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Washington, DC 20591

Characterization of BMS 8-212 for Use in Penetration Simulations

September 2009

Final Report

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16. Abstract

A program to simulate the performance of aircraft composite material as uncontained engine fragment barriers has been initiated and supported by the Federal Aviation Administration (FAA). Computational constitutive models of the shields are an important part of that program. An important initial step in the FAA program was the determination of the lamina properties of a representation carbon-fiber composite panel. BMS 8-212 lamina properties for use in the computational modeling of aircraft shielding systems were determined in this study. The carbon-reinforced BMS 8-212 composite lay-ups that were tested were unidirectional, 0.351″ thick (45-ply) flat panels. The mechanical responses of the flat panel unidirectional specimens were assumed to be representative of the behavior of the matrix-dominated compressive BMS 8-212 lamina responses in the transverse and normal directions. The rate sensitivity of the flat-panel specimens was determined for loading rates between 0.001 s-1 and 1000 s-1. In that regime, the transverse and normal failure stresses were found to increase by approximately 1.5.

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EXECUTIVE SUMMARY

A program to simulate the performance of aircraft composite material as uncontained engine fragment barriers has been initiated and supported by the Federal Aviation Administration (FAA). Computational constitutive models of the shields are an important part of that program. An important initial step in the FAA program was the determination of the lamina properties of a representation carbon-fiber composite panel. BMS 8-212 lamina properties for use in the computational modeling of aircraft shielding systems were determined in this study. The carbon-reinforced BMS 8-212 composite lay-ups that were tested were unidirectional, 0.351″ thick (45-ply) flat panels. The mechanical responses of the flat panel unidirectional specimens were assumed to be representative of the behavior of the matrix-dominated compressive BMS 8-212 lamina responses in the transverse and normal directions. The rate sensitivity of the flat-panel specimens was determined for loading rates between 0.001 s-1 and 1000 s-1. In that regime, the transverse and normal failure stresses were found to increase by approximately 1.5.

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Characterization of BMS 8-212 for use in penetration simulations

G.J.Kay, D.S. Urabe, A.L. Shields, S.J.DeTeresa

October 29, 2007

Auspices and Disclaimer Under LLNS

Auspices

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David Urabe
Albert Shields
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Abstract

BMS 8-212 lamina properties for use in the computational modeling of aircraft shielding systems were determined in this study. The carbon-reinforced BMS 8-212 composite lay-ups that were tested were unidirectional 0.351" thick (45 ply) flat panels. The mechanical responses of the flat panel unidirectional specimens were assumed to be representative of the behavior of the matrix dominated compressive BMS 8-212 lamina responses in the transverse and normal directions. The rate sensitivity of the flat panel specimens was determined for loading rates between 0.001 s^{-1} and 1000 s^{-1}. In that regime, the transverse and normal failure stresses were found to increase by approximately 1.5.

Introduction

A program to simulate the performance of aircraft composite ballistic shields has been initiated and supported by the Federal Aviation Administration (FAA). Computational constitutive models of the shields are an important part of that program. An important initial step in the FAA program was the determination of the lamina properties of a representation carbon-fiber composite panel. BMS 8-212, grade 190, type 3 class 1 (as classified by Boeing and made by Hexel) was chosen for this study. This material is representative of generic composite panels that could be employed in aircraft shielding systems. This report describes the determination of constitutive properties for a unidirectional 0.351" thick (45 ply) flat panel, and also gives results for a quasi-isotropic 0.361" thick (46 lamina) flat panel laminate that could be of use in later modeling efforts.

In the determination of the BMS 8-212 lamina properties it was assumed that the carbon fibers were essentially rate insensitive and brittle in their stress-strain responses. It was further assumed that the tensile and compressive strengths of the lamina in the longitudinal direction were available in the open literature. Furthermore the elastic properties for the lamina material in all directions were also assumed to be available in the open literature. The matrix dominated compressive properties of the BMS 8-212 were determined by tests employing cylindrical specimens of approximately 0.2745" diameter. These unidirectional lay-up specimens, cored out normal to the BMS 8-212 panels, were assumed to be representative of the matrix dominated compressive lamina BMS 8-212 responses. The rate sensitivity of the BMS 8-212 specimens were determined for loading rates between 0.001 s^{-1} and 1000 s^{-1}. A description of the tests and the results are given in Appendix A and a sample input for LSDYNA constitutive model "*MAT_COMPOSITE_FAILURE_MODEL" is given in Appendix B.

Test Results

Results of the tests for the unidirectional BMS 8-212 lay-up are shown in Figures 1. The results are consistent between the two testing regimes that were considered in the LLNL tests. Those two strain rate regimes were the 0.001 s^{-1} to 10 s^{-1} regime (performed on a Centorr universal test machine) and the 1000 s^{-1} strain rate regime (performed on a split Hopkinson Bar test machine). Each point on Figure 1 is the average five separate tests. A least squares fit to the unidirectional compressive test data is also given in Figure 1.

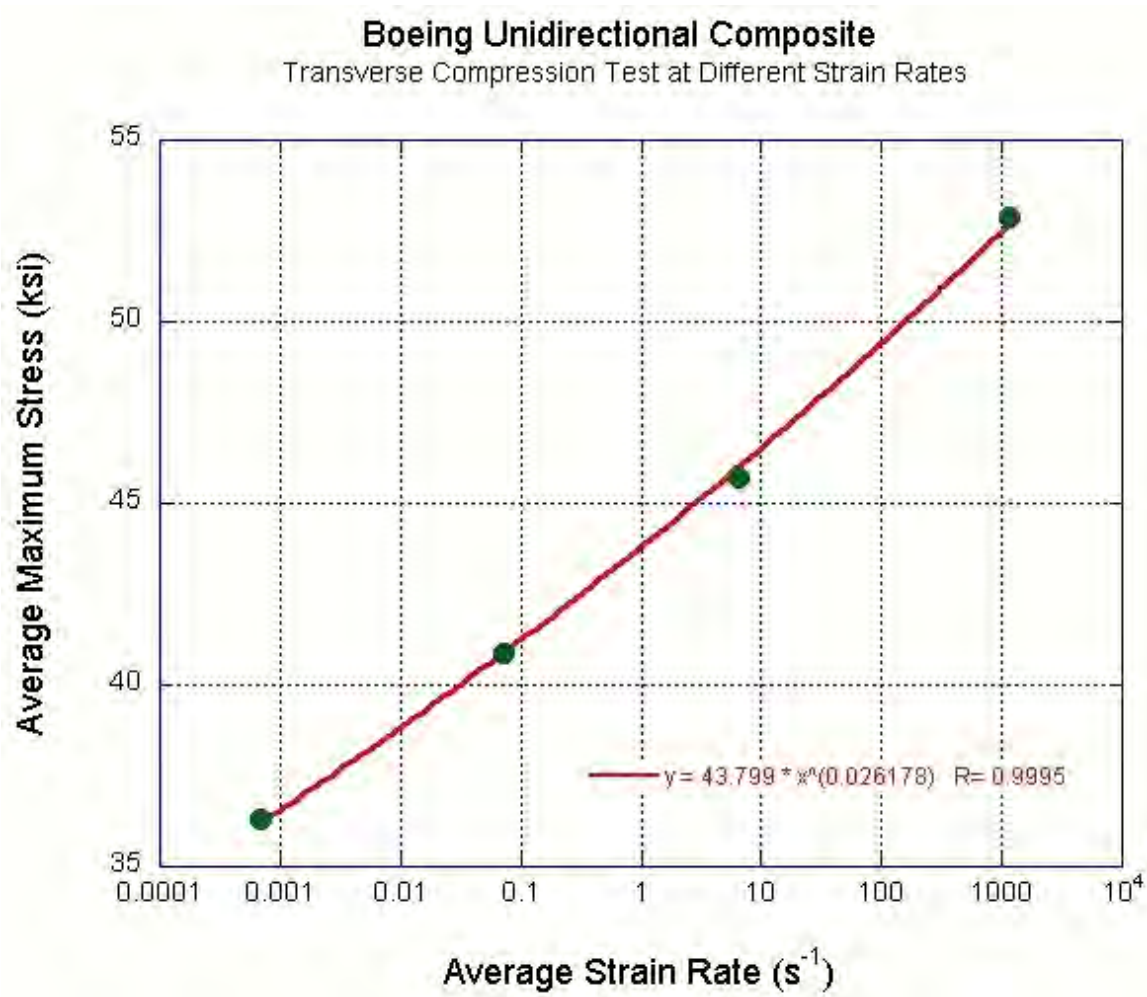

Figure 1. Unidirectional Lay-Up Results for the Matrix Dominated Compressive BMS 2-812 Transverse and Normal Responses

Results of the tests for the quasi-isotropic 0.361" thick (46 ply) flat panel laminate are summarized in Figure 2. Each point on Figure 2 is the average five separate tests. While these results were not used in the determination of BMS 8-212 lamina properties, they are presented here to provide information about possible future carbon-fiber

composite characterization tests. The quasi-isotropic results are not consistent between the .001 s^{-1} to 10 s^{-1} (Centorr universal test machine) and 1000 s^{-1} strain rate (split Hopkinson Bar) regimes. In fact the maximum stress drops considerably between the two regimes. A possible explanation for the quasi-isotropic material maximum stress drop off at the higher loading rate is that a new laminate failure mechanism is being activated, one that did not come into play for the higher loading rate unidirectional lay-up results

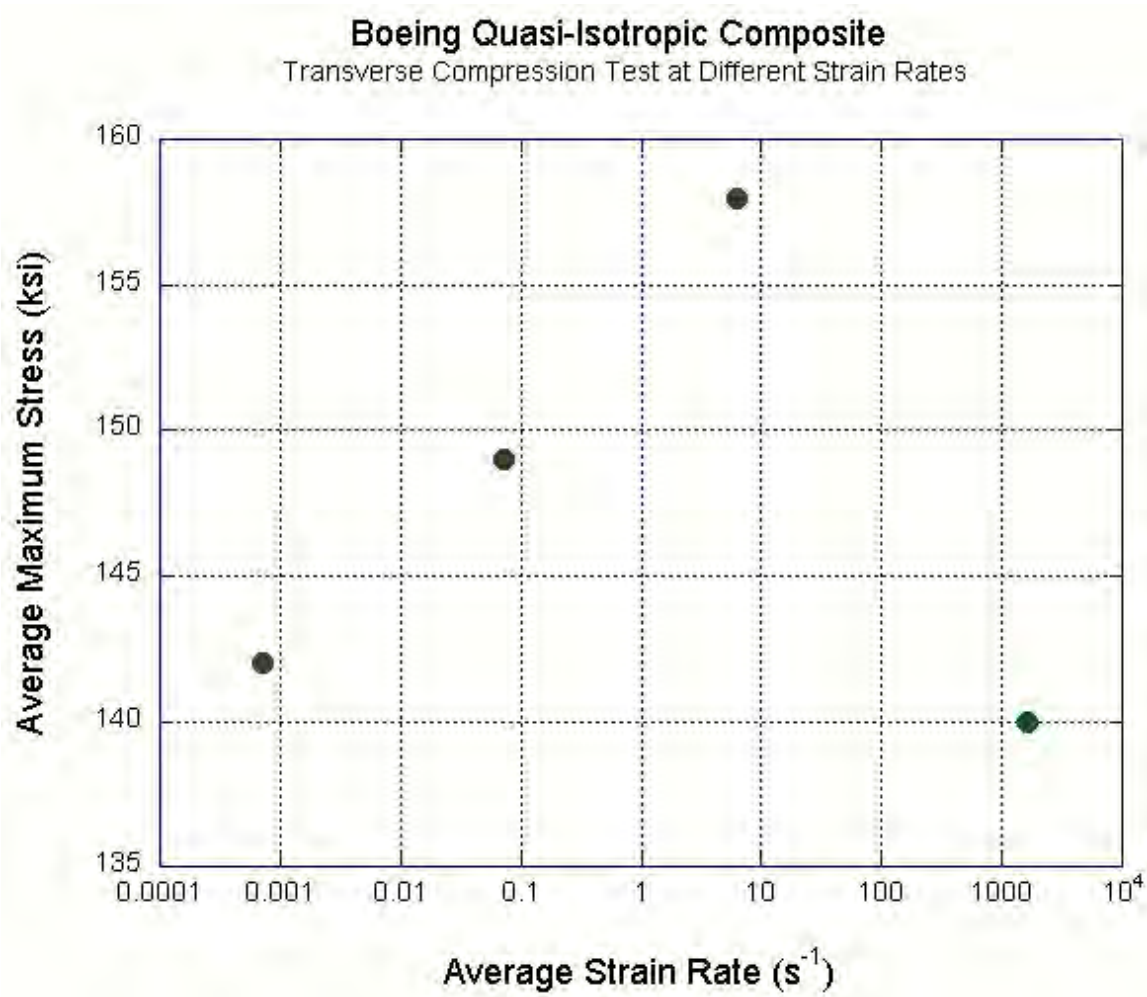

Figure 2. Quasi-Tranverse Lay-Up Results

Summary

The compressive transverse and normal failure stresses of the BMS 8-212 lamina material were found to increase according to the following expression for loading rates between 0.001 s^{-1} and 1000 s^{-1}:

$$Rf = 1.20535*(edot)^{0.026178}$$

where edot, the loading direction strain rate, could be represented in a constitutive model by the effective strain rate.

Appendix A: Testing description memo from David Urabe to Greg Kay

June 26, 2007

To: Greg Kay

From: David Urabe

Subject: Formal Report on ETR M0501053 - Boeing Composites

Test Request

The request was to formalize the documentation of the work done under ETR M0501053 – Boeing Composites. This report details the work performed by Albert Shields and David Urabe under the direction of Steve DeTeresa during late 2004 and early 2005.

Scope

The work consisted of several tests at different strain rates for Boeing composite material identified as 0º and 45º. The overall details of testing are given in Table 1: Scope of Testing.

Test Procedures

The lower strain rate tests, $.001s^{-1}$ through $10s^{-1}$, were performed using a sub-press assembly for compression with an extensometer to measure displacement on the Centorr universal test machine[1] (See Figure 1) and the higher strain rate tests, $>1000s^{-1}$, were done in the split Hopkinson Bar assembly (See Figure 2).

The lower strain rate specimens were measured and the dimensions recorded (See Table 2). Prior to test, the appropriate displacement rates were approximated by multiplying the desired strain rate by the specimen height to obtain an actuator speed for the lower strain rate tests. These values were used for test and compared with actual strain rate values after test. The force, displacement, and time were recorded during test. These and used to calculate stress, strain, and strain rate after test.

The higher strain rate specimens were measured and the dimensions recorded (See Table 3). The test parameters were calculated based on the estimated flow stress of the materials; 40 ksi for 0º and 150 ksi for 45º. The test uses a cylindrical projectile to impact a cylindrical incident bar which has been instrumented with strain gages. The

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[1] Centorr, calibrated on 11/5/2004, ASTM E4

impact sends a compressive stress wave down the incident bar, into the test sample and out through a transmitter bar with similar instrumentation. The signals from the two sets of instrumentation are compared and the net change is used to calculate the stress and strain caused by the stress wave. The strain signals from the two bars during test were captured with an Nicolet Integra system at a rate 5 MHz and evaluated using the KaleidaGraph spreadsheet software to produce graphical representations of stress and strain. These tests were performed at a projectile velocity which would bracket a strain rate of 1000 s^{-1} or greater using a 13 inch projectile in our 0.560 inch diameter high strength steel split Hopkinson Bar compression assembly.

Test Results

The lower strain rate test results are tabulated in Table 2: Lower Strain Rate Summary. The test series are compiled for the different materials by strain rate test in figures 3 through 8 with comparison graphs of the two materials by strain rate in figures 9 through 11.

The higher strain rate test results are tabulated in Table 3: Split Hopkinson Bar Summary. The test series are compiled by material, 0º and 45º, in figures 12 and 13 with a comparison graph of the two materials in figure 14.

Observations

There appears to be a small increase in the maximum stress as the strain rate increases for both materials up to the high strain rate of greater than 1000s^{-1} in the Hopkinson bar tests where the 0º material continues to increase, but the 45º begins to show a decrease. However, this may be part of the statistical variation within the composite properties at the higher strain rates. There were no other supporting references for the Boeing material properties.

The Hopkinson bar testing demonstrated the brittle nature of both materials. There was very little flow prior to fracture in all of the tests which showed in the large decrease in strain rate as shown in Table 3. The remnants of a 0º specimen and a 45º specimen after test are shown in Figures 15 and 16, respectively.

Material ID	Test Strain Rate (s^{-1})	Number of specimens	Specimen IDs
0º	0.001	5	1, 2, 3, 4, 5
0º	0.1	5	6, 7, 8, 9, 10
0º	10	5	11, 12, 13, 14, 15
0º	>1000	5	16, 17, 19, 20, 21
45º	0.001	5	1, 2, 3, 4, 5
45º	0.1	5	6, 7, 8, 9, 10
45º	10	5	11, 12, 13, 14, 15
45º	>1000	5	16, 17, 18, 19, 20

Table 1: Scope of Testing

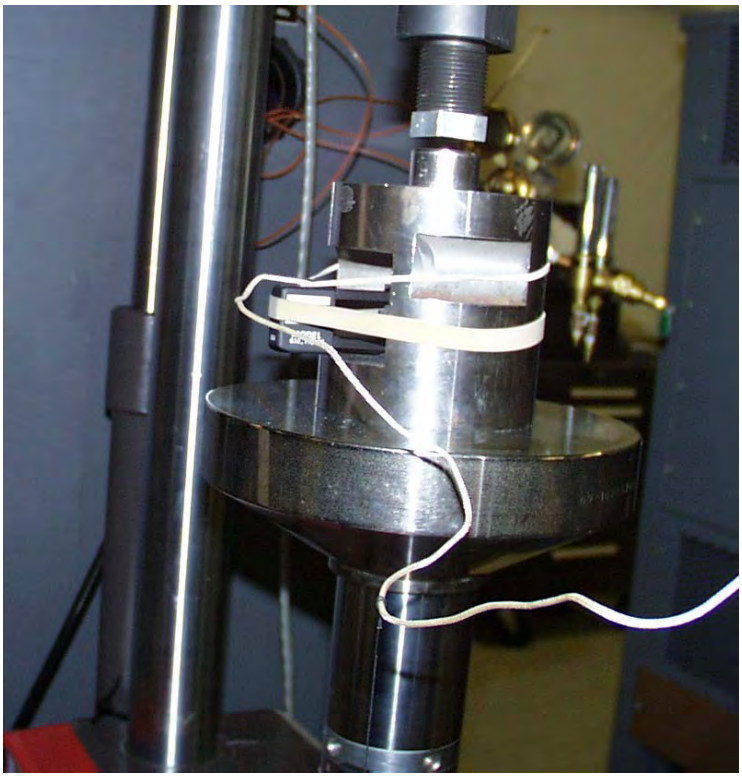

Figure 1: Sub-press Assembly in Centorr Universal Test Machine

Figure 2: Split Hopkinson Bar Compression Assembly

Material ID – Specimen ID	Test Strain Rate (s^{-1})	Actual Strain Rate (s^{-1})	Dia. (in)	Height (in)	Weight (gm)	Max. Stress (ksi)	Strain @ Max. Stress (%)	Strain @ Failure Stress (%)
0˚ - 1	0.001	0.000351	0.2743	0.3504	0.5112	37.5	4.5	4.7
0˚ - 2	0.001	0.000351	0.2745	0.3503	0.5134	35.2	4.2	4.3
0˚ - 3	0.001	0.000350	0.2744	0.3501	0.5149	38.4	4.2	4.4
0˚ - 4	0.001	0.000351	0.2745	0.3502	0.5173	34.7	3.2	3.3
0˚ - 5	0.001	0.000350	0.2744	0.3504	0.5123	35.7	4.1	4.4
0˚ - 6	0.1	0.0352	0.2745	0.3508	0.5186	41.2	4.1	4.2
0˚ - 7	0.1	0.0352	0.2744	0.3504	0.5132	41.6	4.5	4.6
0˚ - 8	0.1	0.0352	0.2744	0.3504	0.5127	41.8	4.6	4.7
0˚ - 9	0.1	0.0352	0.2742	0.3501	0.5097	39.5	4.5	4.5
0˚ - 10	0.1	0.0352	0.2746	0.3503	0.5200	40.6	4.0	4.0
0˚ - 11	10	3.02	0.2745	0.3504	0.5120	45.2	4.4	4.4
0˚ - 12	10	2.99	0.2742	0.3502	0.5101	44.3	4.3	4.3
0˚ - 13	10	2.99	0.2744	0.3501	0.5096	45.9	4.5	4.5
0˚ - 14	10	2.97	0.2745	0.3503	0.5151	46.8	4.8	5.5
0˚ - 15	10	2.99	0.2745	0.3504	0.5174	46.2	4.6	5.2
45˚ - 1	0.001	0.000360	0.2745	0.3604	0.5281	146	10.3	10.3
45˚ - 2	0.001	0.000360	0.2745	0.3609	0.5286	140	10.2	10.2
45˚ - 3	0.001	0.000360	0.2744	0.3614	0.5277	148	11.1	11.1
45˚ - 4	0.001	0.000360	0.2743	0.3611	0.5283	144	10.0	10.0
45˚ - 5	0.001	0.000360	0.2746	0.3613	0.5277	130	9.4	9.4
45˚ - 6	0.1	0.0360	0.2750	0.3613	0.5307	151	10.7	10.7
45˚ - 7	0.1	0.0361	0.2749	0.3615	0.5298	150	10.2	10.2
45˚ - 8	0.1	0.0362	0.2743	0.3611	0.5272	150	10.3	10.3
45˚ - 9	0.1	0.0361	0.2748	0.3612	0.5304	149	10.5	10.5
45˚ - 10	0.1	0.0361	0.2744	0.3610	0.5277	145	9.9	9.9
45˚ - 11	10	3.04	0.2744	0.3611	0.5284	167	11.1	11.8
45˚ - 12	10	3.04	0.2745	0.3619	0.5289	155	10.7	10.8
45˚ - 13	10	3.03	0.2746	0.3605	0.5262	149	10.1	10.1
45˚ - 14	10	3.03	0.2745	0.3615	0.5297	164	11.2	11.2
45˚ - 15	10	3.03	0.2744	0.3601	0.5278	154	10.2	10.2

Table 2: Lower Strain Rate Summary

Material ID – Specimen ID	Height (in)	Dia. (in)	Weight (gm)	Strain rate (s^{-1}) max - min	Maximum Stress (ksi)	Engineering Strain (%)
0º - 16	0.3503	0.2744	0.5144	1730-980	53.3	5.2
0º - 17	0.3503	0.2742	0.5141	1830-1010	54.9	5.6
0º - 19	0.3503	0.2743	0.5101	1680-990	51.5	5.1
0º - 20	0.3502	0.2744	0.5168	1650-970	54.0	5.4
0º - 21	0.3504	0.2746	0.5175	1700-880	51.0	4.9
45º - 16	0.3614	0.2744	0.5282	2610-900	146	11
45º - 17	0.3614	0.2750	0.5297	2460-880	139	13
45º - 18	0.3615	0.2750	0.5305	2510-900	150	11
45º - 19	0.3615	0.2745	0.5291	2720-1020	134	10
45º - 20	0.3607	0.2742	0.5278	2780-1130	132	10

Table 3: Split Hopkinson Bar Summary

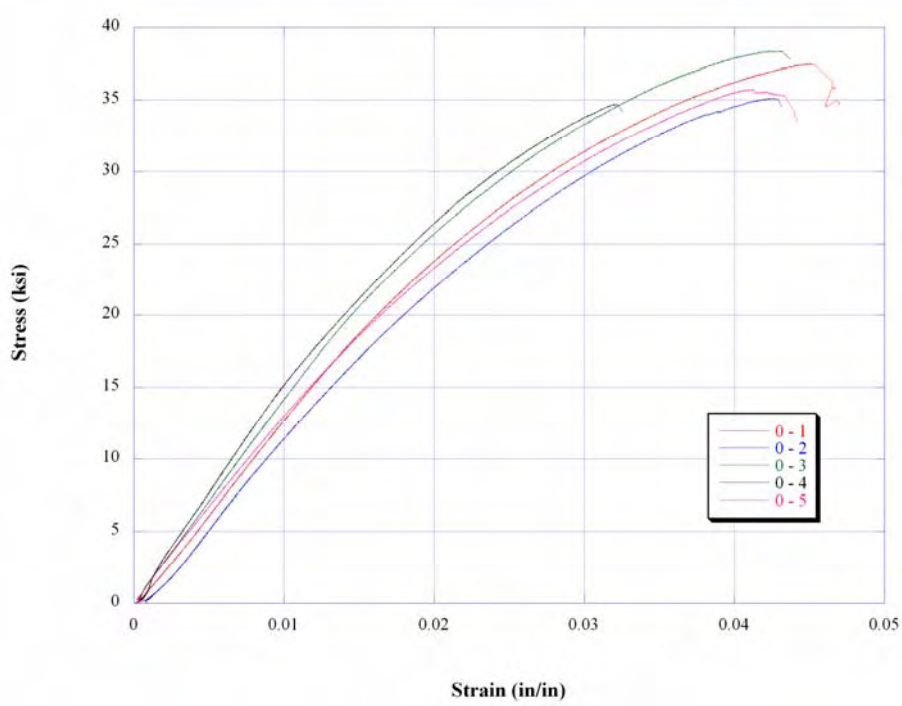

Figure 3: Boeing 0º at Test Strain Rate of 0.001s^{-1}

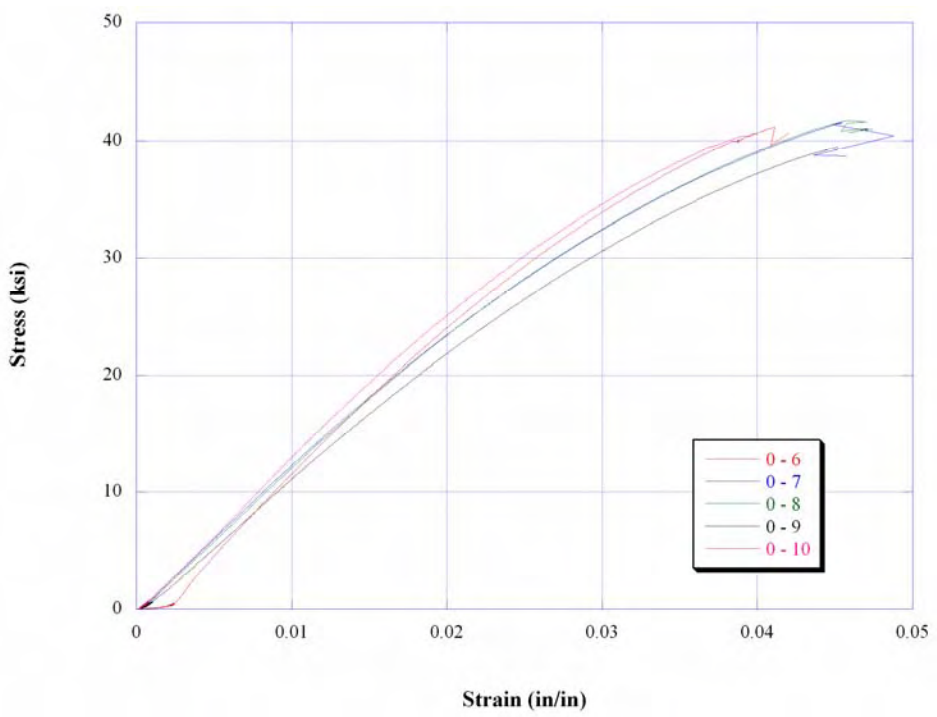

Figure 4: Boeing 0º at Test Strain Rate of 0.1s^{-1}

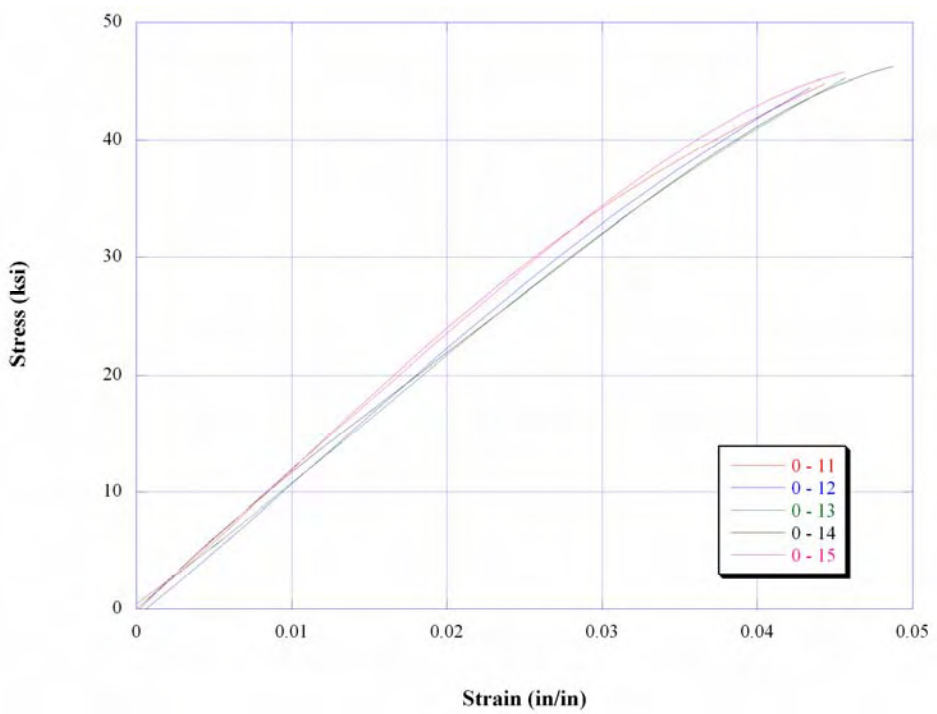

Figure 5: Boeing 0º at Test Strain Rate of 10s^{-1}

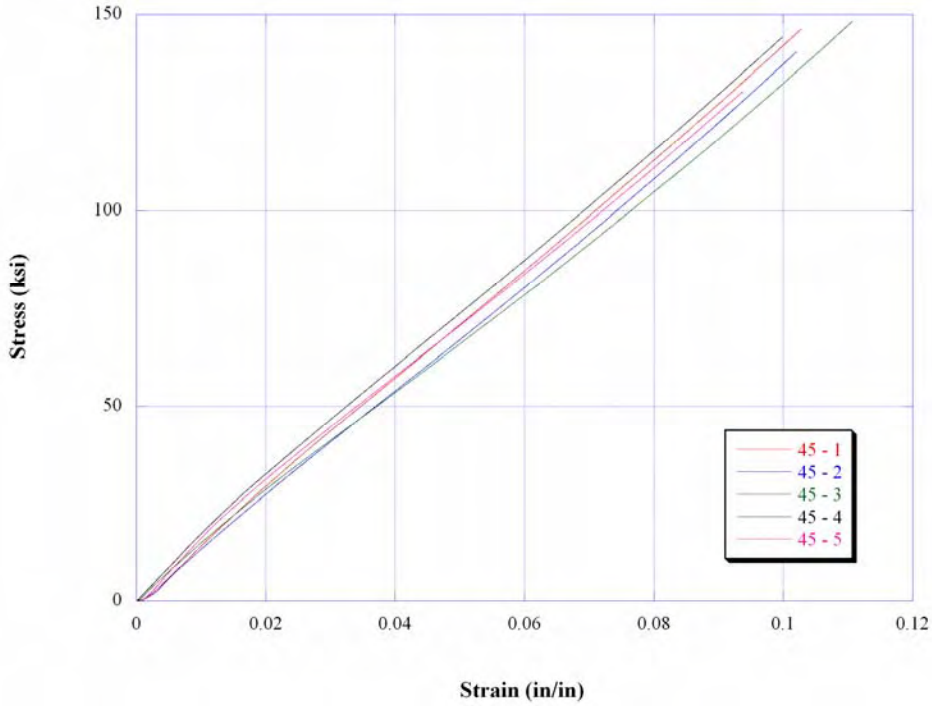

Figure 6: Boeing 45º at Test Strain Rate of 0.001s^{-1}

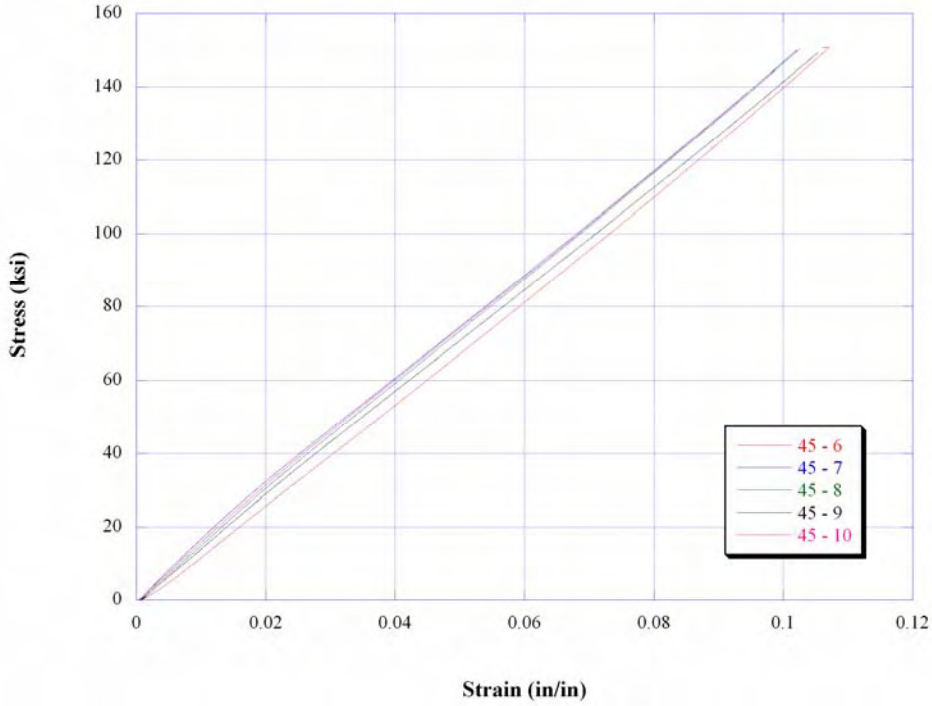

Figure 7: Boeing 45º at Test Strain Rate of 0.1s^{-1}

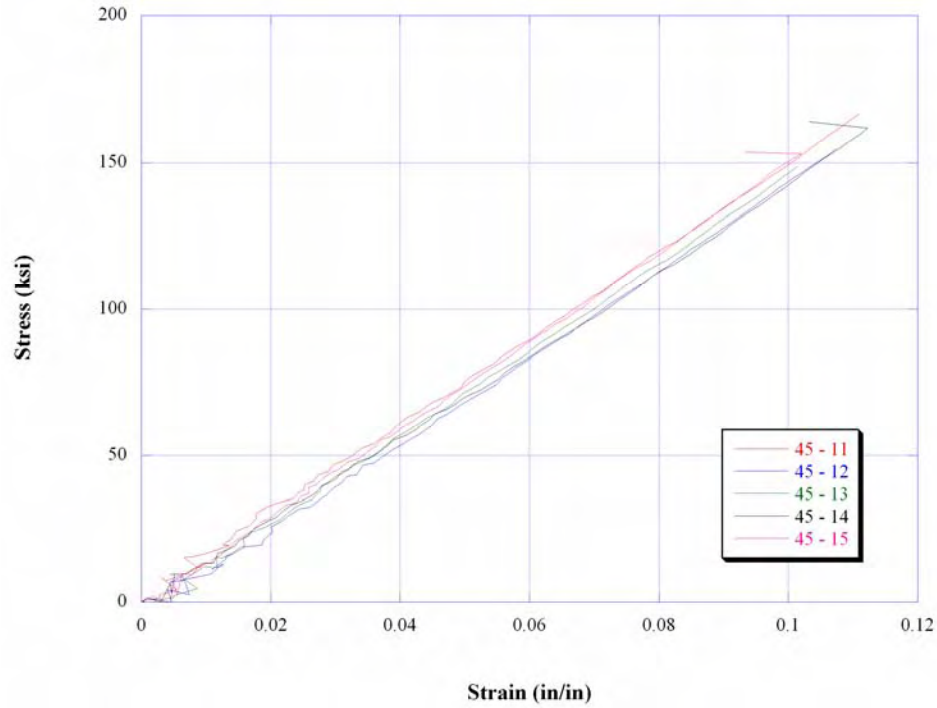

Figure 8: Boeing 45º at Test Strain Rate of $10s^{-1}$

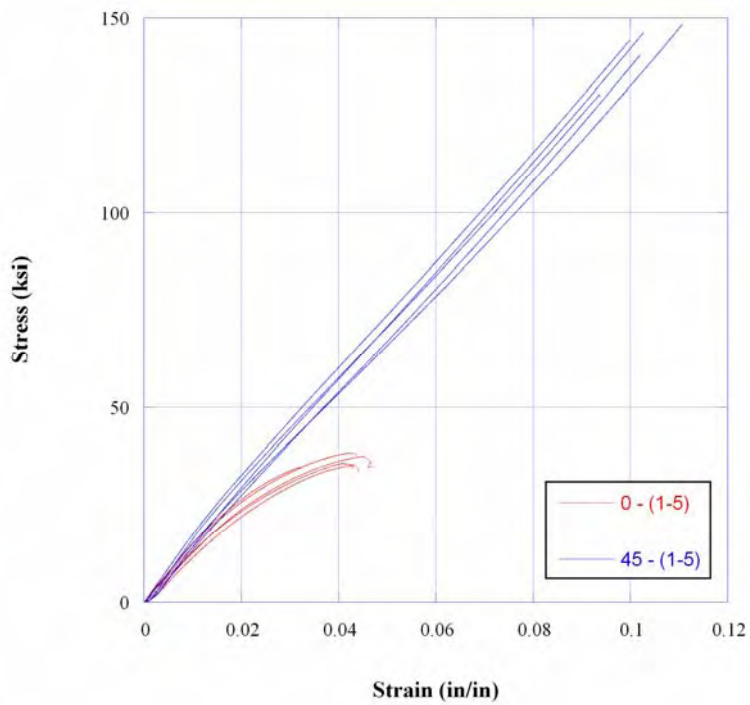

Figure 9: Boeing 0º and 45º at Test Strain Rate of $0.001s^{-1}$

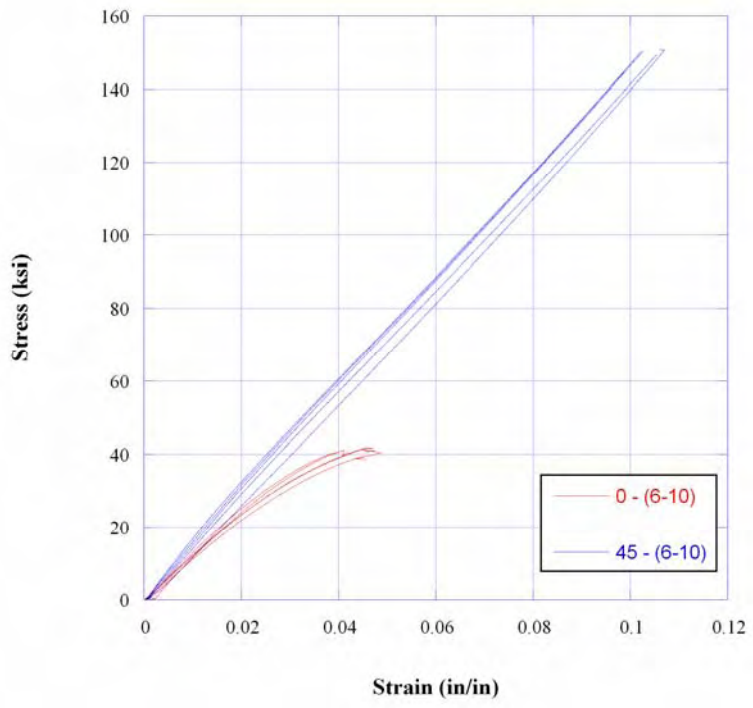

Figure 10: Boeing 0º and 45º at Test Strain Rate of $0.1s^{-1}$

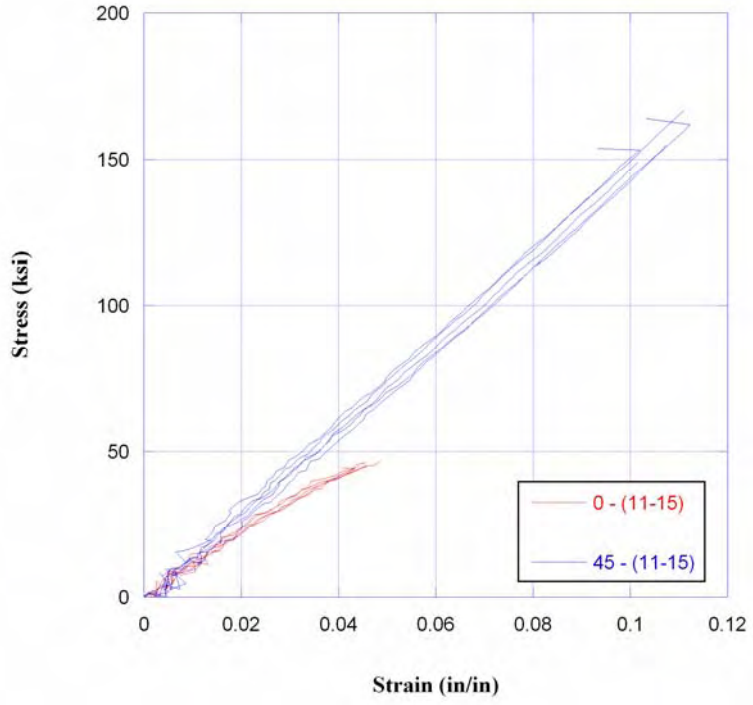

Figure 11: Boeing 0º and 45º at Test Strain Rate of $10s^{-1}$

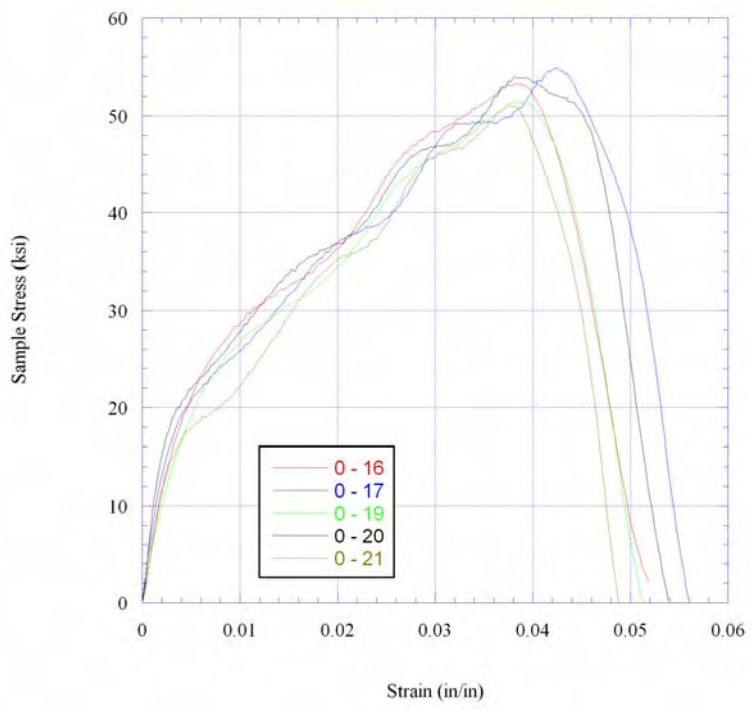

Figure 12: Boeing 0º at Test Strain Rates Greater Than 1000s^{-1}

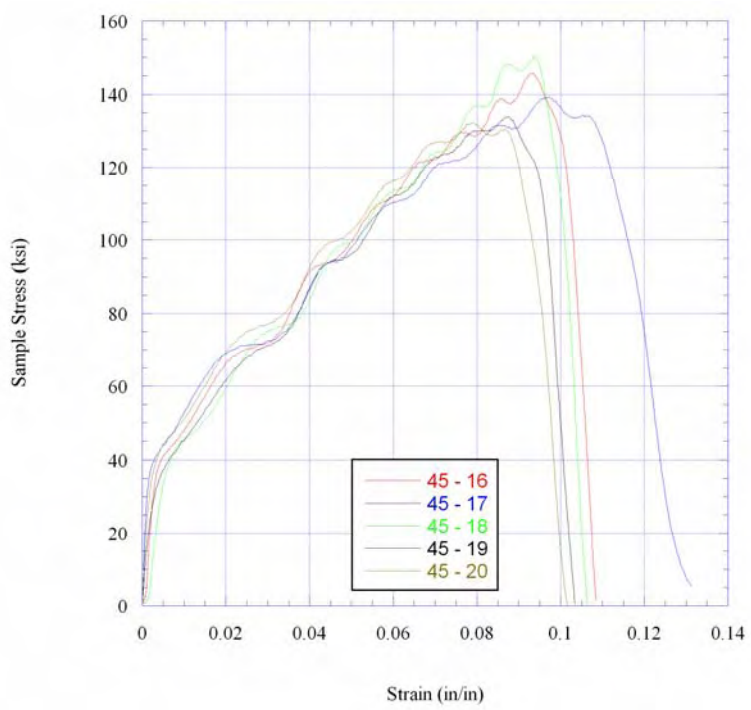

Figure 13: Boeing 45º at Test Strain Rates Greater Than 1000s^{-1}

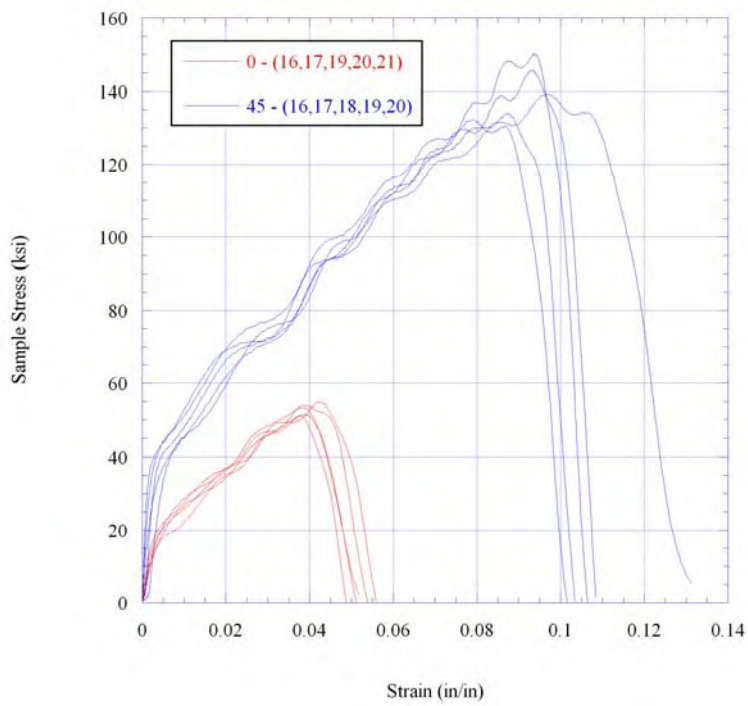

Figure 14: Boeing 0º and 45º at Test Strain Rates Greater Than 1000s^{-1}

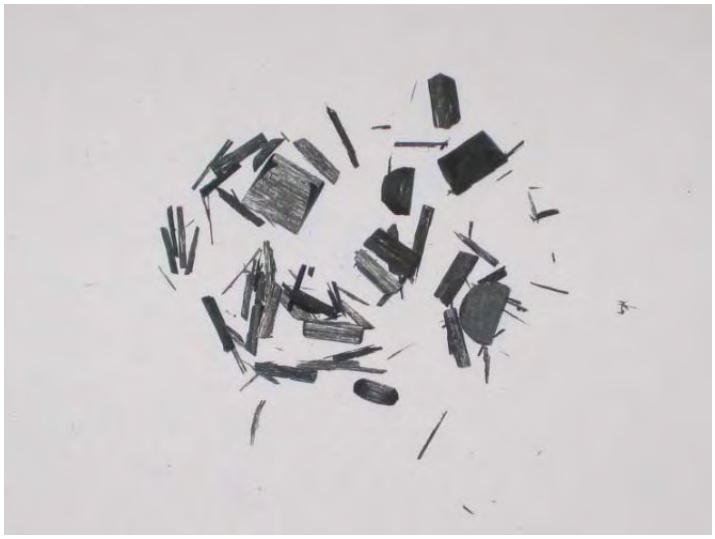

Figure 15: Boeing 0º Typical Specimen Post Hopkinson Bar Test

Figure 16: Boeing 45º Typical Specimen Post Hopkinson Bar Test

If there are any questions about the results of this work, please contact me.

David Urabe
Materials Engineering Group

Reviewer Initials: TA

Authors Initials: DU

Electronic CC:

T. Andrews

<u>Appendix B: BMS 2-812 input for LSDYNA constitutive model</u>
"*MAT_COMPOSITE_FAILURE_MODEL"

The lamina (ply) properties for BMS-2-812 given in the following table are for use in a constitutive model that assumed an orthotropic elastic *averaged* behavior for each element. For a multi-angle composite, this would imply pre-possessing of the plys contained in each element to produce the required averaged behavior input for the constitutive model. Alternatively, if the thickness of each element in a mesh represented a single ply of a laminate material then lamina properties could be used as the constitutive model input. These restrictions were not assumed to apply in the case of a unidirectional laminate, i.e., the unidirectional lamina properties were assumed to be applicable to elements that contained many ply layers.

The elastic properties and fiber direction strengths listed below are from Boeing and from communications with Steve DeTeresa. The matrix dominated compressive strengths were measured at LLNL and the tensile matrix properties were estimated. For the unidirectional lamina the A direction is the fiber direction, X is the fiber direction strength, B is the transverse direction (Y is the transverse strength) and C is the normal direction (Z is the normal direction strength). The fiber direction strengths were was assumed to be is rate insensitive. The strength enhancement factor for the matrix dominated compressive strengths was determined to be:

$$Rf = 1.20535*(edot)^{0.026178}$$

Table. BMS 8-212 input for LSDYNA material model
"*MAT_COMPOSITE_FAILURE_MODEL"

Density	$1.4493*10^{-4}$	lbf-sec^2/in^4
E_a	$1.71*10^7$ psi	Young's modulus in the longitudinal direction
E_b	$1.28*10^6$ psi	Young's modulus in the transverse direction
E_c	$1.28*10^6$ psi	Young's modulus in the normal direction
ν_{ba}	0.02545	Poisson's ratio
ν_{ca}	0.02545	Poisson's ratio
ν_{cb}	0.30000	Poisson's ratio
G_{ab}	$0.800*10^6$ psi	Shear modulus in the ab plane
G_{bc}	$0.367*10^6$ psi	Shear modulus in the bc plane
G_{ca}	$0.367*10^6$ psi	Shear modulus in the ca plane
S_{ba}	10,000 psi	In plane shear strength
S_{ca}	10,000 psi	Transverse shear strength
S_{cb}	10,000 psi	Transverse shear strength
XX_c	$1.5*10^5$ psi	Longitudinal compressive strength
YY_c	36,341 psi	Transverse compressive strength, LLNL testing at zero strain rate
ZZ_c	36,341 psi	Normal compressive strength, LLNL testing at zero strain rate
XX_t	$1.6*10^5$ psi	Longitudinal tensile strength
YY_t	8,000 psi	Transverse tensile strength
ZZ_t	8,000 psi	Normal tensile strength