Potential Weight Benefits of IM7/8552 Hybrid Thin-ply Composites for Aircraft Structures

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Composite materials have increasingly been used for aerospace applications due to improved performance and reduced weight compared to their metallic counterparts. Inclusion of thin-ply material, plies with cured thickness half or less than standard-ply composites, have potential to improve performance and reduce structural weight further. The effect of thin-ply material on the weight of aircraft structure was investigated by examining wing cover weight reduction. To minimize the effects on manufacturing due to using thin plies, hybrid laminates were examined that used thin ±45-degree plies to replace their standard-ply counterparts in laminates. Compression after impact (CAI) tests were conducted to examine the possible weight savings that could be gained by increasing the design allowables that were used to size the wing upper cover of a semi-span test article. A large increase in CAI strength was observed for quasi-isotropic hybrid laminates, whereas less improvement was seen for hard hybrid laminates such as found in the wing cover. For laminates design by CAI strength, weight savings of about 13% were found using the hybrid hard laminates compared to the standard-ply laminates. Whether similar weight savings could be expected for structure sized using tension after impact allowables will have to be investigated further. Notched specimens were tested to examine possible weight savings using hybrid laminates in regions that are sized using discrete source damage requirements. As expected, the hybrid laminate had marginal improvements over the standard-ply laminate for compression with a notch present. The hybrid laminate, however, exhibited about 20% lower strength than the standard-ply laminate counterpart for tension with a notch. The failure mode of the hybrid specimens was a brittle, self-similar crack, which differs from the standard-ply specimens that failed by significant amounts of delamination and fiber splitting. In light of the apparent reduction in notched tensile strength, additional investigation is required to assess the use of hybrid laminates for areas containing discrete source damage, and their effect on weight of such regions.

I. Nomenclature

A_{ij} = Classical lamination theory membrane stiffness matrix terms, i, j = 1,2,6
D_{ij} = Classical lamination theory bending stiffness matrix, i, j = 1,2,6
E_x, E_y = Young's modulii in laminate x-direction and y-direction, respectively
G_{xy} = In-plane shear modulus
N_x = Running load in skin panel
ν_{xy}, ν_{yx} = In-plane poisson ratios

II. Introduction

Aircraft structures fabricated from composite materials can yield weight saving benefits compared to their metallic counterparts, but they can still be overdesigned due to numerous failure mechanisms that must be accounted for during the design process. A standard approach is to use design allowables that are specific to each damage mechanism such as presence of a delamination or of discrete damage. These allowables are called "damage-tolerant allowables", and they can be significantly lower than the unnotched allowables to ensure that the composite structure will retain

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integrity in the presence of such damage. Since damage-tolerant allowables are applicable for the majority of the structure, they lead to a structure that is overdesigned for the normal operational conditions. Significant weight savings for these composite structures can be gained if the pertinent, damage-tolerant design allowables can be increased. The use of thin-ply composite material in a hybrid laminate, a laminate consisting of both thin-ply and standard-ply material of a single or multiple material systems, may provide the desired increase in design allowables. However, it is desirable to utilize a hybrid of thin- and standard-thickness plies to minimize the number of plies that need to be laid down in order to minimize increase in the manufacturing time over all standard-ply laminates. Therefore, it is necessary to understand how these thin-ply hybrids perform, and to develop the associated design methodologies and recommendations.

A joint study was initiated by NASA Langley Research Center (LaRC) and Glenn Research Center (GRC) in 2017 in support of the NASA Convergent Aeronautics Solutions (CAS) project to investigate potential benefits of thin-ply composites in the design of aircraft structures [1]. The study utilized two thin-ply areal weights of IM7/8552 prepreg material produced by the Hexcel corporation, 70 grams per square meter (gsm) and 30-gsm material, along with the associated standard-ply material having an areal weight of 190-gsm. (Note that these areal weights refer to the carbon fiber mass in the prepreg.) A series of standard American Society for Testing and Materials (ASTM) tests for laminates of all thin-ply and hybrid laminates was conducted, and the results of those tests indicated that a potential weight savings over 20% could be obtained by using thin-ply hybrid laminates. This paper reports on a study that is a follow-on to Ref. [1] (called the "coupon study" herein), that was conducted from October, 2017 through September, 2018 to investigate potential weight benefits by incorporating thin-ply material into the design of aircraft structure. In particular, the study focused on two design criteria that have been found to influence certification requirements for significant portions of civil aircraft primary structures. These criteria are the barely visible impact damage (BVID) criteria requirement for ultimate loading conditions, and a 2-bay notch criteria representing self-evident, discrete-source damage that must be withstood for reduced loads to enable safe return and landing. Because of the results found in the coupon study, the present study incorporates testing of larger coupon and subcomponent hybrid specimens to examine the compression after impact (CAI) strength that is used to account for BVID, and the predicted failure load of a stiffened panel with a 2-bay notch. While applicable to all aircraft structures (e.g., wing, fuselage), the proposed paper will examine the possible weight savings based on the cover panel skins from a composite semi-span wing test article that was tested at NASA LaRC [2-7]. This test article has been used as the baseline for a previous weight savings investigation [8]. The study presented herein demonstrates the process flow from initial coupon testing, to additional coupon and subcomponent testing, to the increased allowables (strengths) leading to reduced weight designs.

III. Compression After Impact Specimens

Typical aircraft structure, such as a wing skin, is sized during design using compression and tension after impact allowables in order to account for undetected damage scenarios, because the primary structure must be able to carry design ultimate loads with such damage. The coupon study indicated that benefits of the hybrid thin ply could be obtained for both tension and compression based upon test results [1]. In the current study, compression after impact specimens of hybrid laminates are designed and tested to compare to standard-ply laminates. The design space available for hybrid laminates using various combinations of standard-ply and thin-ply lamina is extremely large, and to date no design guidelines exist for their use. Therefore, rather than try to design the hybrid laminates to fail at the same loads as the standard-ply laminates, the approach herein replaces some of the plies in a standard-ply laminate with thin plies. The resulting hybrid laminate was designed to match the thickness (and thus weight) and the stiffness of the original standard-ply laminate and the CAI strength differences determined. Therefore, the approach taken herein for all hybrid laminates was to replace the ±45 standard-ply lamina with thin-ply lamina, and to keep the 0-degree and 90-degree lamina as standard-ply to provide the main load carrying capability. The thin-ply materials were placed such that no adjacent plies have the same orientation. This was done because design experience with standard-ply composites has shown that ply blocks of the same orientation have reduced resistance to damage, such as splitting between fibers.

Table 1 shows the four specimen laminates used in the CAI study. There were two types of laminates used, a quasi-isotropic laminate and an axially stiff (hard) laminate with the ratio of equivalent axial stiffness to transverse stiffness (stiffness ratio) equal to 2.5 as shown in Table 1. The thin-ply and standard-ply material used for the panel fabrication were the same 70-gsm and 190 gsm IM7/8552 material of Ref. [1]. The 190-gsm IM7/8552 material is also identical to that being used by the Advanced Composites Project (ACP) and the current study compares to results developed by ACP where applicable. All CAI specimens were manufactured at LaRC using the hand layup approach given in Ref. [1]. The standard-ply specimens provide the baseline to which the hybrid specimens are compared. The hybrid

CAI test specimens were designed such that their difference between the equivalent engineering properties and the classical lamination theory membrane and bending stiffness matrix entries (A_{ij} and D_{ij}, respectively) were minimized with respect to the standard-ply specimens to the greatest extent possible. Table 2 presents the differences between the hybrid and standard-ply laminates which shows that agreement is very good and within 5% for most terms. Impact and CAI testing was conducted at the Wichita State University National Institute for Aviation Research (NIAR).

For each of the four laminates, 4 in. by 6 in. specimens were cut to be used in an impact study to determine the BVID energy for each laminate, and then in the CAI testing. The impact study involved impacting specimens with five different energy levels and evaluating the damage imparted to the specimens, with drop-weight impacts imparted per ASTM D7136 [9]. The goal of the impact study was to identify damage that could be classified as BVID, while at the same time to avoid producing a total damage area within the specimen that reached the impact fixture edge supports. Impacted specimens were examined for dent depth, and for the total internal damage area using through-thickness ultrasound (C-scan). Table 3 summarizes the impact energies used in the impact survey, and the resulting damage that was measured. The required BVID criteria was a dent depth of 0.06 in, obvious surface splits/cracks, surface delamination, surface fiber breakage, or a combination of these damage modes. Examination of the C-scans for all of the impact survey specimens indicated that the damage area did not extend to the edge supports at 40 ft-lb and 50 ft-lb impact energy for the quasi-isotropic and hard laminates, respectively. These impact energies were chosen because they were the maximum levels that did not have encroachment of the standard-ply laminate damage areas on the test fixture boundaries. At those impact energies, the dent depths are less than the desired 0.06 in, but there are also surface splits/cracks, so the damage met the BVID threshold. Therefore, the impact energies chosen for the quasi-isotropic and hard laminates were 40 ft-lb and 50 ft-lb, respectively.

Four specimens of each laminate type were impacted and then tested for CAI strength per ASTM D7137 [10]. Table 4 presents the CAI data for the test specimens. Figure 1 shows typical impact site damage for the four CAI test specimen types at the specified impact energy levels. The average dent depth for the standard-ply quasi specimens is only about half of that for the hybrid quasi specimens, however, the corresponding average damage area is about 30% larger for the standard-ply quasi specimen as seen in Figure 2. Figure 3 shows the through thickness damage obtained using computed tomography (CT-scans). From these images, it is seen that for the quasi-isotropic laminates, the through thickness damage in the hybrid laminate contains breaks in the lamina, where the standard-ply laminate does not. However, delamination cracks extend further through more plies in the standard-ply laminate than the hybrid laminate. The average dent depth for the standard-ply hard specimens is comparable to the hybrid hard specimens (within 5%), however, the corresponding average damage area is about 20% larger for the standard-ply hard specimen as seen in Figure 2. For the hard laminates, the images in Figure 3 show little or no lamina breaks for the hybrid-ply laminate, while lamina breaks are clearly seen for the standard-ply laminate. Figure 4 shows the comparison of the average normalized CAI strengths, where it was observed that the hybrid specimen quasi-isotropic and hard laminate strengths exceed the corresponding standard-ply specimen strengths by about 37% and 14%, respectively. The increase in the CAI strength can be applied to adjust design allowables, and thereby lead to reduced weight.

Potential weight savings for the upper cover of the semi-span wing [2-7] were determined using the CAI design allowable curve shift resulting from the improved CAI strength for the hybrid laminates. The semi-span wing in the references was manufactured using composite laminate with specified numbers of "stacks". Each stack consisted of discrete layers of dry, unidirectional carbon fibers held together with fine chain stitching in a stack sequence of $[45/-45/0/90/0/-45/45]_T$, where the 0-degree plies were twice as thick as the other plies. Stacks of material were used as the laminate building blocks to simplify the manufacturing process which consisted of stitching the carbon fiber stacks into preforms and then infusing the preforms with the matrix resin. Design allowables were determined for laminate thicknesses ranging from 2 to 13 stacks for tension after impact (TAI) and CAI, where CAI allowables were used to size the upper cover. The simplified approach used to estimate weight savings for hybrid laminates for this application assumed that the percentage increase in strength could be extrapolated from the limited data points to all thicknesses. Therefore, the CAI design allowable strength values for the stacks were increased by the same percentage, either quasi-isotropic or hard hybrid laminate strength increase percent values. Figure 5 presents the original CAI allowable design strains and the adjusted curves representing the improvements represented by the hard and quasi-isotropic hybrid laminate percentages. To apply these improved allowables to the upper cover design, the upper cover skin was examined. Figure 6 shows the original design for the upper cover skin that was primarily sized by CAI strength, where the regions are identified by the number of stacks ranging from 5 to 13. The first approach taken to calculate impact of hybrid laminates on wing panel mass was to retain the discrete characteristics of the stack approach to manufacturing by calculating the number of stacks that could be removed. Weight savings were calculated by first converting strain allowables into the equivalent running load allowables presented in Figure 7. Then, the revised number of stacks for a hybrid laminate stack was determined from the original number of stacks in the standard-ply composite. In the figure, this process is shown by the dashed line arrows for 6 stacks, where the black lines show the equivalent strength stacks of hard hybrid laminate and the grey lines show the equivalent strength stack of quasi-isotropic hybrid laminate. It is observed that using the hard laminate curve would not lead to a reduction in stacks since the reduction is less than a single stack (about 0.7 stacks), whereas using the quasi-isotropic curve would lead to a reduction of one stack (part of the 1.6 stack reduction). However, the actual wing stack is better represented by the hard laminate, and since the wing was designed using stacks to define thicknesses, no weight reduction would be achieved if the design continued to be defined by stacks. Note that the stack method was used for simplicity in manufacturing where the stack layup specification was optimized for standard-ply composites. It should not be surprising that a simple substitution of hybrid composite into the standard-ply stack design would not be optimal. Therefore, the weight reduction potential of hybrid laminates was then examined using a laminate thickness approach.

Assuming a manufacturing method not limited by the large, discrete thickness changes associated with using stacks designed for standard-ply composites, weight savings potential for hybrid laminates is estimated using the assumption of continuous variation of laminate thickness. To obtain this estimate, the running load allowables computed previously were plotted as a function of laminate thickness as shown in Figure 8. Using a similar approach as for stacks, for a given thickness of standard-ply laminate its strength (i.e., N_x capability) is determined. Locating this N_x value on the adjusted hybrid laminate curve, the equivalent strength hybrid laminate thickness is determined. This approach led to reductions in thickness ranging from about 12% at 0.275 in. (5-stack equivalent) to about 16% at 0.715 in. (13-stack equivalent). Similar percentages would apply to the stringers if they were sized by strength, as well, leading to an average cover panel weight savings for the wing design considered of approximately 13%. As reported in Ref. [8], the original upper cover weight was 1288 lbs, leading to a hybrid cover panel weight of about 1120 lbs. In Ref. [8], the weight of a PRSEUS upper cover panel was reported as 1182 lbs, which indicates that changing to a hybrid laminate could potentially have an improved weight compared to conventional wing construction with blade stiffeners and standard-ply laminates as well as to PRSEUS construction. If the stringers were sized primarily by another mode such as buckling, the wing weight savings will be less. Weight savings presented here are preliminary as this is a work in progress, and the assumption of similar strength increase percentages for all laminate thicknesses must be proven. The open hole tension results of Ref. [1] may imply that similar weight savings is expected to be seen for areas sized with TAI allowables, such as the lower cover panel, but this should be verified through additional testing. BVID is category 2 damage as defined by Federal Aviation Administration (FAA) AC 20-107B, and it appears that hybrid thin-ply laminates may provide benefits for category 1-3 damage [11].

IV. Notched Composite Specimens

Another typical aircraft design criteria involves the presence of discrete source damage, such as a through-crack with a severed stiffener, damage that could be incurred by a high-energy impact during flight. Encouraging results from open-hole tension and compression results provided in Ref. [1] suggested the possibility of increasing the strength of composites under notched damage scenarios using hybrid laminates. The approach chosen in the present study to investigate the potential weight savings of hybrid laminates was to use the 2-bay crack analysis tool developed as part of the ACP to predict the strength of a stiffened panel with a 2-bay crack [12]. In order to use this tool, the failure load of an unstiffened skin with a through-notch is required, where the notch length is dependent upon the stiffener spacing. Two smaller specimens are used to determine the tool input to minimize material usage and to simplify testing. Therefore, to use the tool for a particular laminate, small and intermediate sized through-notch specimens are tested, and the results of those tests are used to create a curve to predict the strength for an unstiffened panel with any notch length. Table 5 provides the designs for the unstiffened panels to be tested in the present study to provide the failure loads needed for the 2-bay crack analysis tool. As with the CAI test specimens, the notched hybrid test specimens were designed to have similar weight and stiffness of the standard-ply specimens. The difference between the equivalent engineering properties and the A_{ij} and D_{ij} entries were minimized to the greatest extent possible as shown in Table 6. The standard-ply panels were being tested as part of the ACP tool validation testing and the hybrid panels were tested for the present study. Table 7 gives the definitions of the small tension and compression specimens, the intermediate tension specimens, and the loading conditions. ACP is also testing intermediate compression specimens, but due to limited availability of thin-ply material and schedule requirements, the present study did not include intermediate compression specimens. Instead, the intermediate compression specimen strength would be approximated using the strength of the intermediate unstiffened tension standard-ply panel and the ratios of the tested unstiffened compression hybrid and standard-ply test article strengths. The hybrid specimens were manufactured at LaRC using the hand layup approach given in Ref. [1]. ACP specimens were manufactured by Spirit Aerosystems, of Wichita, KS, as part of the ACP activities.

Test results for the standard-ply and hybrid notched skin laminates provide the necessary unstiffened panel data to be used with the ACP 2-bay crack analysis tool for predicting the failure load of standard-ply and hybrid thin-ply stiffened panels. Due to delays in testing, only the small notch specimens were tested in time to be included herein. Table 8 gives the average strengths for the small notch specimens. As expected from the results of Ref. [1], the small hybrid compression specimens showed very marginal improvement in strength, less than 2%, compared to the standard-ply specimens. The compression failures for the two types of specimens show a slightly different character as seen in Figure 9. Damage in the standard-ply specimen exhibits more delamination with bulging and fiber splitting, as highlighted by the yellow boxes in part a) of the figure. The hybrid laminate shows significantly less bulging and no fiber splitting, as highlighted by the yellow boxes in part b) of the figure. Additionally, the damage appears to be more self-similar crack formation (the two crack tips move at nearly constant velocity and symmetrically from a zero initial length) in the hybrid laminate as the cracks extend out of the yellow highlighted regions and reach the edge of the specimen nearly directly across from each other. Despite these difference in the appearance of the failures, for a notch in a compression dominated region, the hybrid laminate provides only a marginal increase in strength performance compared to the standard-ply laminate based on this limited data (less than 2%). In contrast to the small compression specimens, the small notched tension results were not as expected based on the results of Ref. [1]. Based on the open hole tension coupon test results, it was anticipated that the hybrid laminate would provide improved strength over standard-ply laminates in notched tension tests. However, the small hybrid notched specimens exhibited significantly lower strength, being about 20% lower in tension strength than the standard-ply specimens. Figure 10 shows representative failures for the two types of specimens, where it is clearly seen that the hybrid specimen exhibits self-similar cracking, but the standard-ply laminate exhibits significant areas of delamination and fiber separations. It appears that including the thin-ply material suppressed damage, such as microcracking and delamination, which resulted in a more brittle failure. On the other hand, the standard-ply specimens exhibited numerous small failures prior to final failure, which relieve stress concentrations and allow the higher loads to be reached. Figure 11 shows the load vs. test machine stroke for the six small notched tension specimens, where it can be seen by the curves being nearly coincident that the specimens had similar stiffness, as designed, with the standard-ply specimens failing at higher loads. Strip strain gage locations adjacent to the notch tips are seen in Figures 9 and 10, and were located on both surfaces of the specimen (4 total strip gages). Figure 12 shows the average strains measured near the two crack tips, where it is seen that the hybrid specimens show little nonlinearity in strain compared to the standard-ply specimens that exhibited significant nonlinearity and jumps in the measured strain due to redistribution in load resulting from local failures. Additionally, these gages adjacent to the notch tips for the standard-ply specimens failed due to the very high strains at loads significantly below the maximum load as indicated in Figure 12. This trend was seen by successive gage regions along the strip gages as loading was continued. Because of these unexpected results for the notched tension specimens, additional investigation is required to understand these discrete source damage results.

Several directions are possible to evaluate and understand the performance potential of the hybrid laminates to discrete source damage and, in particular, to understand the notched results described herein. Consistency in the quality of the laminates produced by the two organizations needs to be investigated, including voids, fiber volume fractions and lamina thicknesses. Only a cursory quality assessment has been performed at present to examine these features. In addition, the design of a hybrid laminate optimized for discrete source damage is unknown, and the design space for laminates allowing for a ply-thickness variation is largely unexplored. In the present study, thin-ply layers were only used for ±45-degree plies, and the hybrid panel design was constrained to closely match the stiffness and mass of the existing standard-ply notched test specimens.

Improved understanding of the damage mechanics of hybrid laminates is essential to their optimum design. An initial step in improving our understanding should include detailed comparisons of the failure development and propagation of standard-ply laminates with hybrid laminates. While the strengths for the small notched tension specimens were not as expected, there may be applications where a brittle but easily predictable failure at a lower load could be more desirable than failure at a higher but less predictable load. In addition, if there are containment features to arrest crack propagation, such as stitching of composite materials that has been shown to arrest crack growth [13-15]), having a more easily predictable failure at a lower load, where the crack is then arrested, may be a more favorable failure scenario than one that exhibits progressive damage growth prior to failure. Therefore, it is unclear at this time whether the hybrid thin-ply composite would be beneficial in meeting the recommendations of the FAA AC 20-107B for category 4 and 5 damage [11].

V. Concluding Remarks

The effect of thin-ply composite material on aircraft structure was investigated for potential increases in laminate strength for failure modes that influence wing cover weight. To minimize the effects on manufacturing cost due to using a large number of thin-ply layers, hybrid laminates that combine thin-ply layers with standard-ply layers were the focus of the assessment. In the present study, thin plies of 70 gsm and 30 gsm material were only used for the ±45-degree plies to replace their counterparts in standard-ply laminates. Compression after impact tests were conducted to determine typical strength improvements possible with hybrid laminates. These results were used to examine the possible weight savings that could be obtained from increasing the design allowables in sizing the wing upper covers of the reference semi-span test article [5-8]. While a more significant increase in CAI strength for hybrid laminates was observed for quasi-isotropic laminates, reasonable improvement was seen for hard laminates such as found in the wing upper cover. Estimated weight savings of about 13% were found using the hybrid laminates compared to the standard-ply laminates. Whether similar weight savings could be expected for lower cover structure sized using tension after impact allowables will have to be investigated, particularly in light of the notched tension results presented herein. Notched specimens were tested to examine possible weight savings from using hybrid composites in regions that are sized using discrete source damage requirements. As expected, the hybrid laminate had marginal improvements over the standard-ply laminate for compression with a notch present. However, contrary to expectations, the hybrid laminate exhibited about 20% lower strength than the standard-ply laminate counterpart for tension with a notch present, likely due to the more brittle failure mode exhibited by the hybrid laminate. In addition, the hybrid specimens exhibited a brittle, self-similar crack, which differs from the standard-ply specimens that contained significant amounts of delamination and fiber splitting prior to ultimate failure. Additional investigation is required to assess the use of hybrid laminates for areas containing discrete source damage, and their effect on weight of such regions. Therefore, it appears that the hybrid thin-ply laminates can provide weight benefit in terms of category 1-3 damage as defined in the FAA AC 20-108B [11], but further investigation is required to assess the benefits of hybrid thin-ply laminate for category 4 and 5 damage conditions.

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Table 1. CAI test specimen laminate information.

Laminate Type	Stacking Sequence
Standard-ply Quasi	$[45/0/-45/90]_{4s}$
Hybrid Quasi	$[\pm45/(\pm45/0/\mp45/90)_4/\pm45/\pm45/\mp45/\mp45/(90/\pm45/0/\mp45)_4/\mp45]$
Standard-ply Hard	$[45/0/90/0/-45/0/45/0/-45/0]_{2s}$
Hybrid Hard	$[\pm45/0/\mp45/90/\pm45/0/\mp45/0/\pm45/0/\mp45/0/\pm45/0/90/0/\mp45/0/\pm45/0/\mp45/0/\pm45]_s$

NOTE 1: All plies in the standard-ply laminates and all 0- and 90-degree plies in the hybrid laminate are 190 gsm.
NOTE 2: All 45-degree and -45-degree plies in the hybrid laminates are 70 gsm plies.

Table 2. Percent differences of standard-ply and hybrid thin-ply CAI test specimen properties and stiffness.

Panel Designation	Compared To	Equivalent Engineering Properties, % Difference		A Matrix, % Difference		D Matrix, % Difference	
Hybrid Quasi	Standard-ply Quasi	E_x	-0.810	A11	1.138	D11	4.178
		E_y	-0.810	A12	2.901	D12	3.682
		ν_{xy}	1.743	A22	1.138	D22	6.324
		ν_{yx}	1.743	A66	2.807	D66	3.752
		G_{xy}	1.225				
Hybrid Hard	Standard-ply Hard	E_x	-0.925	A11	0.603	D11	0.063
		E_y	-0.022	A12	2.800	D12	12.668
		ν_{xy}	1.262	A22	1.519	D22	4.738
		ν_{yx}	2.184	A66	2.670	D66	11.927
		G_{xy}	1.403				

Table 3. CAI panel impact survey results.

Panel Designation	Specimen ID	Thickness (in)	Impact Energy (ft-lb)	Dent Depth (in)	Damage Area (in^2)
Hybrid Quasi	HQ-IS-1	0.2449	80	0.2775	5.056
	HQ-IS-2	0.2452	60	0.1260	4.398
	HQ-IS-3	0.2436	50	0.0720	5.608
	HQ-IS-4	0.2435	40	0.0350	5.844
Standard-ply Quasi	SQ-IS-1	0.2408	80	0.0965	9.987
	SQ-IS-2	0.2401	60	0.0620	9.070
	SQ-IS-3	0.2398	50	0.0455	10.502
	SQ-IS-4	0.2388	40	0.0165	8.327
Hybrid Hard	HH-IS-1	0.3063	80	0.0425	12.555
	HH-IS-2	0.3061	60	0.0275	8.304
	HH-IS-3	0.3054	50	0.0150	7.194
	HH-IS-4	0.3055	40	0.0110	6.155
Standard-ply Hard	SH-IS-1	0.3004	80	0.0600	13.987
	SH-IS-2	0.3010	60	0.0305	11.553
	SH-IS-3	0.3002	50	0.0165	11.595
	SH-IS-4	0.3014	40	0.0155	8.608

Table 4. CAI specimen test results.

Panel Designation	Specimen ID	Thickness (in)	Dent Depth (in)	Damage Area (in^2)	Strength (ksi)
Hybrid Quasi	HQ-CAI-1	0.2446	0.0230	6.441	29.589
	HQ-CAI-2	0.2443	0.0300	6.618	28.180
	HQ-CAI-3	0.2433	0.0345	6.426	27.716
	HQ-CAI-4	0.2451	0.0335	6.230	27.219
Standard-ply Quasi	SQ-CAI-1	0.2390	0.0145	8.046	21.298
	SQ-CAI-2	0.2399	0.0160	8.256	21.341
	SQ-CAI-3	0.2409	0.0160	8.899	20.552
	SQ-CAI-4	0.2408	0.0140	8.669	19.323
Hybrid Hard	HH-CAI-1	0.3071	0.0195	7.038	39.493
	HH-CAI-2	0.3060	0.0155	8.576	39.818
	HH-CAI-3	0.3053	0.0190	10.084	39.534
	HH-CAI-4	0.3062	0.0150	8.271	40.331
Standard-ply Hard	SH-CAI-1	0.3000	0.0170	9.733	34.180
	SH-CAI-2	0.2996	0.0190	11.674	35.720
	SH-CAI-3	0.3005	0.0185	10.022	34.162
	SH-CAI-4	0.3000	0.0175	9.708	36.159

Table 5. Notched test specimen laminate information.

Test Specimen	Laminate
Standard-ply	$[45/90/\text{-}45/0_2/45/90/\text{-}45/0]_s$
Hybrid	$[\pm45/90/\mp45/0/\pm45/0/\pm45/90/\mp45/0/\pm45_{(t30)}]_s$

NOTE 1: All plies in the standard-ply laminate and all 0- and 90-degree plies in the hybrid laminate are 190 gsm.
NOTE 2: All 45-degree and -45-degree plies in the hybrid laminate are 70 gsm, except for the hybrid hard laminate where the "(t30)" subscript for the mid-laminate plies indicates 30 gsm material was used to maintain symmetry and total thickness.

Table 6. Comparison of standard-ply and hybrid thin-ply notched test specimen properties and stiffness.

Panel Designation	Compared To	Equivalent Engineering Properties, % Difference		A Matrix, % Difference		D Matrix, % Difference	
Hybrid Notched	Standard-ply (ACP) Notched	E_x	-0.823	A11	0.874	D11	5.490
		E_y	-0.582	A12	2.850	D12	-3.873
		ν_{xy}	1.712	A22	1.119	D22	8.565
		ν_{yx}	1.958	A66	2.737	D66	-3.343
		G_{xy}	1.329				

Table 7. Notched test specimen definitions.

Specimen	Overall Dimensions (in)	Notch Description	Replicates
Small Compression	4 by 12	0.8 inches long, 0.25 inches wide, 0.25-inch diameter notch tips	3
Small Tension	4 by 16	0.8 inches long, 0.25 inches wide, 0.25-inch diameter notch tips	3
Intermediate Tension	20 by 70	4.0 inches long, 0.25 inches wide, 0.25-inch diameter notch tips	2

Table 8. Notched small specimen average strength (ksi).

Specimen	Standard-ply	Hybrid	% Diff.
Compression	34.05	34.65	1.76
Tension	63.80	50.88	-20.2

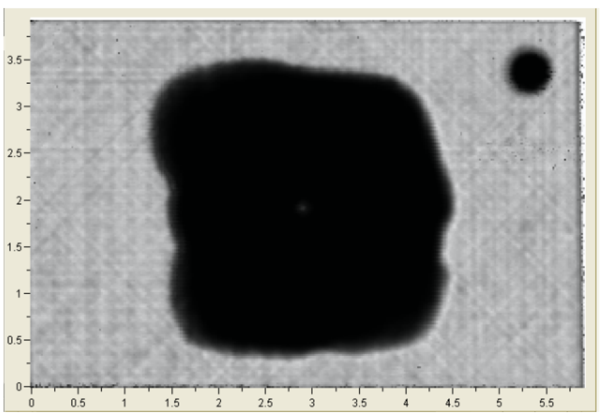
a) Standard-ply Quasi (40 ft-lb)

b) Hybrid Quasi (40 ft-lb)

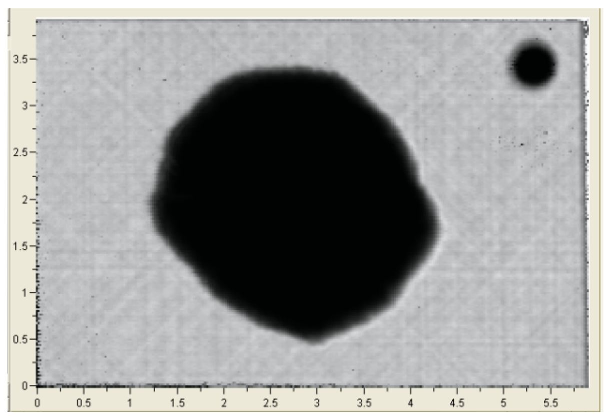
c) Standard-ply Hard (50 ft-lb)

d) Hybrid Hard (50 ft-lb)

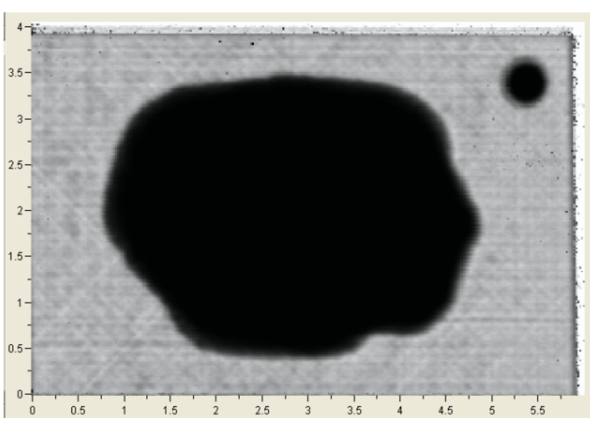
Figure 1. Typical impact site damage of CAI test specimens.



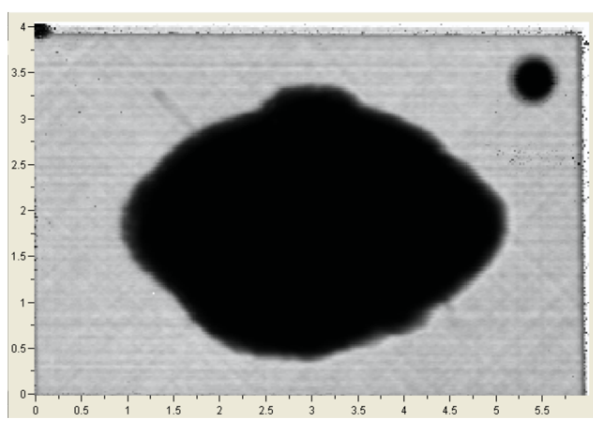
a) Standard-ply Quasi (40 ft-lb)



b) Hybrid Quasi (40 ft-lb)

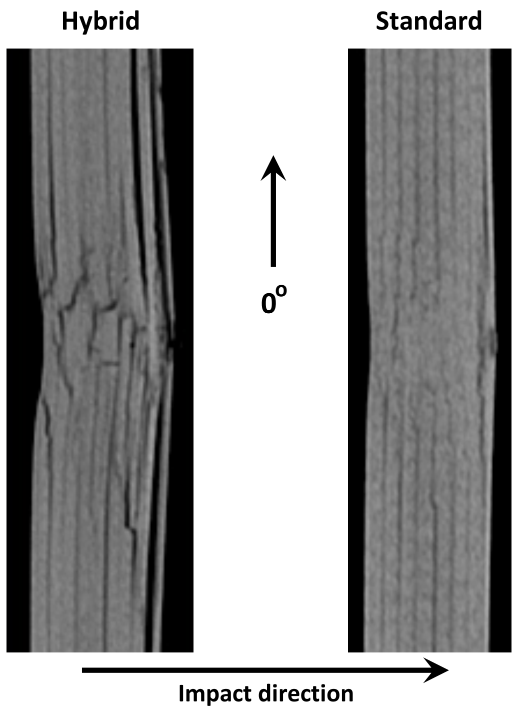


c) Standard-ply Hard (50 ft-lb)

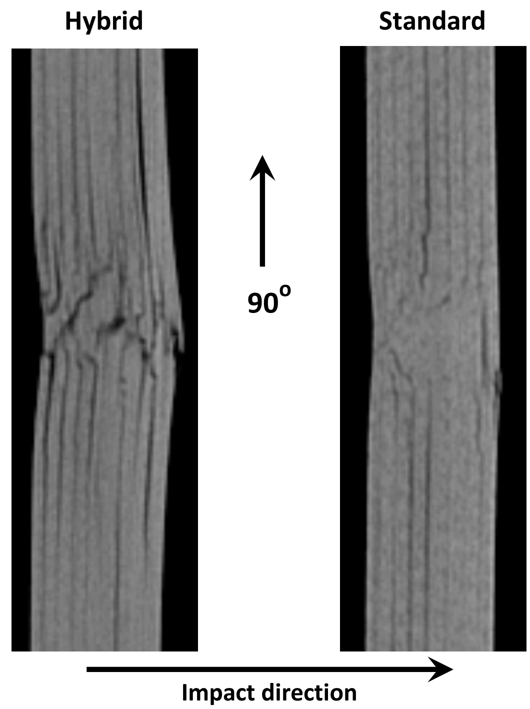


d) Hybrid Hard (50 ft-lb)

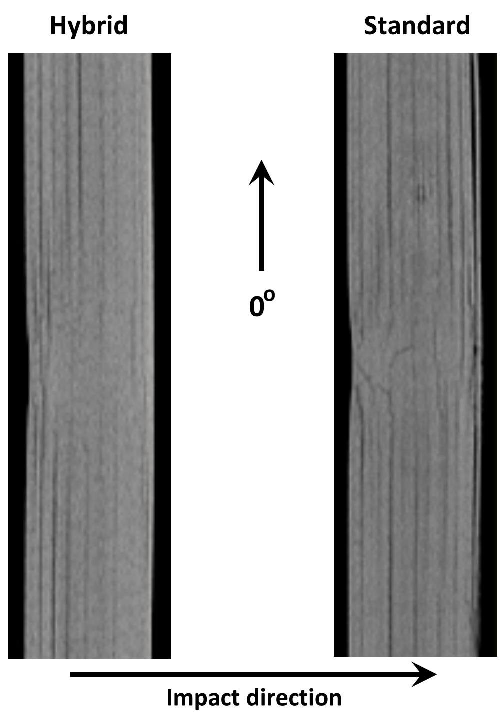
Figure 2. Typical C-scans of CAI test specimens comparing damage area. (Scales shown in inches)



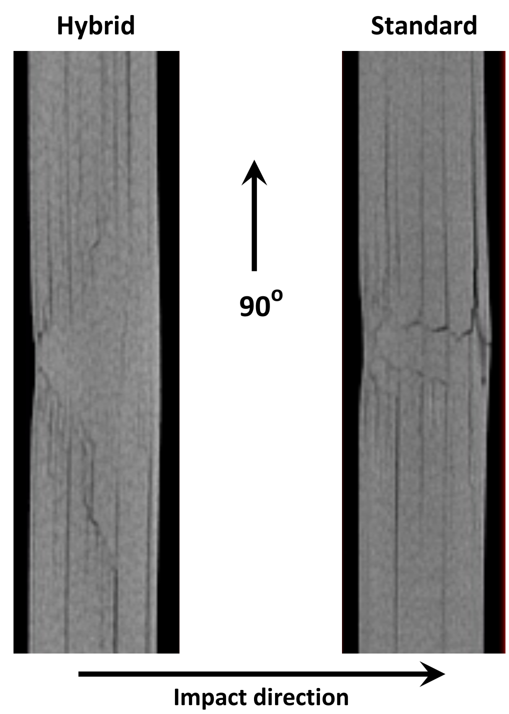
a) Quasi 0° direction (40 ft-lb)



b) Quasi 90° direction (40 ft-lb)



c) Hard 0° direction (50 ft-lb)



d) Hard 90° direction (50 ft-lb)

Figure 3. Typical through thickness damage CT-scans of CAI test specimens, cut through impact location. (Direction angle indicates alignment with ply fiber direction, e.g., 0° aligned with 0° fiber)

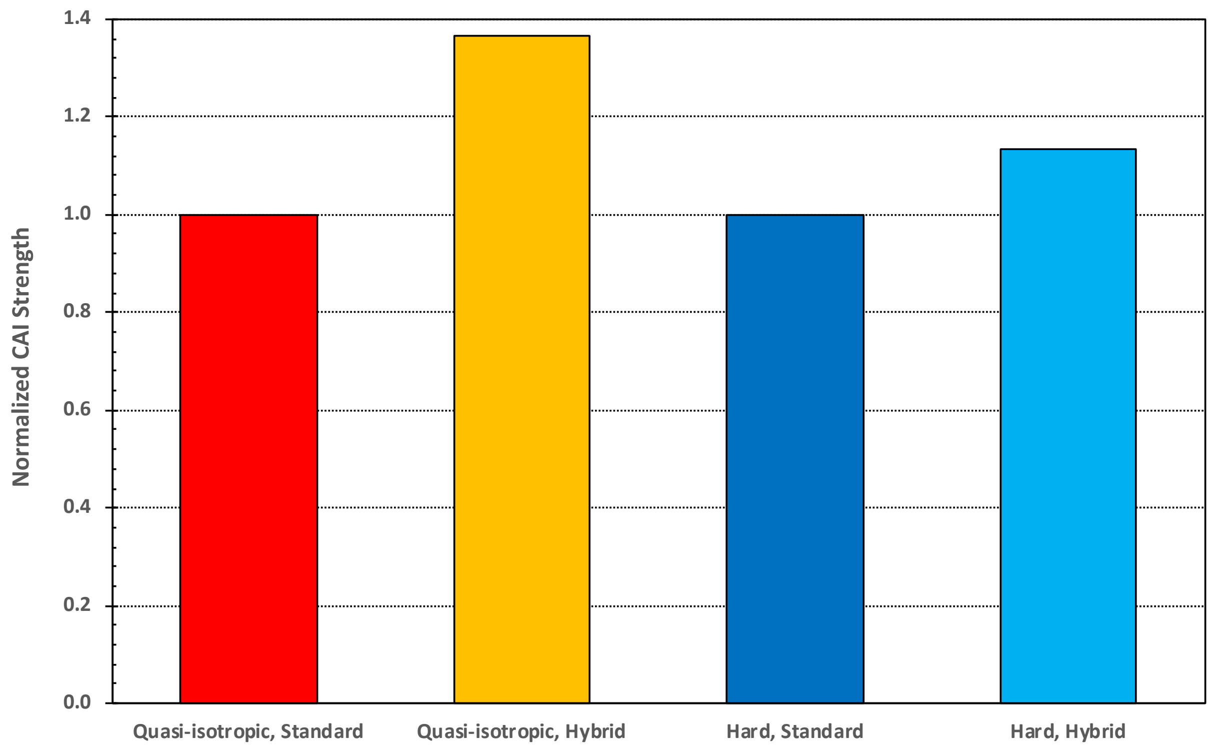


Figure 4. Comparison of normalized average CAI strength for hybrid and standard-ply laminates.

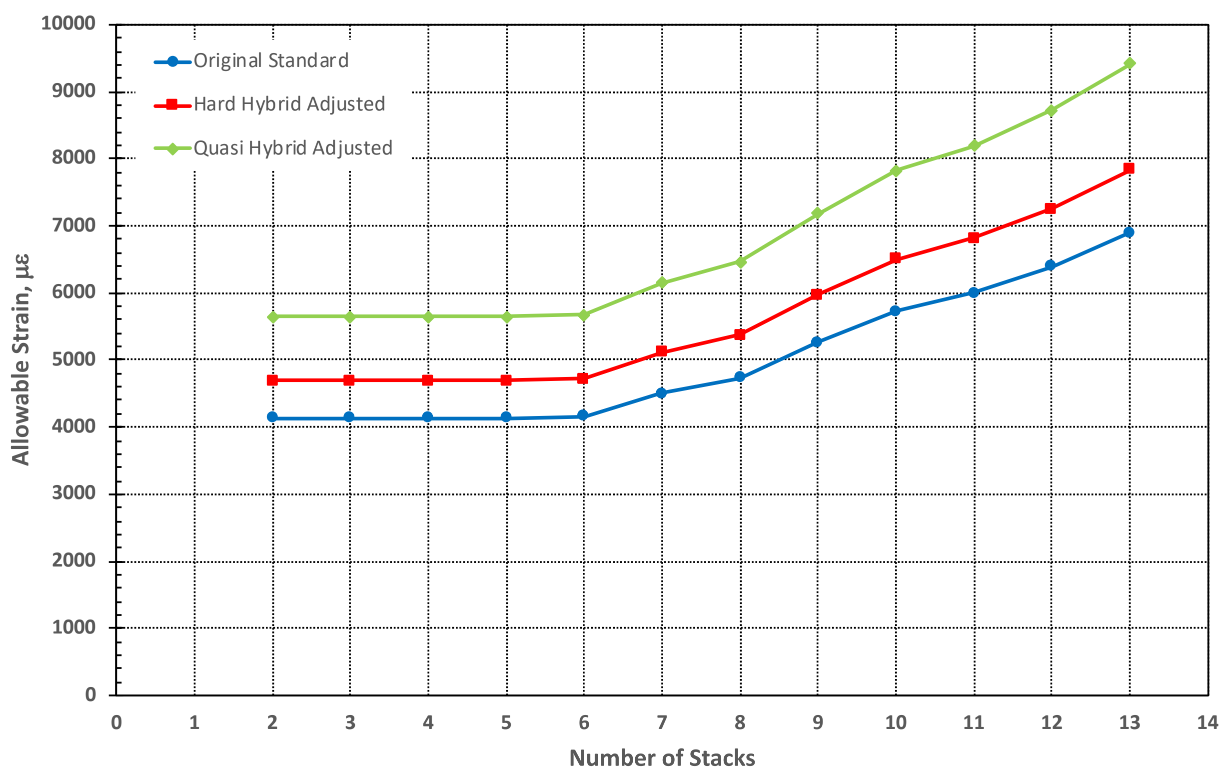


Figure 5. CAI allowable design strains for the original semi-span, and adjusted values incorporating increased performance percentage due to hybrid laminates.

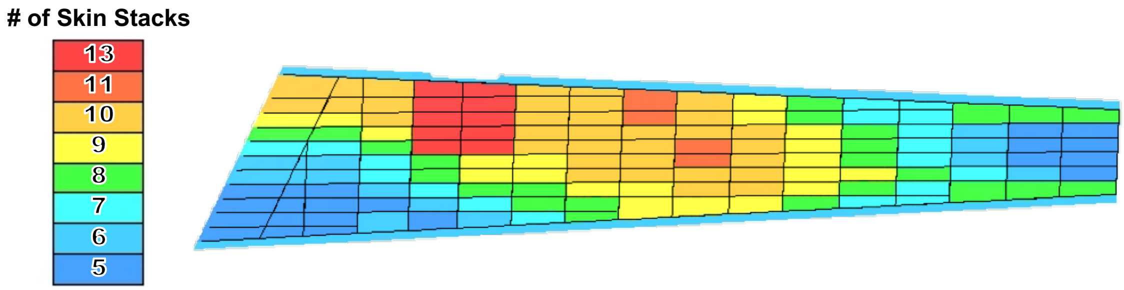


Figure 6. Semi-span upper cover skin design. Lines indicate locations of ribs, stringers and spars.

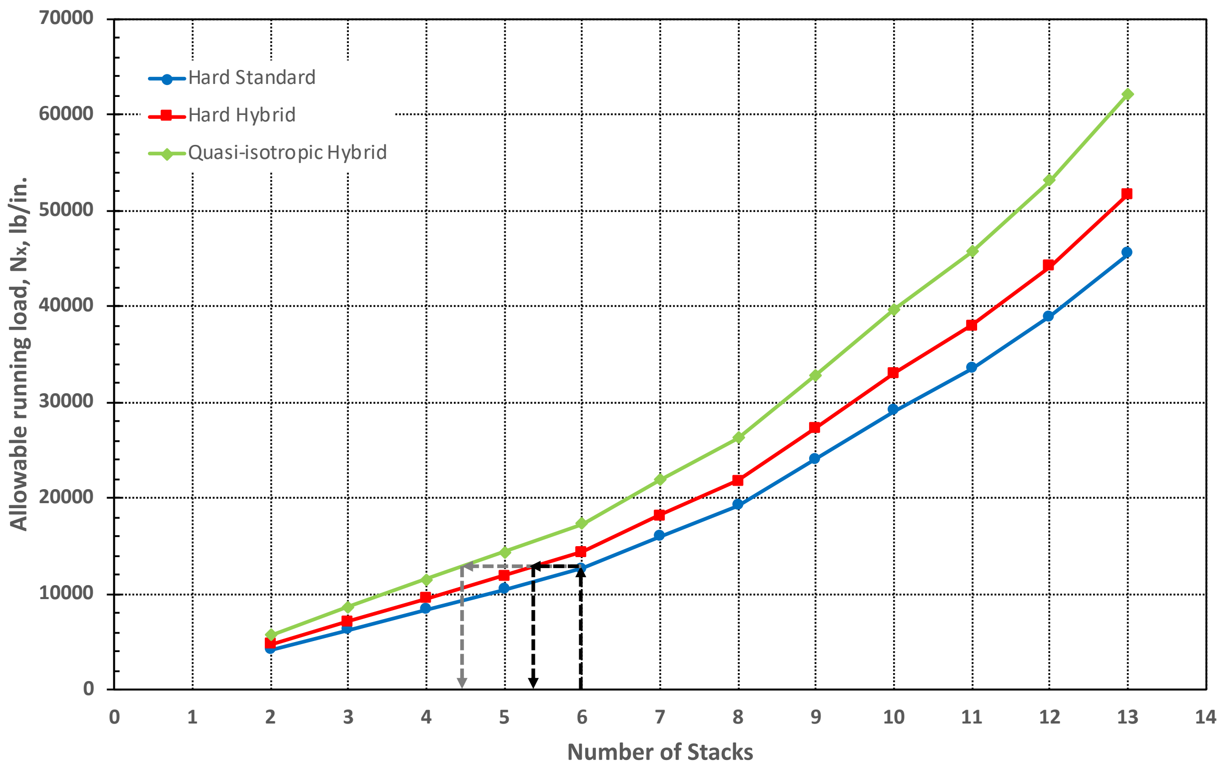


Figure 7. CAI allowable design running loads, N_x, for the original semi-span, and adjusted values incorporating increased performance percentage due to hybrid laminates, by number of stacks.

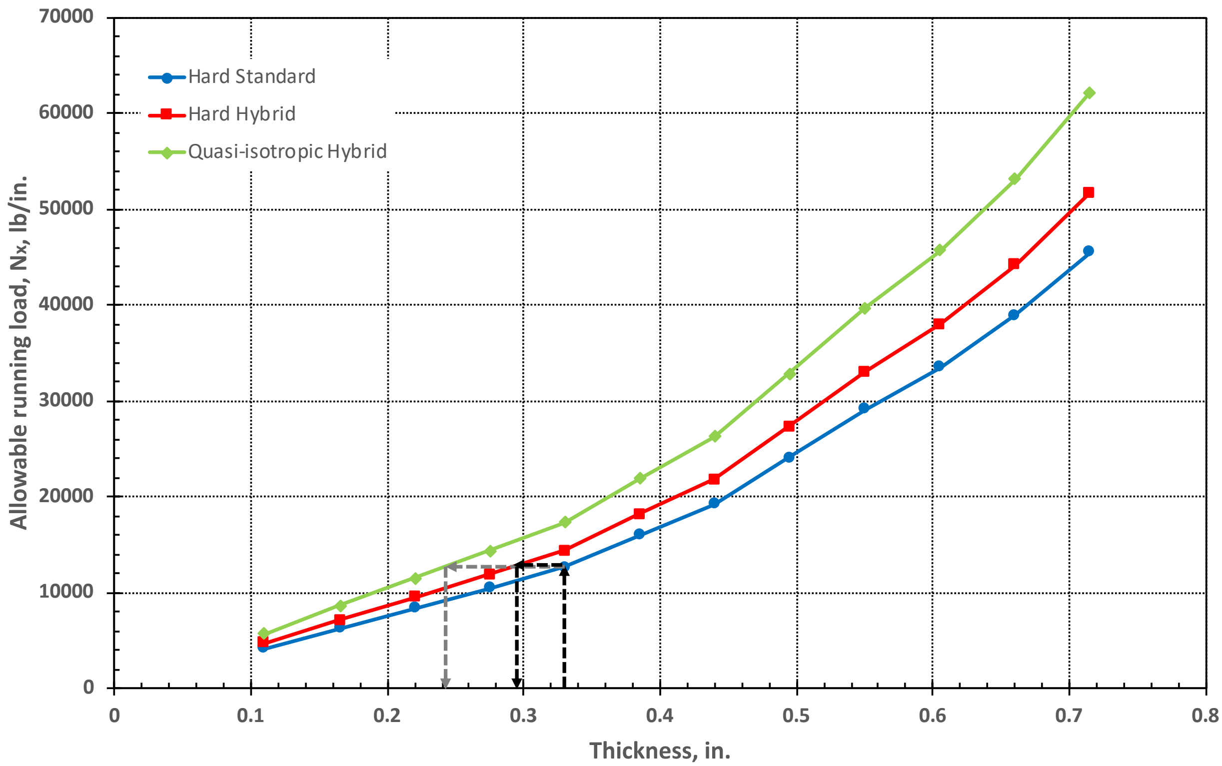
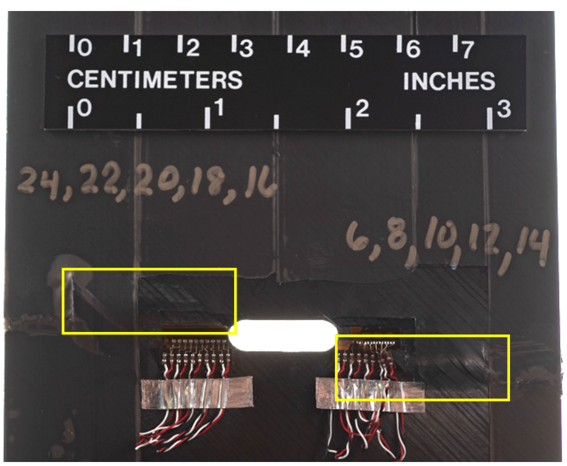
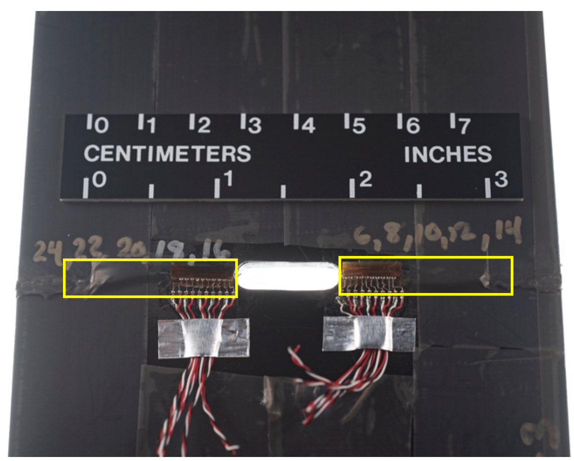


Figure 8. CAI allowable design running loads, N_x, for the original semi-span, and adjusted values incorporating increased performance percentage due to hybrid laminates, by laminate thickness.

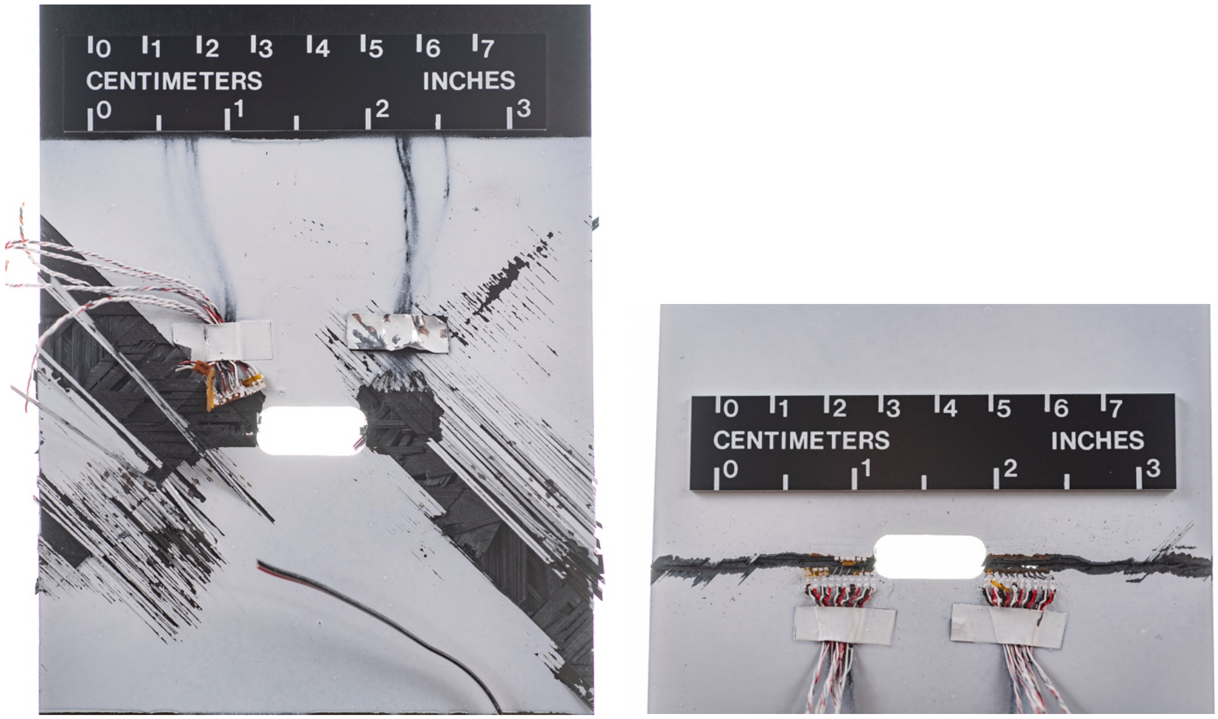


a) Standard-ply



b) Hybrid

Figure 9. Typical small compression specimen failures.



a) Standard-ply b) Hybrid

Figure 10. Typical small tension specimen failures.

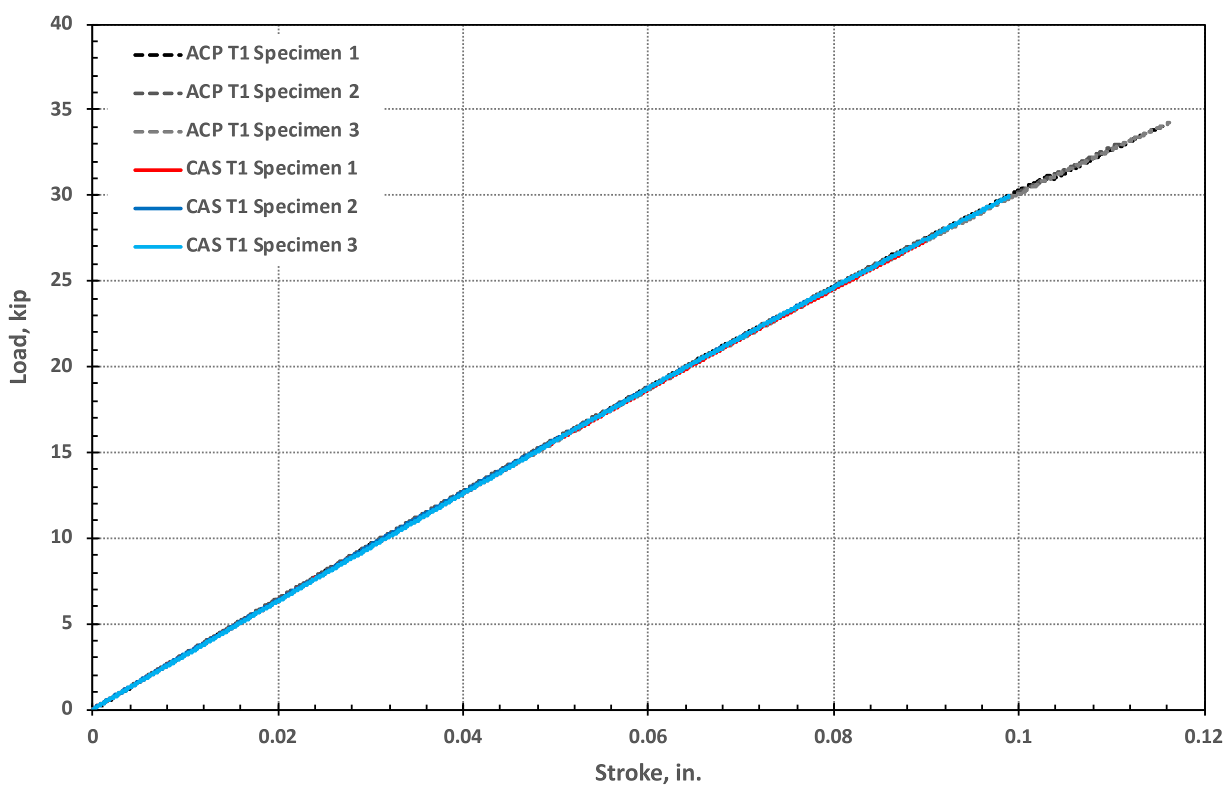
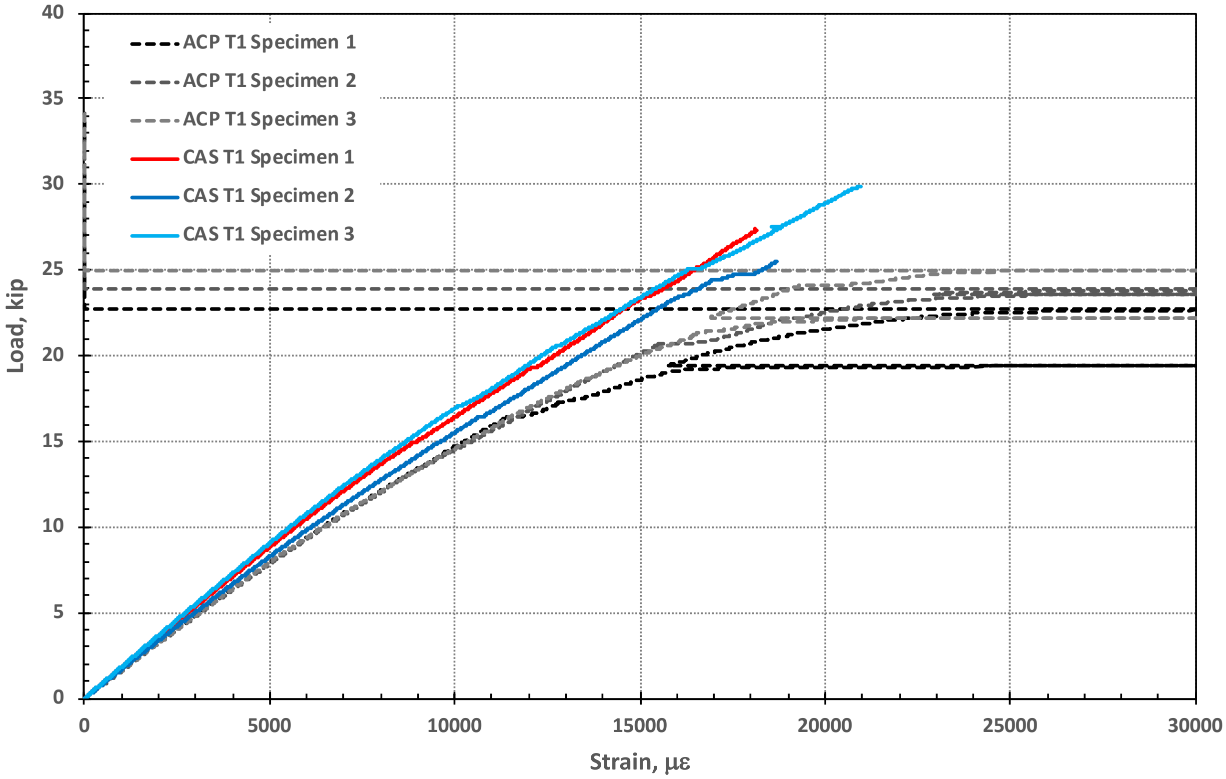
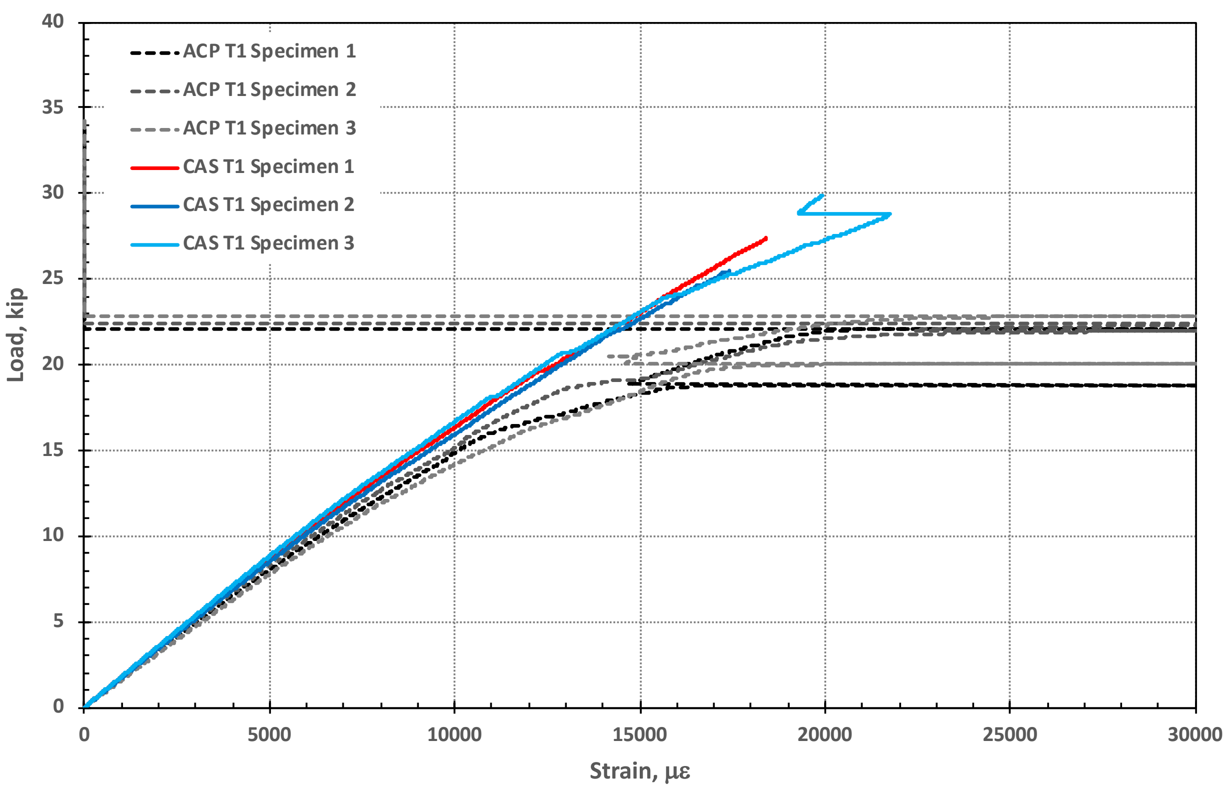


Figure 11. Load vs. stroke for the six small notched tension specimens.



a) First notch tip



b) Second notch tip

Figure 12. Strains adjacent to the notch tips in the small tension specimens.