Damage Progression and Residual Tension Strength Study of Fatigued Open-Hole Titanium-Graphite Hybrid Composite Laminates

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SUMMARY: Titanium graphite hybrid composite laminates are under consideration as a potential material for future supersonic aircraft. In order to be used on such vehicles, issues of damage tolerance and durability must be investigated. This study examines one lay-up with respect to damage progression and residual tension strength. The Open-Hole Tension (OHT) specimens are fatigued for one million cycles. The damage created in the laminates is monitored by the use of x-ray radiographs taken at various intervals. Finally, the residual tension strength is measured in a tensile test. The lay-up in this study demonstrates an average increase in residual strength of approximately 14% after significant damage accumulates in the titanium plies. Results from this study show that extensive cracks in the titanium plies do not reduce the structural integrity of the laminates when loaded in tension.

KEYWORDS: Hybrid Composite Laminate, Fatigue, Titanium, Damage Tolerance, Residual Strength

INTRODUCTION

This is an age where technology continues to push the limits of the aerospace industry. Airplanes are being built to go higher, farther, faster, and the competitive economic forces demand that they last longer. As these limits are continually expanded, new materials must be developed to meet the high demands. One such material that is under consideration for use in the High-Speed Civil Transport is Titanium-Graphite (TiGr) hybrid laminates.

In order to use this material on the aircraft, issues of durability and damage tolerance must be addressed. This program analyzes a specific lay-up that is under consideration. The lay-up is a 3Ti-6C-2C TiGr Laminate. It uses 0.0005" Ti-15-3 foils and 0.0054" PIXA-M/IM7 prepreg layers in the following configuration [Ti/90/0_3/Ti/0_3/90/Ti]. The three primary objectives of this program include:

1. Understand the development and progression of damage mechanisms accumulated during the fatigue of the 3Ti-6C-2C laminate.
2. Determine the changes in residual strength of 3Ti-6C-2C open-hole laminates subjected to fatigue with respect to the baseline specimens.
3. Develop understanding of stress distribution through the thickness of the laminate with respect to lamina stiffness of the constituent plies.

BACKGROUND

Before discussing the specifics of HTCL laminates, it is important to note the evolution of knowledge that led us from using solid metal as structural components to using

the hybrid laminates. In the mid 1960's Kaufman[1] showed that the fracture toughness of adhesively-laminated aluminum plies was improved in comparison with that of an equivalent monolithic plate due to the individual plies failing in a plane stress state. In the mid 1970's Johnson and colleagues[2] showed that laminated aluminum exhibited improved fatigue and crack-growth resistance. Later, Johnson[3] showed that the benefits shown by the laminated aluminum also applies to titanium laminates. He showed that the laminated titanium plies improved fracture toughness by almost 40%, increased fatigue life by an order of magnitude, and slowed down through-the-thickness crack growth rates by 20%. The history of laminated metals has shown definite advantages in fatigue and weight savings over traditional structural materials.

In 1995, Miller and colleagues[4] performed a preliminary evaluation of HTCLs. They concluded that the laminates have an improved fatigue life of almost two orders of magnitude over monolithic titanium. They also showed that the laminates remained whole and continued to carry the load after cracks had propagated through all of the titanium plies. Miller also used AGLPLY and showed that its predictions of stress-strain behavior are in excellent agreement with experimental data. Johnson, Li, and Miller[5] also performed an analytical examination of changing various factors in the HTCL. They found that the combination of fibers and fiber volume fraction must be such that the PMC has a higher modulus than that of the titanium in order to see significant advantages of HTCLs. They concluded that both Boron and intermediate modulus carbon fibers look attractive for HTCLs. In 1997, Li and others[6] evaluated different fabrication methods and found that an autoclave fabrication procedure produced more consistent material with more predictable mechanical properties.

Finally, Li and Johnson[7] studied the effects of fatigue on HTCLs. They looked at fatigue of HTCLs at both room temperature and elevated temperature. The most important findings in this study are that HTCLs have a longer fatigue life than monolithic titanium and that the fatigue life of the laminate is determined by the fatigue life of the titanium foils.

MATERIALS

Constituent Materials

The titanium alloy in the TiGr laminates is Ti-15V-3Cr-3Al-3Sn (Ti-15-3). Titanium foils are solution treated at a temperature above the α/β transition temperature, air cooled to room temperature, cold rolled to a thickness of 0.005 in and aged to produce the desired mechanical properties. This process creates a titanium foil with the mechanical properties listed in Table 1.

The PMC layers are composed of a thermoplastic polyamide matrix (PIXA-M) reinforced with IM7 fibers. The PMC layers have a volume fraction of $V_f = 0.62$ and a thickness of 0.0054 in. The Lamina mechanical properties are listed in Table 2.

Hybrid Laminate

The HTCL in this study is a 3Ti-6C-2C lay-up, or [Ti/90/0_3/Ti/0_3/90/Ti]. This configuration creates a laminate with 26% titanium, 56% 0° PMC, and 18% 90° PMC. The titanium foils are oriented such that the rolling direction is parallel to the 0° fiber direction. The titanium foils are cleaned and prepared with a Sol-Gel surface treatment prior to lay-up with the PMC layers. The laminate is then consolidated through an autoclave cure cycle. The effective laminate properties are calculated using Classical Laminate Plate Theory (CLPT) and presented in Table 3.

Specimens

The specimen geometry used for these tests are flat, rectangular open-hole tension (OHT) specimens. The dimensions are as follows:

Length	12 in
Width	2.0 in
Thickness	0.058 in
Diameter	0.25 in
Net Area	0.1015 in^2
Gross Area	0.116 in^2

In order to ensure that the specimens would not develop fatigue cracks in the grip section, tabs were placed on the ends giving a gripping length of 2 in. Figure 1 shows a photograph of the specimens with tabs.

Table 1. Mechanical properties of Titanium foils.

Properties	(Msi)
E_{11}	15.5
E_{22}	16.3
G_{12}	6.0
υ_{12}	0.33
σ_{UTS}	0.168

Table 2. Mechanical properties of PMC layer.

Properties	(Msi)
E_{11}	22.5
E_{22}	1.0
G_{12}	0.74
υ_{12}	0.35
σ_{UTS}	0.360

Table 3. Mechanical Properties of Laminate.

Properties	(Msi)
E_{11}	18.1
E_{22}	9.4
G_{12}	1.7
υ_{12}	0.18

Figure 1. Specimen Geometry.

TESTING PROCEDURE

The approach to this study is to fatigue the laminates for 1,000,000 cycles creating damage in the titanium layers at the region of the stress concentration. Then a static tensile test is performed on the specimens to determine the residual strength. Precrack fatigue is performed on the specimens in a SATEC servo-hydraulic 20 Kip test frame. The test frame is equipped with a lead cabinet and Faxitron x-ray unit, so that x-ray radiographs may be taken during the test with the specimens under load. The hold open load was 60 – 80 % of the maximum fatigue load. The x-ray exposure was set at 80 kV for 6.5 min and the image

was recorded on Polaroid Type 55 instant film. The specimens are gripped using 2 in wide screw action wedge grips. For the tensile test, the specimens are loaded in a 50 Kip screw driven test machine and a 1 in gage extensometer is used to record the strain values.

The precracking is conducting by constant amplitude fatigue test in accordance with ASTM standards. The tests are performed at an R ratio of 0.1 and a frequency of 10 Hz. In Li's study[8], he precracked the specimens to specific crack lengths. In this study, the cracks will be grown to a specific number of cycles at specific loads. For this study the cracks were all grown for 1,000,000 cycles at loads of 50 ksi, 40 ksi, and 30 ksi. Table 4 list the test matrix used in this study. These stress levels were chosen based on the results of Li's study. They are higher than the stress levels used in Li's study in order to give more damage and thus help to better define a "nuisance" crack. Figure 2 shows a photograph of the test setup. In this photograph, the lead box used in the x-rays and the stand that holds the film can be seen.

Table 4. TiGr nuisance program test matrix.

	No. of Specimens
Baseline	2
30 ksi	4
40 ksi	2
50 ksi	2
Total	10

Figure 2. Fatigue Test Setup

RESULTS AND DISCUSSION

Predictions and Analysis

The AGLPLY program performs an elastic-plastic analysis of symmetric composite laminated plates under in-plane mechanical loads. The lamina properties are calculated via the vanishing fiber diameter (VFD) model and Classical Laminated Plate Theory (CLPT). This program is limited to symmetric laminates such that the input values required only include material properties and volume fractions. AGLPLY does not account for stress concentrations, so predictions will be based on a solid composite with no center hole. AGLPLY was used in this study to predict the overall behavior of the laminate. By comparing the AGLPLY predictions to the experimental data provided by Boeing, the validity of the laminate constituent properties could be checked. Figure 3 shows the stress-strain graph of the AGLPLY predictions and the experimental results for an un-notched specimen. The predictions are in excellent agreement with the experimental results.

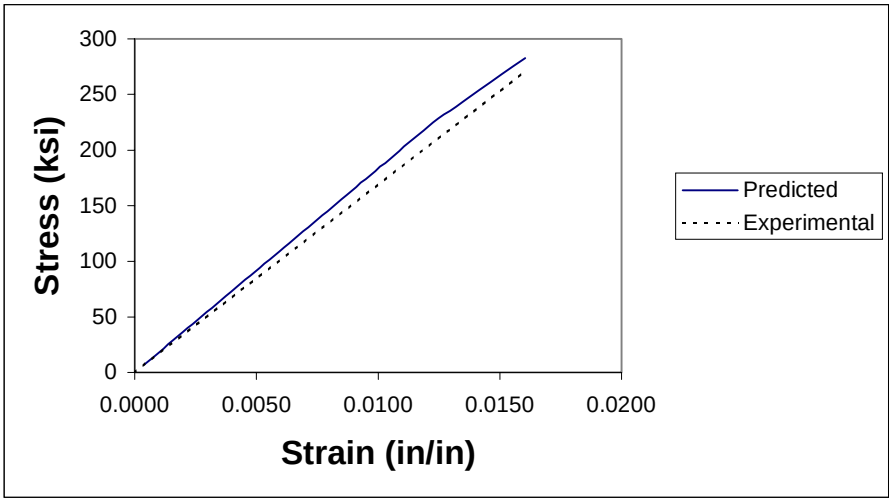

Figure 3. Stress-Strain Behavior of 3Ti-6C-2C Laminate.

In order to predict the behavior of the notched specimens, the stress concentration factor must be calculated. For solid metals, the stress concentration factor for an infinite plate with a circular hole is 3.0. For a solid plate with the same dimensions as these specimens, the stress concentration factor is 2.65. However, this does not work for composites. Lekhnitskii proposed a method for calculating stress concentrations in infinite orthotropic plates, which is given by equation 1. This method can be used along with a finite width correction factor to give an estimate of the stress concentration in the laminate, equations 2 and 3.

$$K_t^{\infty} = 1 + \sqrt{\frac{E_L}{G_{LT}} - 2\nu_{LT} + \sqrt{\frac{E_L}{E_T}}} \tag{1}$$

$$\alpha = \frac{2 + (1 - d/W)^3}{3(1 - d/W)} \tag{2}$$

$$K_t = \alpha K_t^{\infty} \tag{3}$$

Using the effective properties from Table 3, a stress concentration factor for the laminate is $K_t^{\infty} = 4.42$ with a finite width correction factor as $\alpha = 1.017$. Therefore, the estimated stress concentration factor for this laminate is $K_t = 4.50$. In order to determine when the titanium plies will crack, the stress concentration factor of the titanium foil must be known. By assuming that strain levels are equal in all plies, Li[8] wrote equations 4 and 5.

$$\sigma_{Lam} = \sigma_{Ti} \frac{E_{Lam}}{E_{Ti}} \tag{4}$$

$$K_{t,TiLayer} = K_{t,Lam} \frac{E_{Ti}}{E_{Lam}} \tag{5}$$

Therefore, $K_{t,\ Ti\ Layer}$ can be estimated as 3.85. In Naik and Johnson's[9] study they reported that the stress concentration factor reduced from as much as 3.65 to as low as 2.00 to 2.66 once the fiber and matrix debonded. In the present study, the stress concentration would also be expected to reduce significantly once fibers around the notch broke and the titanium plies cracked and delaminated. Although the analysis of stress concentrations is not the focus of this paper, it is an integral part in understanding the stress distribution through the thickness of the laminate. Therefore, as an addition to this study, a strain gage was added next to the notch of baseline specimen #2. Using the global and local values of strain, a K_t was calculated and Figure 4 shows a plot of how the K_t changed as the specimen was loaded to failure. In this plot, the K_t starts out as high as 4.5, but drastically reduces to about 1.80. Because the strain gage has some width and was not *exactly* on the edge of the hole, these values are slightly lower than the actual values. However, this gives us an indication of how the stress concentration factor reduces as damage around the notch increases. This behavior is outside the scope of this paper; however, more research into this phenomenon may be warranted.

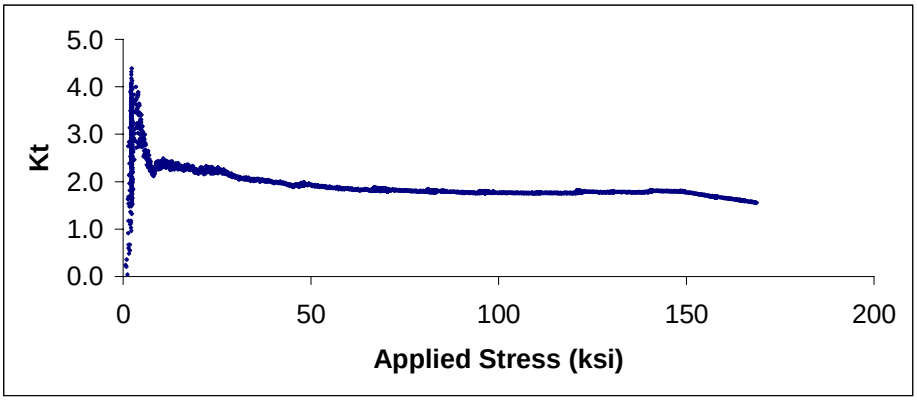

Figure 4. K_t Values as Recorded During Tension Test of Specimen #2.

Applying this factor of approximately 1.8 to the AGLPLY stress-strain curve gives some indication of when the laminate would fail. AGLYPLY predicts the un-notched specimen to fail at a stress of 283 ksi and a strain of 0.016 in/in. Dividing by a factor of 1.8 predicts the notched specimen to fail at 157 ksi and a strain of 0.009 in/in. In order to predict initiation in the laminates, the S-N data for the titanium foils can be used along with a stress concentration factor. If 3.85, the K_t in the titanium foil, is used, then initiation is grossly under predicted. The K_t that gives the best prediction is 2.65, K_t for a solid plate with circular hole and finite width. Using this K_t and the S-N data from reference 8, initiation is predicted to be 10,000, 50,000, and 900,000 cycles for 50 ksi, 40 ksi, and 30 ksi respectivly. These values are plotted in Figure 6. These values are highly dependent on the accuracy of the stress concentration factor.

The ability to predict the properties of the laminate is an important aspect in conducting test and evaluating the effectiveness of a material. This section is just a brief description of some possible methods. Table 5 summarizes the predictions and the experimental results of this analysis. In conclusion, it is noted that AGLPLY does an excellent job of predicting the behavior and properties of laminates, however, more analysis needs to be conducted on the stress concentration factors of HTCL.

Damage Progression

One of the main objectives of this study is to characterize and document the progression of damage accumulated during the fatigue of the 3Ti-6C-2C laminate. At all stress levels, the progression of damage followed the same pattern, only the amount of

damage changed with increasing fatigue loads. The first step in the damage progression was initiation. The tests were stopped periodically and examined for signs of crack initiation. Often it was hard to capture exactly when initiation occurred, but in general, when the crack reached 2 mm long, it was large enough to see and this point was called initiation. Figure 5 is an enlarged x-ray of a 50 ksi specimen after cracks have initiated and grown for a total of 50,000 cycles.

Table 5. Predicted vs Experimental

	Un-notched		Notched	
	Predicted	Experimental	Predicted	Experimental
Modulus (Msi)	18.1	16.2	18.1	21.9
Ultimate Strength (Ksi)	283	269	157	170
Strain to failure (in/in)	0.016	0.016	0.009	0.007

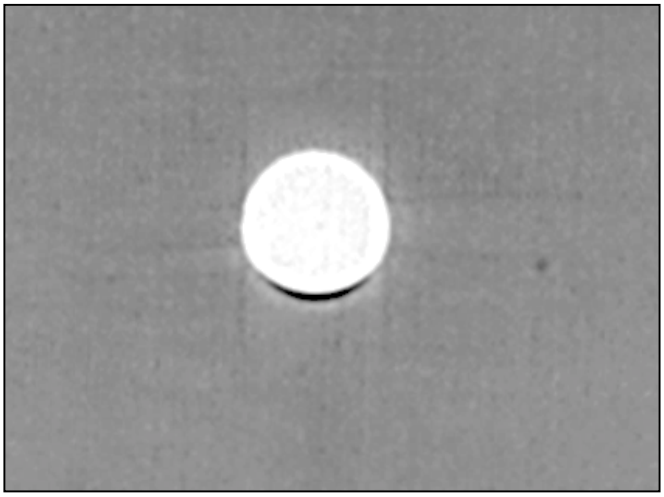

Figure 5. X-ray showing initiation and beginning crack growth.

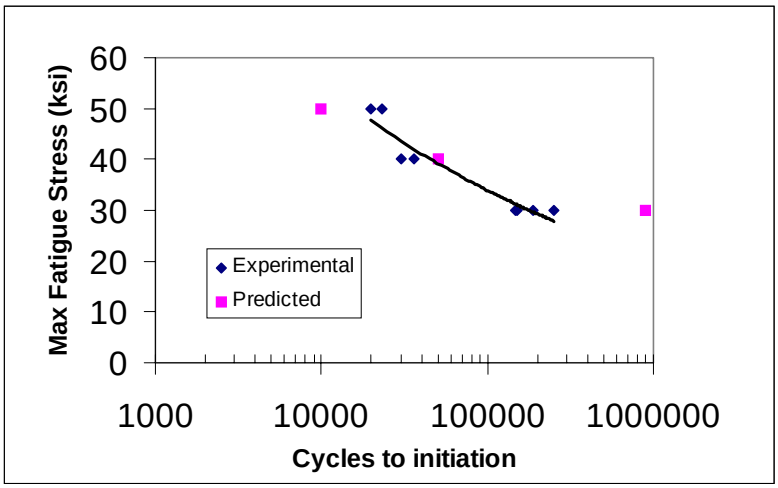

Figure 6. S-N curve for 3Ti-6C-2C laminate (initiation is defined as definite cracks in the outer Ti ply).

As seen in Figure 5, at initiation the fiber and matrix debonded at the edge of the notch causing the vertical line seen in the radiograph. The vertical crack was caused by the shear stress between the fiber and matrix at the notch root. Once the crack initiates, it grows normal to the loading direction. Note that multiple cracks initiate and grow from the hole. In

all cases, the cracks initiated on the outer Ti plies first, followed by the 90° PMC plies and the inner Ti ply. This is similar to the damage progression noted in Li's study[8]. Figure 6 shows a plot of the maximum fatigue stress versus the cycles to initiation in the titanium plies. This plot shows the normal S-N behavior where decreasing load increases cycles to initiation. After the cracks initiated, they grew across the laminate from the hole to the outer edge. Once the cracks reached the edges, delamination began from the outer edges and worked its way back towards the hole. Figure 7 shows x-rays of three specimens at an intermediate level of damage after approximately 650,000 cycles. Figure 8 shows x-rays of three specimens at their final accumulation of damage after 1,000,000 cycles. As expected, the level of damage becomes increasingly worse with higher load levels and more cycles. One observation that was unexpected was the fact the inner titanium ply never cracked completely across the laminate. In most cases, there was an obvious crack that initiated in the inner titanium ply, but it never grew to the edge. This is due to the fact that as the outer titanium plies cracked, the load was transferred to the 0° PMC plies. As those plies carried more load, the load in the middle ply decreased and the crack arrested.

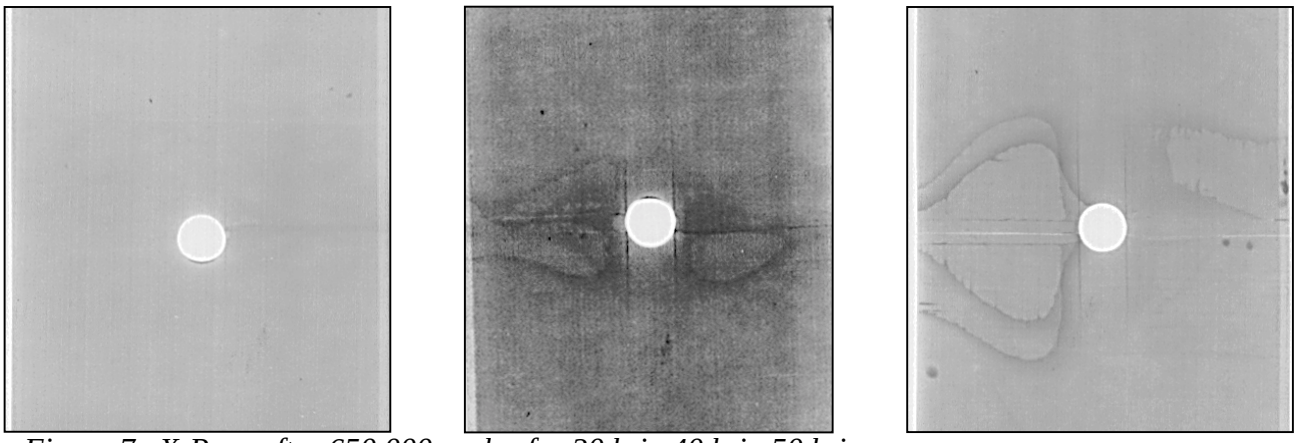

Figure 7. X-Rays after 650,000 cycles for 30 ksi , 40 ksi , 50 ksi.

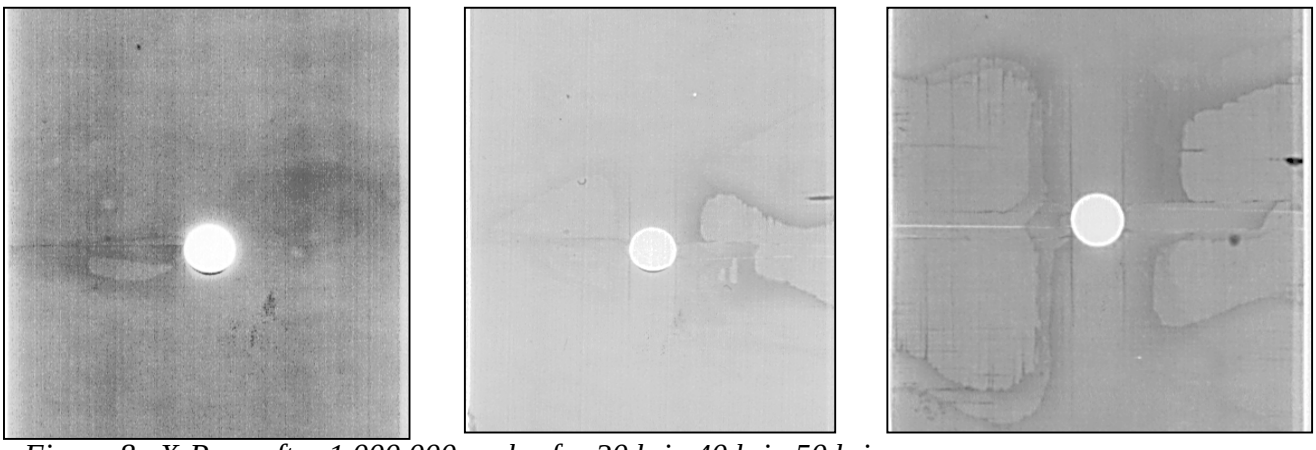

Figure 8. X-Rays after 1,000,000 cycles for 30 ksi , 40 ksi , 50 ksi.

Residual Strength

One of the main objectives of this study was to determine the residual strength of the laminate after it had undergone 1,000,000 fatigue cycles. First, two specimens that were both un-notched and not fatigued were tested for baseline properties[*]. This showed an average

[*] Boeing supplied the data for the two un-notched specimens.

residual strength of 269 ksi. Next, two specimens were tested that were notched, but not fatigued. This resulted in an average residual strength of 170.5 ksi. The reduction in ultimate strength from an un-notched specimen to a notched specimen is 36%. After fatiguing for 1,000,000 cycles at maximum fatigue stresses of 50 ksi, 40 ksi, and 30 ksi, the average residual strength was found to be 191 ksi, 204 ksi, and 189.25 ksi, respectively. This is an average increase of 14% above the notched, un-fatigued specimens. Figure 9 summarizes the residual strength data. The bars indicate the scatter in the experimental data. At first glance, this data seems counter intuitive. One would think that after cracking the strength of the laminate would be less. However, this is not necessarily true. In Li's study[8] , the residual strength either remained the same or reduced slightly in all cases. However, Li only partially cracked the laminates to crack lengths of 0.25 in. In this current study, the laminates were fully cracked in every case with varying degrees of delamination. At 50 ksi, the laminates were cracked completely across on each side of the hole in the titanium plies and there was major delamination between the titanium plies and the 0° PMC plies. At this level of damage, the majority of the load is being carried by the fibers and the multiple fatigue cracks have reduced the stress concentration. A simple calculation illustrates this fact. Knowing that the ultimate tensile strength of the 0° PMC layer is 360 ksi, the maximum load to fail the laminate if only the 0° plies carry load and there is no stress concentration is 20,400 lbs. This is comparable to the actual failure loads of the 50 ksi specimens of 20,080 and 18,661 lbs. Similarly, the failure loads of the 40 ksi specimens were 20,900 and 20,470 lbs. The residual strength of the 30 ksi specimens was lower, however, there was not as much damage, and thus a higher stress concentration factor. Naik and Johnson[9] also observed this behavior of reducing the stress concentration factor after cracking in a study they conducted on titanium MMC. Figure 10 shows the same data but with lines indicating the predictions of the residual strength. As the plot shows, the predictions are very close to the experimental data. This plot validates the results as the predictions follow the trends of the data.

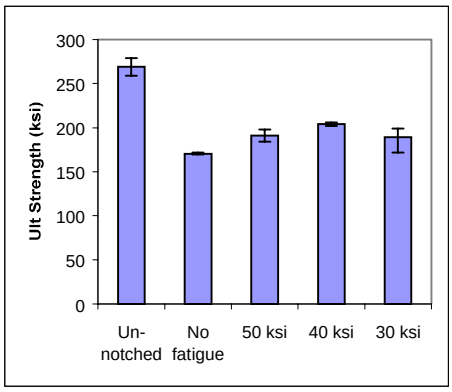

Figure 9. Residual Strength of HTCL (based on net section).

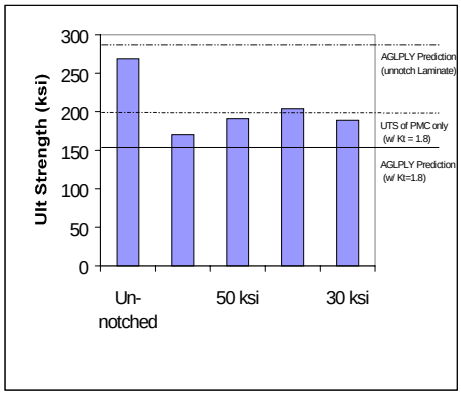

Figure 10. Residual Strength plot with lines indicating expected results.

Conclusion

The goal of this study was to examine the damage tolerance and durability of a 3Ti-6C-2C hybrid composite titanium laminate for possible use in advanced aircraft. In a similar study by Ed Li, he showed that shorter cracks (up to 0.25 in long) in the titanium plies result in almost no reduction in strength. These cracks are considered "nuisance" cracks and do not significantly affect the structural integrity of the material. In this study, cracks were grown for 1,000,000 cycles at loads of 50 ksi, 40 ksi, and 30 ksi. This resulted in larger cracks completely across the laminate (0.875 in long) in the outer titanium plies and significant delamination around the hole. However, when these laminates are loaded to failure, their

residual strength is actually greater. This is a phenomenon that results from the load transferring from the titanium plies to the 0° PMC layers and a reduction in the stress concentration factor. The results of this study show that this material has excellent durability and damage tolerance. Even with multiple cracks and severe delamination, the fibers are able to maintain all of the load. The level of damage seen in these specimens is greater than what is normally characterized as "nuisance" cracks. However, they do not affect the structural integrity of the material and therefore are nothing more than a "nuisance". In general, damage of this level would be repaired if for no other reason than aesthetics. The results of this study dictate that the strength that is important in design considerations is the ultimate strength of the notched laminate. The damage in the titanium whether it be small or large does not significantly affect the overall strength of the laminate, so long as the 0° PMC layers remain intact. This results in a material that is very damage tolerant. Future works should look at the effects of stress concentrations in laminates and examine methods of analytically determining the stress concentration factor. Also, mechanisms of load transfer from the titanium plies to the PMC plies should be investigated. This study only examined the strength of the laminate in tension and nuisance cracks could be much more detrimental if environmental factors or other modes of loading were considered.

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