the BMI laminates, higher microcrack densities after exposure were not reflected in any concomitant property reductions. In light of these considerations, the small differences between control and exposed PMR-15 samples noted here cannot reasonaby be tied to any exposure-related degradation, but rather to small-population data scatter and slight differences in as-cured laminate quality.

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*Due to a disproportionate reduction in cross-sectional area relative to load-carrying contribution of the missing 90° ply.

SUMMARY AND CONCLUSIONS

The principal effects of almost six years exposure in a low-earth-orbit environment on the condition of several carbon-reinforced resin-matrix composites were: (1) superficial erosion of the resin-rich surface by atomic oxygen; and (2) the development, or increase in density, of microcracks through the thickness of the laminates.

Atomic oxygen erosion was visibly apparent as a powdery white residue on the laminates exposed to an oblique incidence (approximately 22°) of atomic oxygen on the leading face of LDEF. It is believed to be responsible for a slightly greater weight loss among these laminates (1 percent) as compared to laminates on LDEF's trailing face (0 to 0.5 percent) where the atomic oxygen fluence was four orders of magnitude less. However, the erosion was confined to the resin-rich surface; there was no evidence of fiber loss or loosening and no indication that such erosion was sufficient to have a detectable influence on composite physical characteristics (specific gravity, thickness, resin content) or mechanical properties.

The development of microcracking in laminates which contained no microcracks in the as-cured state (carbon/epoxy), and the increase in microcrack density in laminates which did exhibit some cracking in the as-cured state (BMI), are attributed to thermal cycling (more than 30,000 cycles) over a temperature range of -40 to 200°F due to varying solar exposure. This microcracking is believed to be responsible, in large part, for the relief of cured-in residual stresses manifested by the reduced warpage measured in the flight articles as compared to their preflight condition. However, the microcracking did not appear to be of sufficient magnitude to have a measurable effect on the mechanical properties measured in this study.

In the BMI laminates, which contained some microcracking in the as-cured condition (for both the precured and cocured conditions), a notable increase in microcrack density did not produce any measurable effect on strength or stiffness properties. The modest differences in properties that were observed for these materials appeared to be of a random nature, either higher or lower in the exposed material relative to baseline control material.

In the PI laminates which showed somewhat wider variations in mechanical properties (than the BMI) but not necessarily an increase in microcrack density, the observed variations are more readily attributable to differences in original as-cured laminate quality rather than to any exposure effect.

The epoxy sandwich panel exhibited generally comparable mechanical properties between exposed and control, indicating no measurable degradation of bondline or honeycomb core strength due to the exposure. The lone exception to this is a small reduction in facesheet compressive modulus, but the small number of replicas, the inherent scatter in such measurement, and the lack of any independent evidence of a mechanism to explain such a difference make this observation rather inconclusive.

The primary conclusion, therefore, from LDEF Experiment A0175 is that the structural performance of a range of resin-matrix composites was not measurably affected by the almost six-year exposure in low-earth-orbit. The observation of some evidence of atomic oxygen erosion of the resin matrix in these materials, together with the knowledge that AO erosion was much more pronounced in similar materials located on the leading edge of LDEF, confirm the need for some sort of protection for such materials intended for long-life LEO missions. Likewise, the evidence of increased microcracking provides a mechanism for structural degradation in these materials which could become significant under certain types of loading or longer periods of exposure.

References

1. Bourassa, R. J. and Gillis , J. R., Atomic Oxygen Exposure of LDEF Experiment Trays; NASA Contractor Report No. 189627, May 1992.

2. Bourassa, R. J. and Gillis, J. R.; Solar Exposure of LDEF Experiment Trays; NASA Contractor Report No. 189554, Feb. 1992.

3. Berrios, W. M. and Sampair, T. R.; LDEF Post-Flight Thermal Analysis Calculated Flight Temperature Data Package - Preliminary.

4. See, T. et al; Meteroid and Debris Impact Features Documented on The Long Duration Exposure Facility, A Preliminary Report, Publication No. 84, JSC No. 24608, August 1990.

5. LDEF Spaceflight Environment Effects Newsletter, Vol. 1, No. 3, May 15, 1990.

Table 1

COMPARISON OF PREFLIGHT AND POSTFLIGHT PANEL WEIGHTS

Panel Description	Dry Weight (grams)		% Change
	Preflight	Postflight	
Tray A1 (Trailing Face)			
T300/934 sandwich panel	2642	2630	-0.45
T300/934 laminate	753.0	749.9	-0.41
C6000/LARC 160 laminate	579.5	579.1	-0.07
C6000/LARC 160 laminate	566.5	566.9	+0.07
Tray A7 (Leading Face)			
T300/F178 laminate			
. precured at 85 psi	1318	1303	-1.14
. cocured at 45 psi	1341	1324	-1.27
C6000/PMR-15 laminate	1490	1476	-0.94

Table 2

COMPARISON OF PREFLIGHT AND POSTFLIGHT WARPAGE

Panel Description	Deflection (inches)*	
	Preflight	Postflight
T300/F178 precured at 85 psi	0.339 0.270 0.221 0.277	0.000 0.012 0.000 0.004
T300/F178 cocured at 45 psi	0.243 0.100 0.155 0.166	0.209 0.175 0.185 0.190
C6000/PMR-15	0.204 0.270 0.221 0.232	0.107 0.083 0.130 0.107
C6000/LARC 160	0.445 0.350 0.315 0.370	0.149 0.189 0.194 0.177
C6000/LARC 160	0.530 0.600 0.715 0.615	0.008 0.007 0.040 0.018

*Three values listed correspond to corner-midpoint-corner of freestanding edge of laminate with opposite edge held down against surface table.

Table 3

LAMINATE PHYSICAL PROPERTY MEASUREMENTS

LAMINATE	THICKNESS (INCH) MIN.	MAX.	SPECIFIC GRAVITY	RESIN CONTENT(%)	FIBER VOL. (%)	VOID[1] VOL. (%)
BMI						
Precured Control	.088	.097	1.59	31.3	62.3	-2.6
Precured Exposed	.089	.095	1.59	31.4	61.8	-2.8
Cocured Control	.088	.097	1.56	33.7	59.1	-1.7
Cocured Exposed	.093	.098	1.57	34.0	59.3	-2.4
LARC-160						
Control (LD 3-1)	.090	.108	1.38	42.5	44.8	10.3
Control (LD 3-4)	.080	.087	1.54	37.4	54.3	1.5
Control (LD 3-6)	.078	.090	1.56	35.7	56.6	0.6
Exposed (LD 3-3)	.074	.092	1.55	37.6	54.5	0.8
Exposed (LD 3-5)	.080	.095	1.53	41.0	51.0	0.9
PMR-15						
Control	.100	.106	1.55	44.0	49.0	-1.5
Exposed	.105	.114	1.52	45.1	47.2	0.0

[1] "Void Volume" is a calculated value which typically yields a negative value (physically meaningless) for good quality laminates. Positive values are an indication of abnormally high porosity content.

Table 4

SUMMARY OF MICROCRACKING EXAMINATION

	SAMPLE I.D.	No. of Microcracks per Inch in Indicated Plies: 0_2	(± 45)	90_2	(∓ 45)	0_2	(∓ 45)	90_2	(± 45)	0_2
BMI CONTROL										
Precured	A1	10.7		-		4.0		-		14.7
	A2	11.1		-		3.7		-		9.3
	A3	9.3		-		3.7		-		13.0
Cocured	B1	9.5		-		3.2		-		17.2
	B2	-		2.0		-		6.0		-
	B3	10.9		-		4.3		-		17.4
	B4	10.9		-		4.3		-		17.4
	B5	-		2.3		-		0.0		-
	B6	-		2.3		-		1.1		-
Average		10.4		2.2		3.9		2.4		14.8
BMI EXPOSED										
Precured	C1	16.0		-		34.0		-		18.1
Cocured	D1	19.0		-		46.6		-		20.7
	D2	15.9		-		38.6		-		15.9
Average		17.0		-		39.7		-		18.2
POLYIMIDE										
LARC-160 Control	E1	19.4		-		(No 0° plies		-		14.6
LARC-160 Exposed	F1	17.6		-		at midplane)		-		16.5
PMR-15 Control	G1	6.0		-		(No 0° plies		-		7.1
PMR-15 Exposed	H1	10.6		-		at midplane)		-		8.5

Table 5

SUMMARY OF MECHANICAL TEST RESULTS FOR T300/934 EPOXY HONEYCOMB SANDWICH PANEL

PROPERTY	CONTROL RT	CONTROL 350F	EXPOSED RT	EXPOSED 350F
FWT STRENGTH (PSI)	343	271	338 (6)*	225 (6)
(as % of "Control")			98.8	83.1
SANDWICH BEAM CORE SHEAR TRANSVERSE STRENGTH (PSI)	85.9	68.0	80.7	67.2
(as % of "Control")			94.0	98.8
SANDWICH BEAM FACESHEET COMPRESSIVE MODULUS (MSI)	13.6	13.7	11.3	11.5
(as % of "Control")			83.3	83.7

*Five replicates per test unless indicated otherwise in ( ).

Table 6

MECHANICAL TEST RESULTS FOR BISMALEIMIDE AND POLYIMIDE LAMINATES

Property	C=Control E=Exposed	T300/F178 BMI				Polyimide			
		Precured		Cocured		LARC 160		PMR 15	
		75°F	450°F	75°F	450°F	75°F	550°F	75°F	550°F
0° Tensile Strength	C	66.5	71.7 (3)*	68.4	67.1 (2)	69.8 (6)	62.4 (6)	65.2	50.5 (6)
F_{tu} (ksi)	E	71.1	71.4	61.7	65.4 (6)	66.0 (6)	49.3 (6)	57.1 (6)	45.8 (6)
0° Tensile Modulus	C	7.9	7.8 (3)	7.6	8.2 (3)	7.1 (6)	[1]	6.0	[1]
E_t (msi)	E	8.2	9.5	7.7	9.1 (6)	6.7 (6)	[1]	5.4 (6)	[1]
90° Compressive Strength	C	48.7 (3)	40.9 (3)	47.1 (3)	38.3 (3)	60.4	48.2	50.2 (6)	37.2 (6)
F_{cu} (ksi)	E	47.4	42.0 (6)	46.2	40.3 (6)	52.9	31.7	53.5 (6)	34.3 (6)
90° Compressive Modulus	C	6.1 (3)	6.2 (3)	6.2 (3)	6.6 (3)	---	[1]	3.4 (6)	[1]
E_c (msi)	E	5.7	[1]	5.2	[1]	4.8	[1]	4.7 (6)	[1]
Rail Shear Strength	C	27.9 (2)	22.3 (1)	24.8 (2)	19.4 (1)	35.7 (4)	24.1 (2)	26.6 (2)	27.0 (2)
F_{su} (ksi)	E	38.7	31.9	35.5	31.0	34.2 (4)	29.9 (4)	28.3 (2)	25.9 (2)
Rail Shear Modulus	C	2.3 (2)	2.4 (1)	2.0 (2)	3.0 (1)	2.2 (4)	2.0 (2)	2.1 (2)	2.3 (2)
E_s (msi)	E	2.1	2.4	2.1	2.4	2.0 (4)	2.3 (4)	1.9 (2)	2.0 (2)

* Five replicates per test unless indicated otherwise in ( ).

[1] Compressive modulus was not measured on these specimens to avoid causing premature failure at compressometer knife edges (which was observed to occur in the previous testing of control specimens)

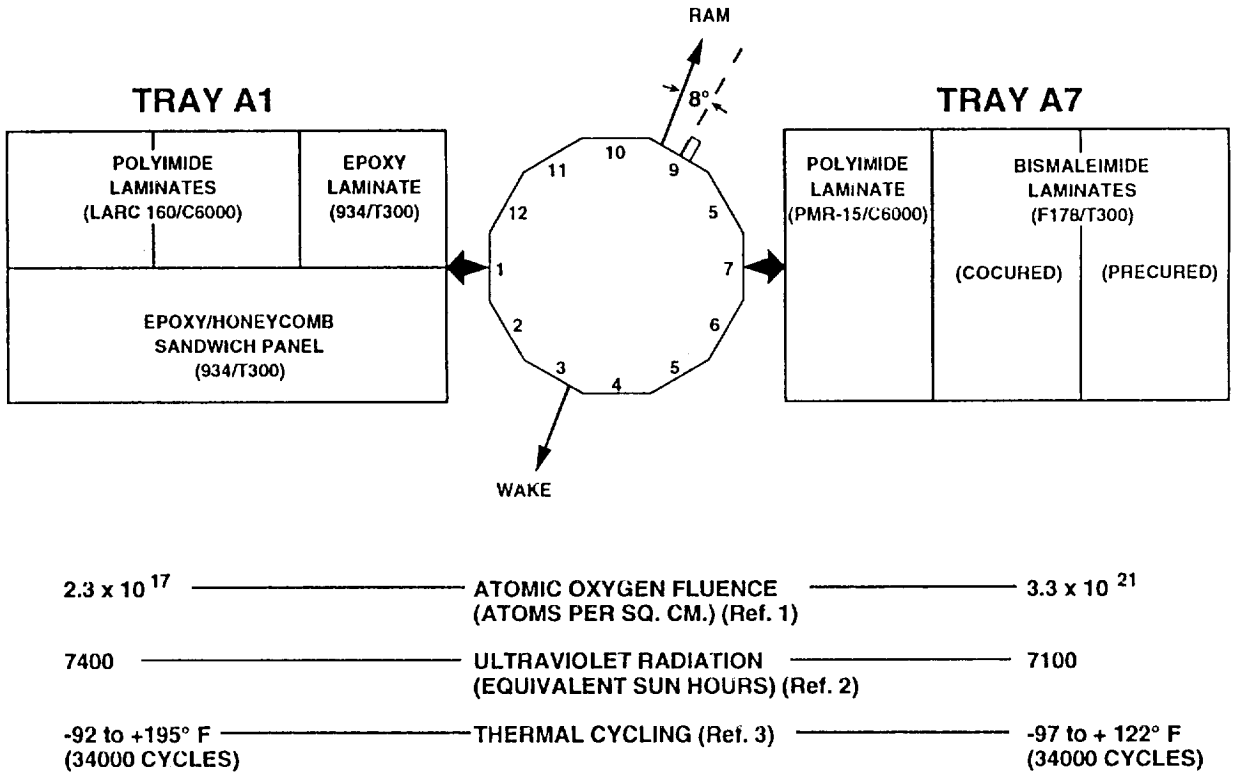

Figure 1. Composite Systems and Exposure Conditions

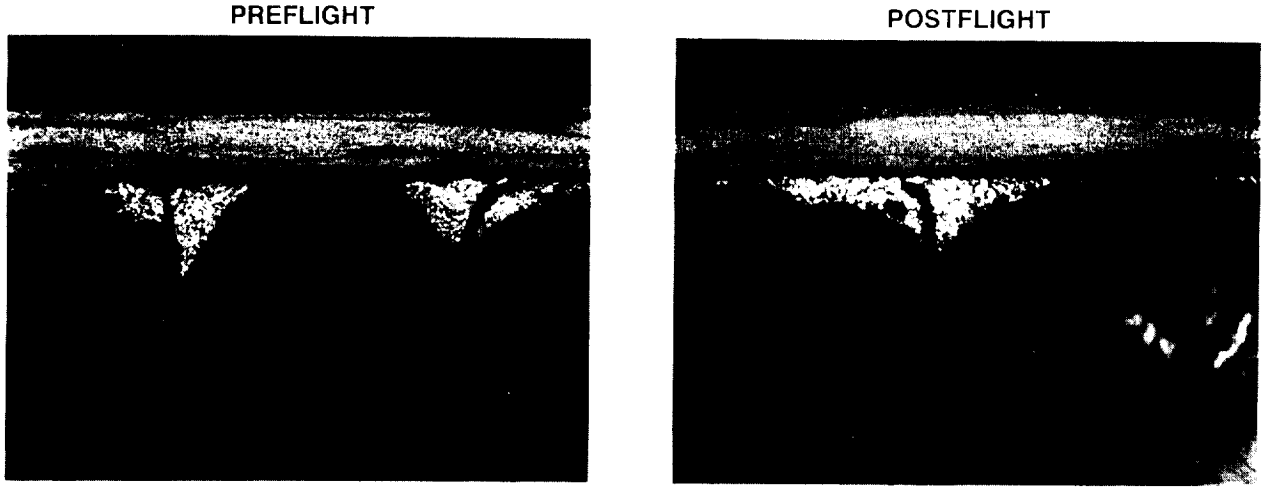

Figure 2. Polished Cross-Section of Honeycomb Sandwich Panel

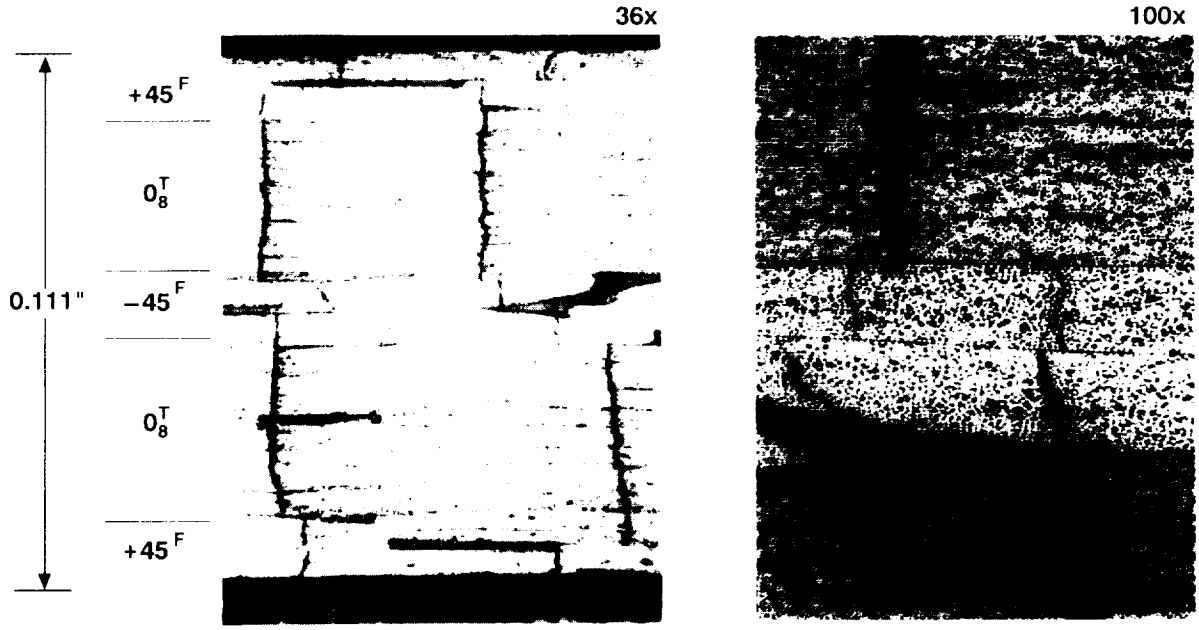

Figure 3. Polished Cross-Section of Postflight Epoxy Laminate, Showing Microcracking

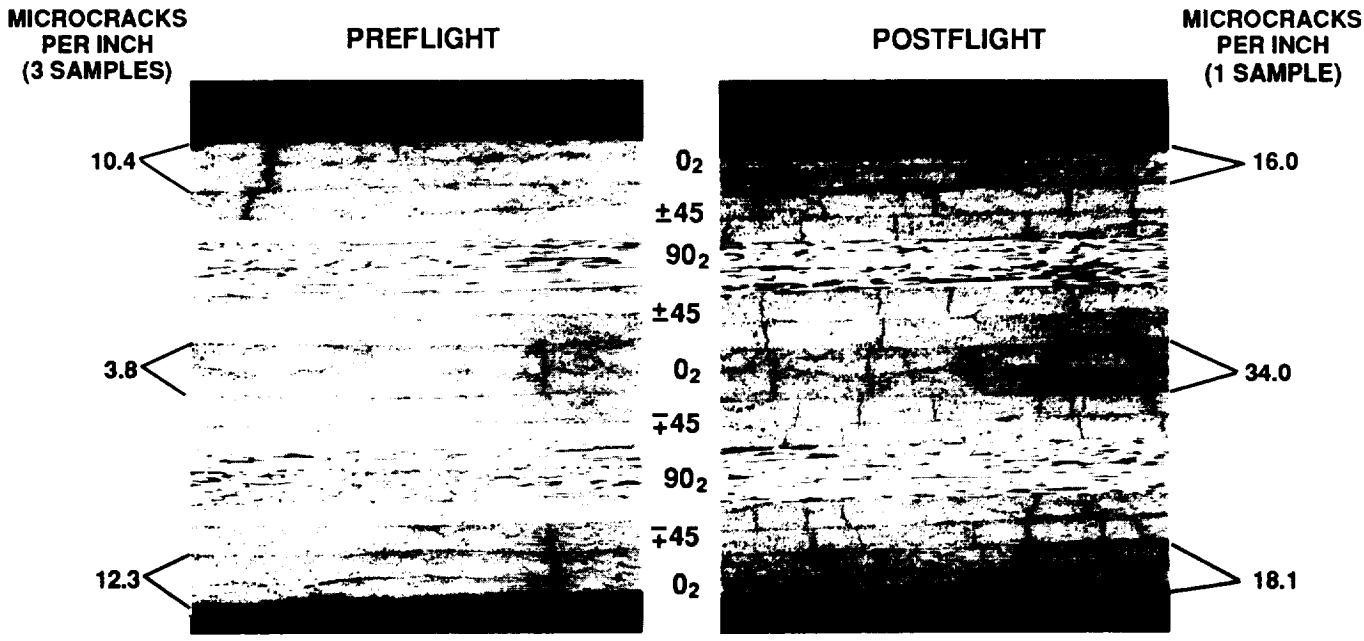

Figure 4. Polished Cross-Sections of Precured Bismaleimide Laminates, Showing Increase in Microcrack Density

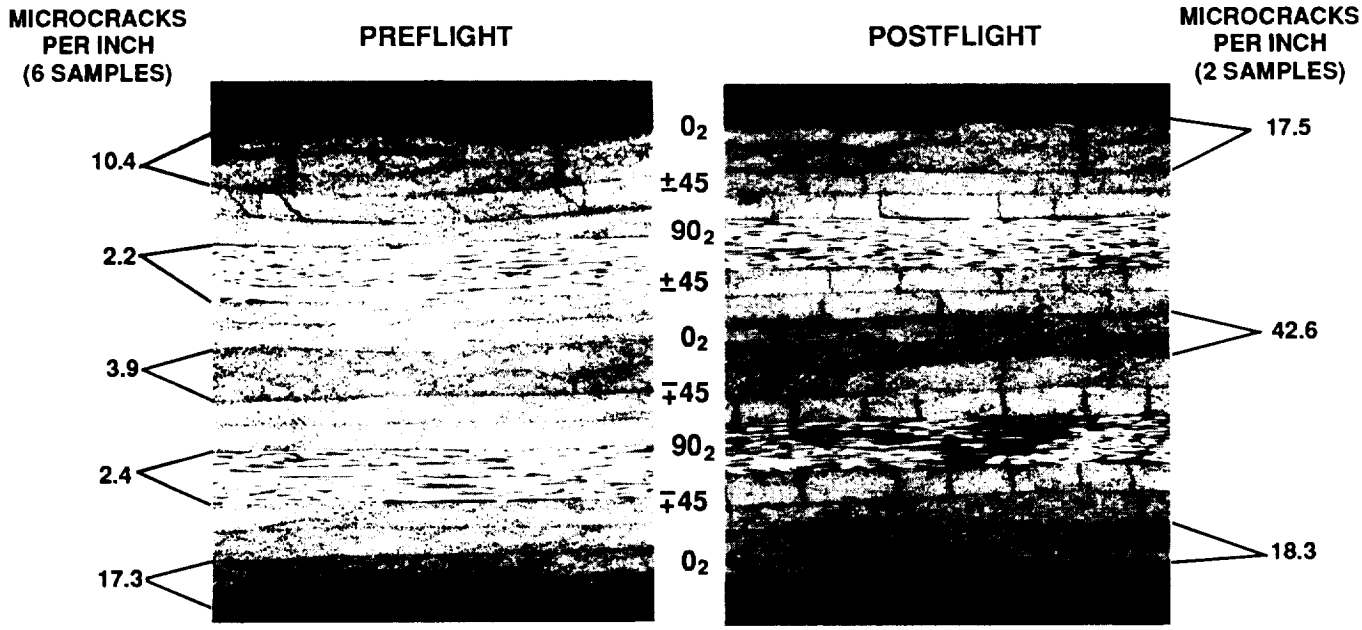

Figure 5. Polished Cross-Sections of Cocured Bismaleimide Laminates, Showing Increase in Microcrack Density

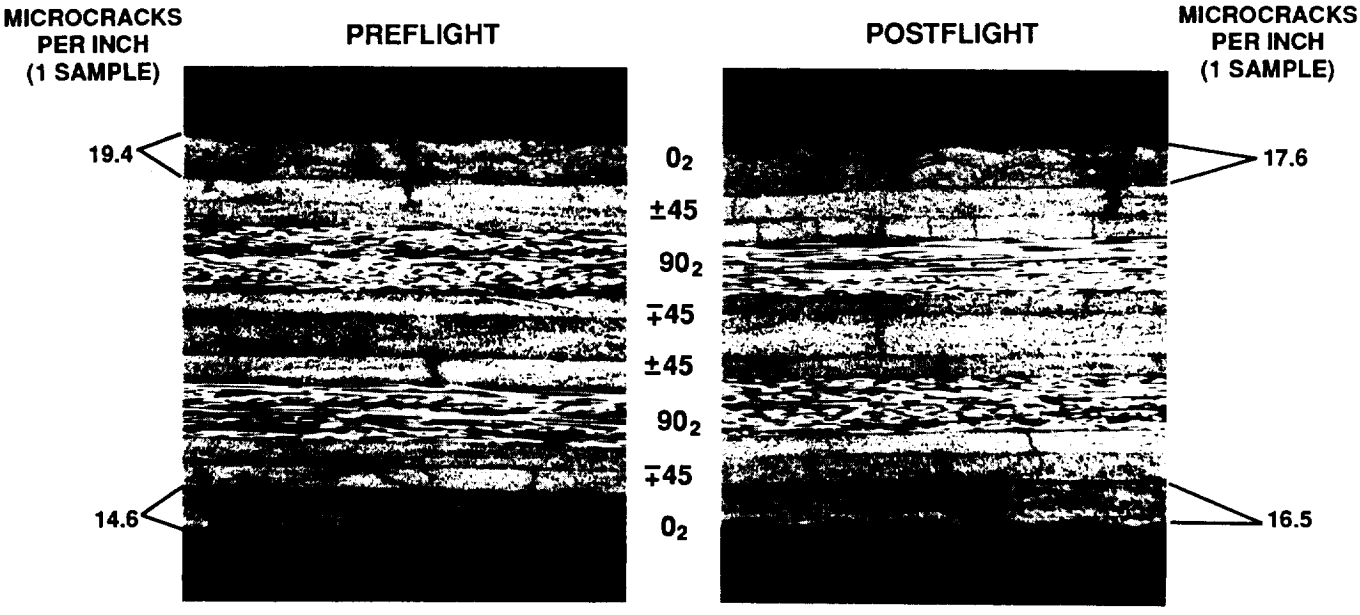

Figure 6. Polished Cross-Sections of LARC 160 Polyimide Laminates, Showing Comparable Microcrack Densities

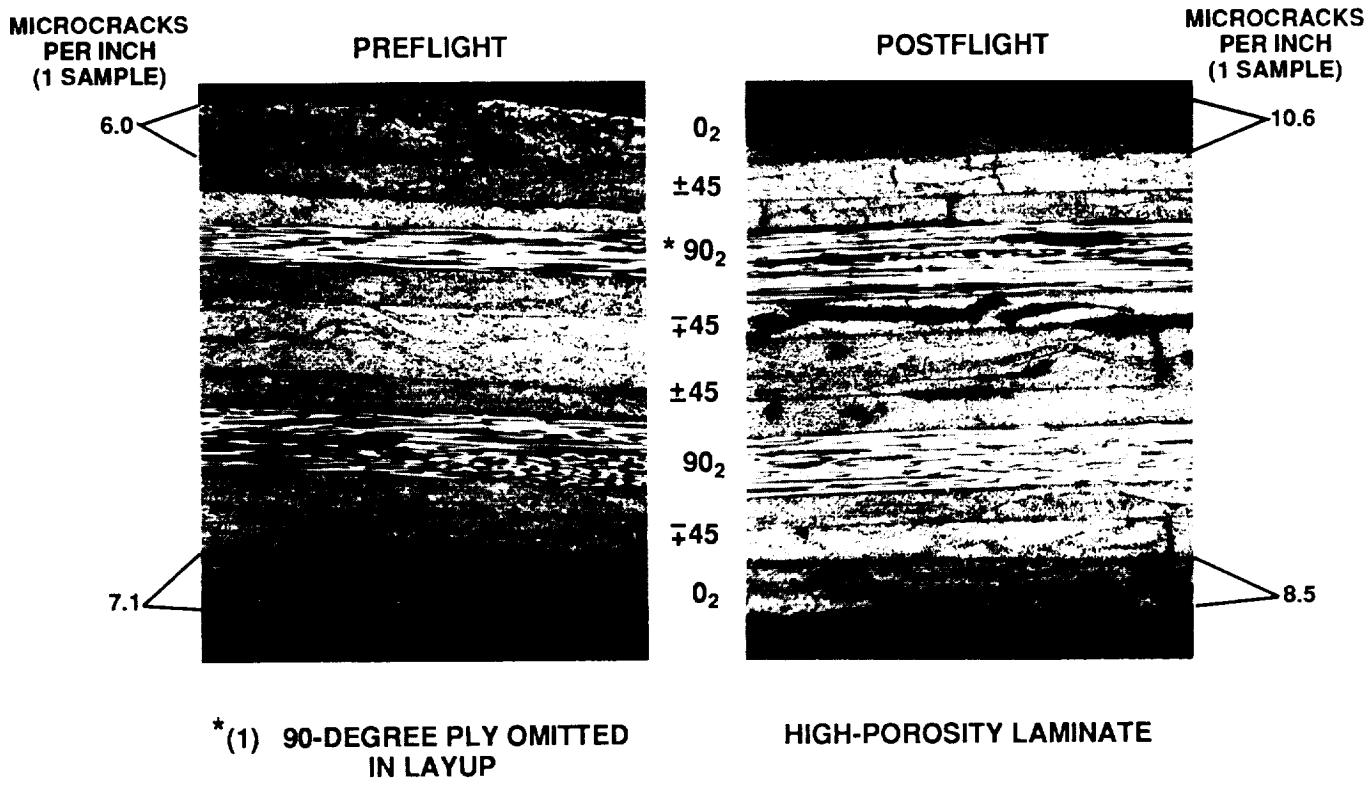

Figure 7. Polished Cross-Sections of PMR-15 Polyimide Laminates, Showing Microcrack Density, Missing 90° Ply (*), and Cured-In Porosity

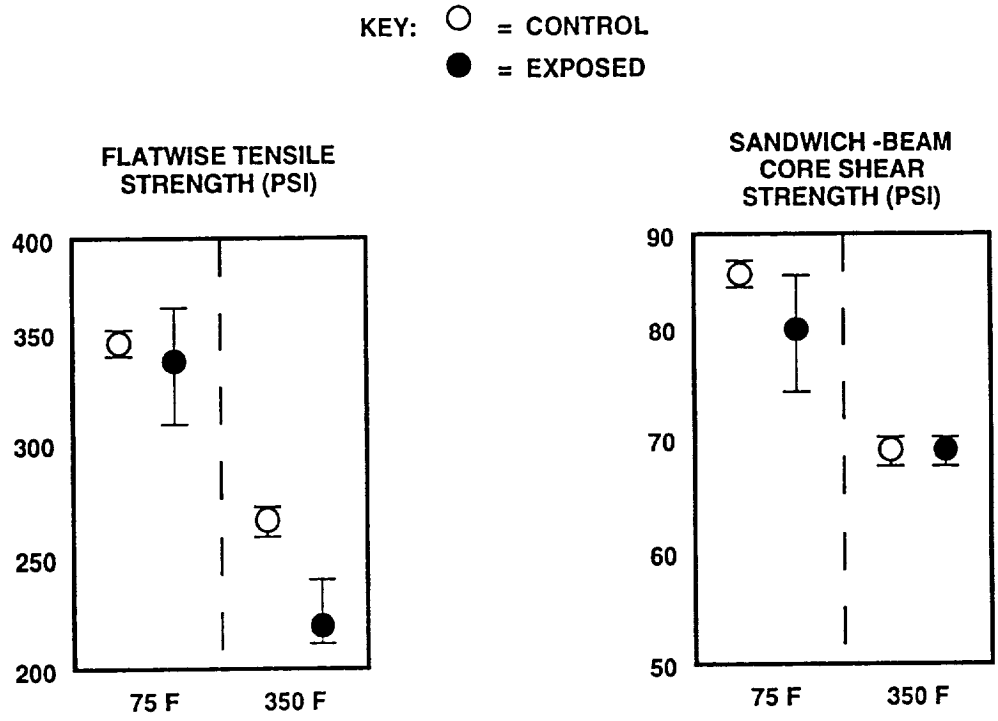

Figure 8. Mechanical Test Results for Epoxy Sandwich Panel

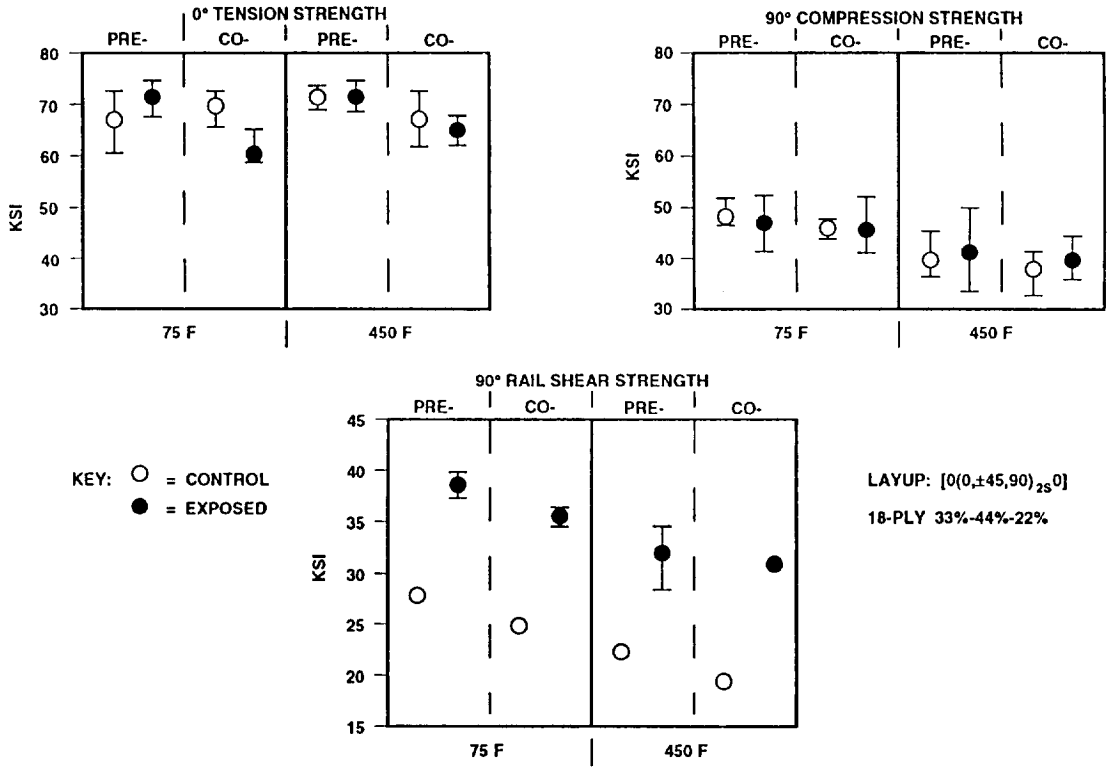

Figure 9. Mechanical Test Results for Precured and Cocured Bismaleimide Laminates

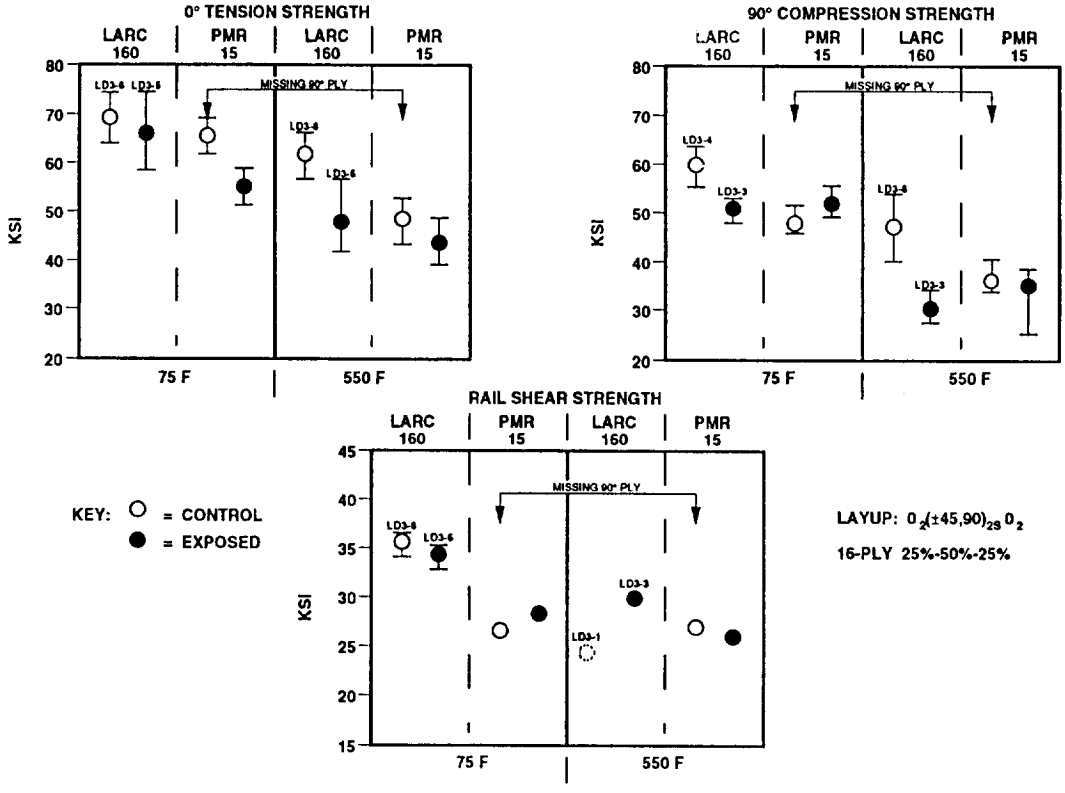

Figure 10. Mechanical Test Results for LARC 160 and PMR-15 Polyimide Laminates