Compression After Impact (CAI) Response of Quasi-Isotropic Laminates with and without Hy-Bor® for the Zero-Degree Plies

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ABSTRACT

Quasi-isotropic laminates consisting of 8-plies were tested for undamaged and post-impact compression strength. The material used was IM7/MTM-45 carbon/epoxy, however in one of the manufactured panels, the 0°-plies of IM7/MTM-45 were replaced with 0°-plies of Hy-Bor® 208 boron-carbon reinforced epoxy. Another manufactured panel consisted of Hy-Bor® 100 boron-carbon reinforced epoxy. To eliminate problems with end-brooming and/or global buckling, the four point bend method was utilized to induce compressive stresses into the laminates. Results show that the laminates with the Hy-Bor® 208 0°-plies had an average of 61% more compression strength for undamaged laminates and an average of 70% more compression strength for laminates that had experienced a 1.3 ft-lb impact. Results show that the laminates with the Hy-Bor® 100 0°-plies had an average of 44% more compression strength for undamaged laminates and an average of 20% more compression strength for laminates that had experienced a 3.1 ft-lb impact. The laminates with the Hy-Bor® material demonstrated a multiphase compressive failure event for impacted samples.

INTRODUCTION

In order to be more widely used and accepted, carbon fiber/polymer structural composites must improve upon their compression-after-impact (CAI) strength to meet the many damage tolerance requirements typically levied upon them. More damage tolerant (toughened) resins where introduced in the late 1980's and are still evolving today. This has enabled such structures as Boeing's 787 civil transport aircraft to make heavy use of structural components of carbon/epoxy composites.

It has been demonstrated that when placed in compression, microbuckling (or kink band formation) in the plies most aligned in the loading direction (usually denoted as"0°-plies") are responsible for ultimate failure of notched laminates [1, 2]. Therefore, in addition to increasing the toughness of the resin, it stands reason that improving the compression strength of the fibers, particularly of the 0°-plies could lead to a higher CAI strength. It came to the author's attention that Specialty Materials, Inc. has a product available (Hy-Bor®) that is a carbon/epoxy prepreg with boron fibers embedded in the prepreg. Since boron fibers have a high compression strength, this material was tested for residual compression strength before and after impact damage.

MATERIALS

The carbon/epoxy system chosen to make the laminates for this study was IM7/MTM-45 carbon/epoxy. This resin system is a toughened epoxy that can be cured in or out of an autoclave that is currently undergoing extensive testing as part of a Stanford University study. The two types of Hy-Bor® material were designated as" Hy-Bor® 208" since there are 208 boron filaments per inch of width and "Hy-Bor® 100" since there are 100 boron filaments per inch of width. The all carbon/epoxy laminates had a stacking sequence of $[+45,0,-45,90]_S$ and were cured at 80 psi in a platen press. The nominal thickness of these panels was 0.040 inches. The laminates with the 0°-plies of Hy-Bor® 208 had a stacking sequence of $[+45,90,-45,0_H]_S$ for the low-level impacts and a stacking sequence of $[0_H,45,90,-45]_S$ for the high-level impacts (the subscript H referring to Hy-Bor®) and where cured in the same manner. These panels, designated as "Hy-Bor® 208" in this report, had a nominal thickness of 0.043 inches. The laminates with the 0°-plies of Hy-Bor® 100 had a stacking sequence of [+45,0,-45,90]s for the low-level impacts and a stacking sequence of $[0_H,45,90,-45]_S$ for the high-level impacts and where cured in the same manner. These panels, designated as "Hy-Bor® 100" in this report, had a nominal thickness of 0.042 inches.

The four point bend method (ASTM D 5467) was to be used since it has been the author's experience that this test is a more representative test of the material's compression load carrying capability when performing CAI tests. In addition, by utilizing this method, undamaged laminates can be tested exactly as the damaged ones since end-brooming and global buckling are not failure modes. This gives a more direct comparison of the impacted and unimpacted laminates.

In order to manufacture the compression specimens needed for the four point bend method, honeycomb core and a suitable "tension" face sheet material needed to be chosen. High density aluminum core 1.5 inches thick with a density of 12 lb/ft^3 was chosen due to the high strength of the material being tested. The "tension" face sheet chosen was a bi-directional laminate of IM7/MTM-45 since the modulus of this laminate had already been characterized (13.0 Msi). This modulus would be needed when calculating the compressive stresses in the "top" or "compression" face sheet. Both face sheets were prepared for bonding to the aluminum core by mechanical abrading until water no longer "beaded" on the surface. An epoxy film adhesive was used to bond the laminates that made up the face sheets to the high density aluminum honeycomb core. This process took place at 250 °F and 80 psi to ensure good filleting of the adhesive between the core and face sheets. Once the face sheets had bonded to the core, beam sections 2.0 inches wide were cut from the sandwich panel to form the compression test specimens. A schematic of the compression test specimen used in this study is shown in Figure 1.

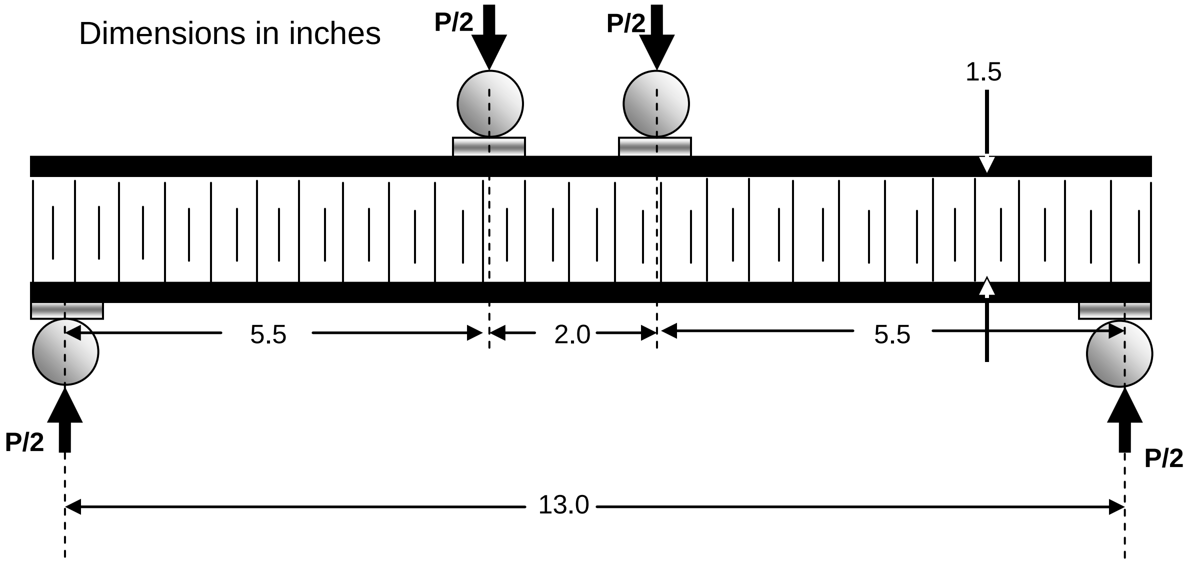


Figure 1. Schematic of compression specimen used in this study.

IMPACT TESTING

Some of the beam specimens of both IM7/MTM-45 and Hy-Bor® 208 and Hy-Bor® 100 were impacted utilizing an instrumented drop weight tower. The impact diameter used was 0.25 inches in order not to cause a large damage zone that would interact with the specimen edges. The two impact energies utilized were 1.25 and 3.1 ft-lbs. The aim of these impacts was not so much to create a realistic situation as it was to simply cause levels of damage in the specimen that would lower its compression strength. A visual indication of these levels of damage severity on the baseline and both type of Hy-Bor® specimens are shown in Figure 2 for the low-energy impacts and in Figure 3 for the high-energy impacts.

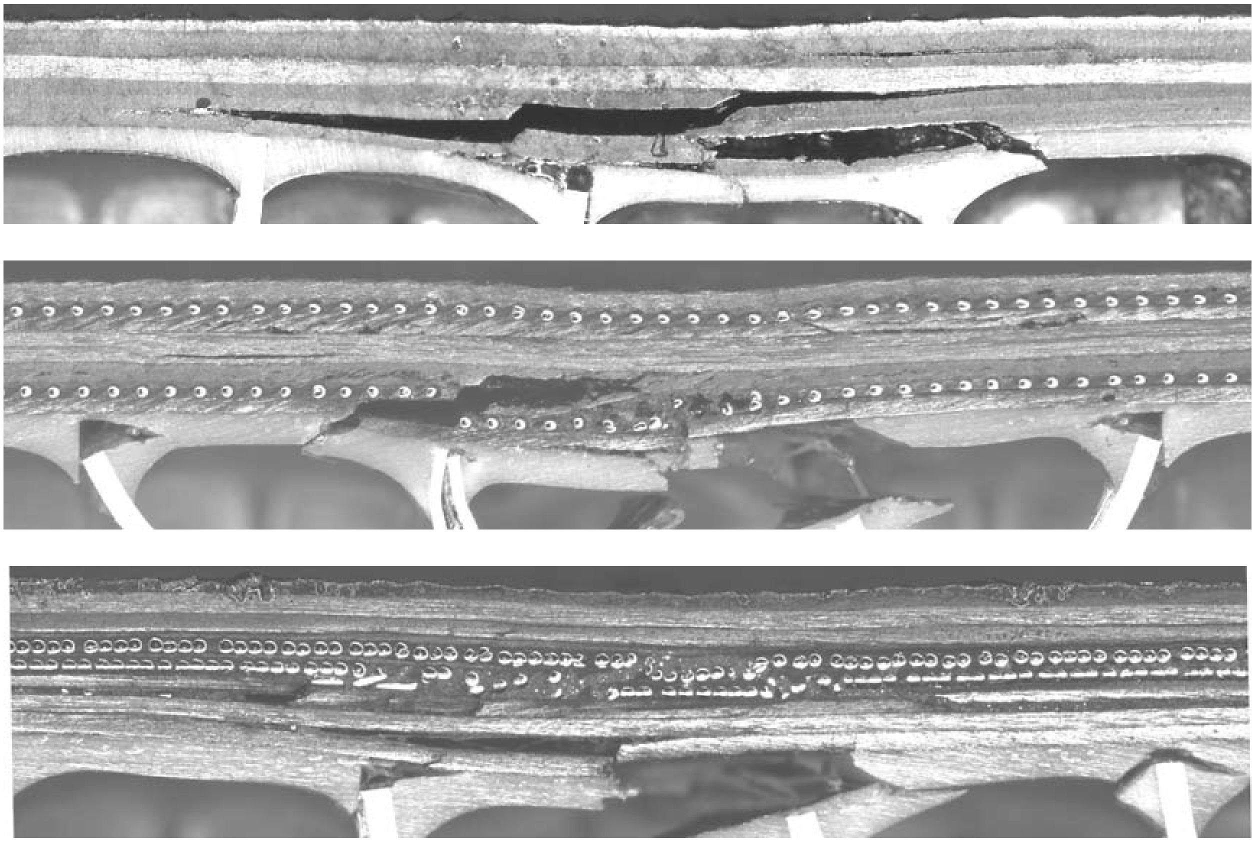


Figure 2. Low impact level damage to specimens used in this study.

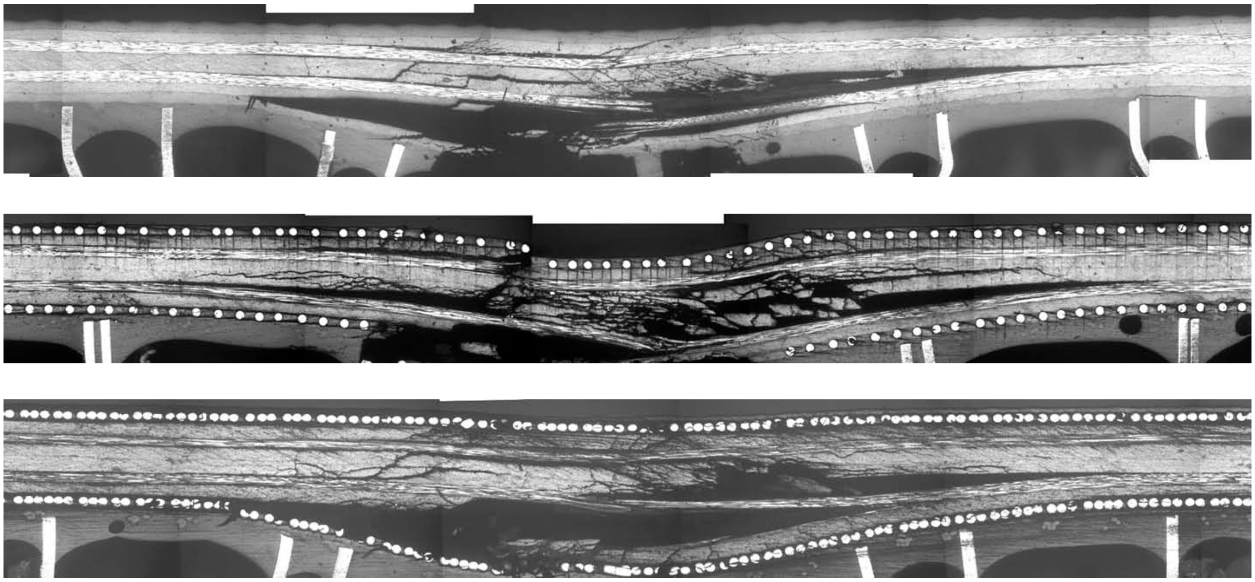


Figure 3. High impact level damage to specimens used in this study.

RESULTS

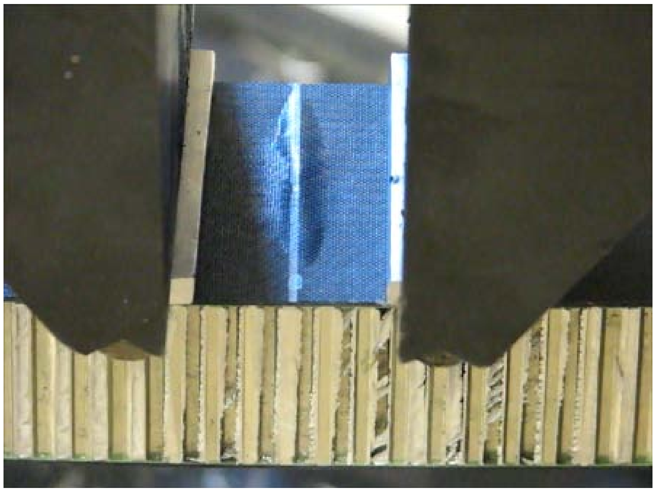
A summary of the compression strength values obtained are given in Table 1 for the specimens tested in this study.

Table 1. Compression strength values obtained in this study.

Specimen	Impact Energy (ft-lbs)	Ultimate Compressive Stress (ksi)
IM7/MTM-45	0	86.6 ± 3.1
Hy-Bor® 100	0	124.9 ± 4.3
Hy-Bor® 208	0	139.7 ± 12.1
IM7/MTM-45	1.25	58.1 ± 1.6
Hy-Bor® 100	1.25	75.8 ± 7.0
Hy-Bor® 208	1.25	98.8 ± 15.6
IM7/MTM-45	3.1	53.8 ± 1.7
Hy-Bor® 100	3.1	64.6 ± 3.1
Hy-Bor® 208	3.1	73.5 ± 5.1

One unique aspect of the impacted Hy-Bor® samples was that the failure occurred in two distinct phases. In the first phase of failure some of the plies “pop up” in what is thought to be sublaminate buckling from a large delamination above or below the 0°-plies. Instead of failing catastrophically the boron fibers apparently have enough strength to still carry the applied load. As the load is increased further, catastrophic failure occurs as the second and final phase of failure. This behavior highlights the structural aspects of the added boron and results in a type of “fail-safe” material akin to skin/stringer structure. Figure 4 shows photographs of the phases of compressive behavior for the Hy-Bor® 208 laminates. These photographs were taken from a digital video of a CAI test.

1. Blister pops up above impact site upon sublaminate buckling between 80-90% of ultimate failure load



2. Blister has undergone very little growth right before ultimate failure

3. 0°-plies fail resulting in ultimate failure

Figure 4. Failure phases of impacted Hy-Bor® 208 specimens.

A graphic summary of the results in Table 1 are shown in Figure 5.

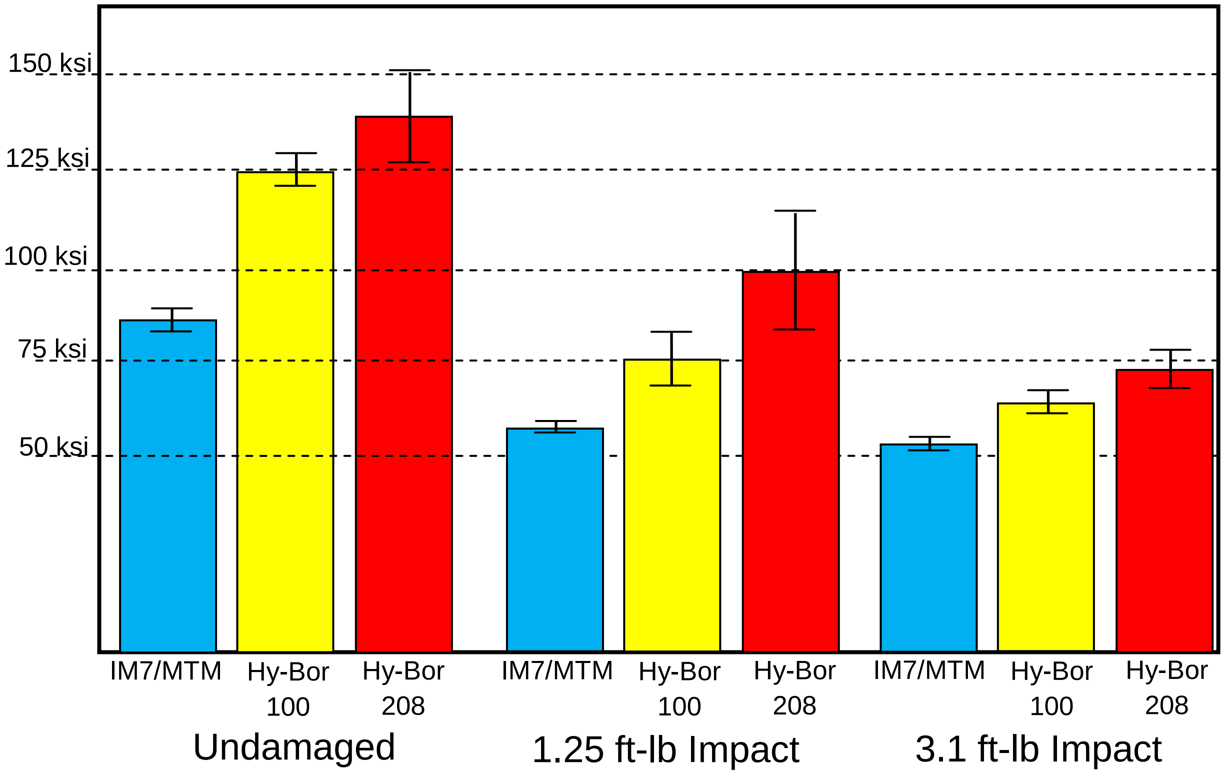


Figure 5. Graphical summary of compression strength values

1. Soutis, C. and Curtis, P.T. (1996). Prediction of the Post-Impact Compressive Strength of CFRP Laminated Composites, *Composites Science and Technology*, **56**: 677-684.
2. Soutis, C. and Fleck, N.A. (1990). Static Compression Failure of Carbon Fiber T800/924C Composite Plate with a Single Hole, *Journal of Composite Materials*, **24**: 536-598.