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Durability and Damage Tolerance of Large Composite Primary Aircraft Structure (LCPAS)

John E. McCarty and William G. Roeseler

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John E. McCarty and William G. Roeseler
Boeing Commercial Airplane Company
Seattle, Washington

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DURABILITY AND DAMAGE TOLERANCE OF LARGE COMPOSITE PRIMARY AIRCRAFT STRUCTURE (LCPAS)

John E. McCarty
Chief, Structures Technology

William G. Roeseler
Professional Engineer
Advanced Composite Development Program

Boeing Commercial Airplane Company
Seattle, Washington

ABSTRACT

Analysis and testing that addressed the key technology areas of durability and damage tolerance were completed for wing surface panels. The wing of a fuel-efficient, 200-passenger commercial transport airplane for 1990 delivery was sized using graphite-epoxy materials. Coupons of various layups used in the wing sizing were tested in tension, compression, and spectrum fatigue with typical fastener penetrations. The compression strength after barely visible impact damage was determined from coupon and structural element tests. One current material system and one toughened system were evaluated by coupon testing. The results of the coupon and element tests were used to design three distinctly different compression panels meeting the strength, stiffness, and damage-tolerance requirements of the upper wing panels. These three concepts were tested with various amounts of damage ranging from barely visible impact to through-penetration. The results of this program provide the key technology data required to assess the durability and damage-tolerance capability of advanced composites for use in commercial aircraft wing panel structure.

NOMENCLATURE

A	area (in.^{2})
deg	degree
E	Young's modulus (Msi)
f	stress (ksi)
ft	foot
G	shear modulus (Msi)
Gt/Nx	nondimensional shear stiffness parameter
in.	inch
kip	1000 lb
ksi	1000 lb/in.^{2}
lb	pound
lb/in.^{3}	pound per cubic inch
Msi	1,000,000 lb/in.^{2}
Nx	end load (kip/in.)
P	skin load including embedded plies (kip)
$\overline{P}$	total panel load (kip)
t	thickness (in.)
$\overline{t}$	smeared thickness (in.)
(xx/yy/zz)	fiber percentage in the laminate (0 deg/45 deg/90 deg)
δ	deflection (in.)
[ ]	standard laminate orientation code

INTRODUCTION

The design of an advanced composite commercial aircraft wingbox requires information from several key technology areas. In addition, a data base is required in each of these areas to adequately define the total wingbox development program. The objective of the Large Composite Primary Aircraft Structure (LCPAS) program was to address two of these key technology areas, durability and damage tolerance. Because the wing surface panels for a commercial transport airplane typically represent 65 to 70% of the wingbox mass, all design parameters affecting the structural efficiency of wing panels are of primary importance in establishing the mass of the box.

The notch and impact sensitivity of graphite-epoxy material has led to the suppression of the usable design strain level in most current applications.The industry typically has used design strain levels of 0.005 for tension and 0.004 for compression (0.0045 average). These reduced strain levels provide a built-in reduction in the damage sensitivity of composites and also provide good fatigue durability. This has been documented by Johnson, McCarty, and Wilson (ref. 1) and others.

The efficient use of graphite-epoxy in commercial aircraft wing structure requires a significant mass reduction from current and future aluminum alloy wing designs. A mass reduction of 23%for the wingbox is needed for cost-effective commercial aircraft applications. Because there are many fasteners, metal fittings, and joints that cannot be replaced with composites, a mass reduction of 30% is required on the wing surface panels.

This program was organized to develop the key technology requirements and meet the mass reduction goals.

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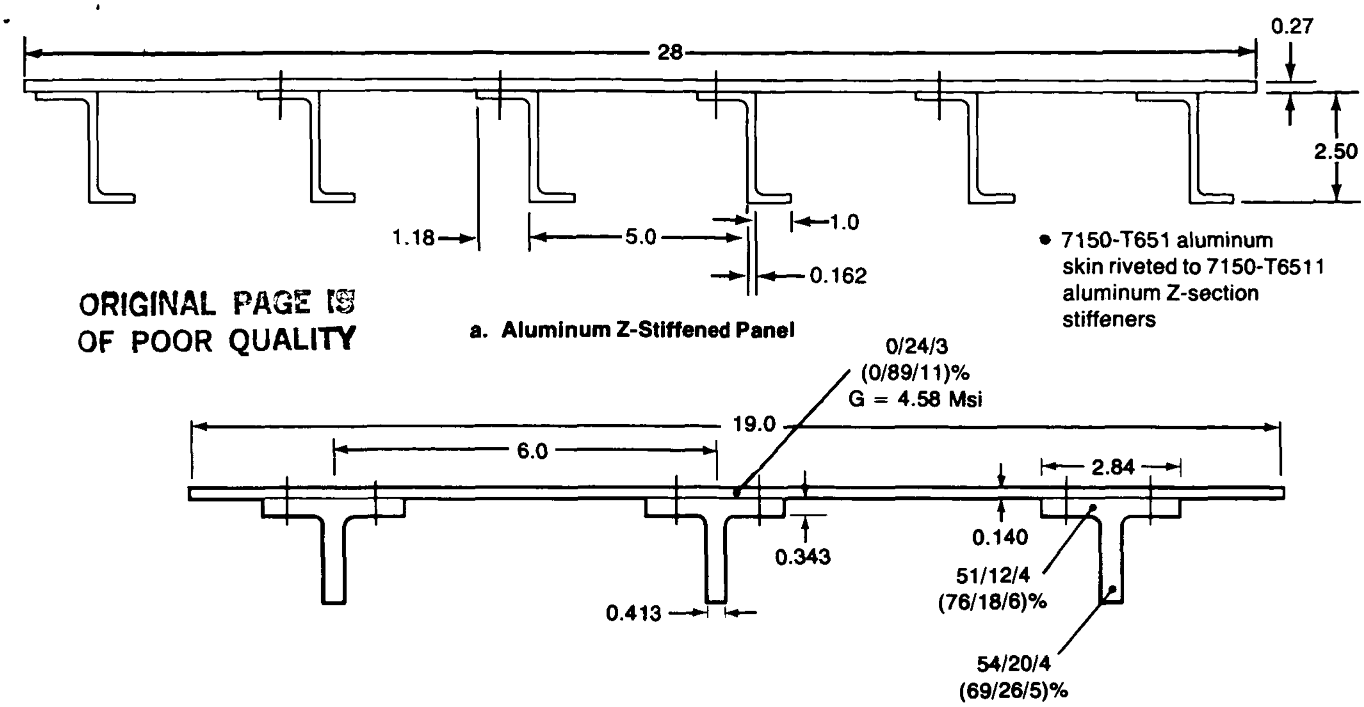
PROGRAM APPROACH

A limited preliminary design was made of a 200-passenger commercial transport wing. The planform of the wing was taken from reference 2. The strength, stiffness, and fatigue requirements for this wing were established under this contract, and a theoretical mass distribution was available for the aluminum wing. The most heavily loaded portion of the wing was selected for study because this area would require high strength as well as stiffness. Considering the high stress levels in the aluminum panels in this portion of the wing, it can be shown that composite panels must work to 48-ksi average at ultimate load to achieve the required 30% mass reduction mentioned earlier. For the graphite-epoxy panels to meet a 48-ksi strength requirement at an average strain level of 0.0045, the average modulus would have to be 10.6 Msi, which requires a high percentage of fibers aligned with the primary load direction. The resulting 0-deg-dominated panels may have inadequate damage tolerance, especially when one of the 0-deg-dominated load paths is destroyed. Also, the shear stiffness requirements of the high-aspect-ratio wing are difficult to meet with an overall modulus of 10.6 Msi. Therefore, a target design value strain of 0.006 was selected because this would yield 48-ksi average stress and 30% mass reduction with a more achievable modulus of 8 Msi.

Because current-technology material systems generally cannot meet 0.006 strain levels after impact damage, a material improvement was required to meet the objectives of LCPAS. Several toughened systems were evaluated before selecting AS4/X2220-3 (Hercules) for the LCPAS key technology program. In general, any improvement in toughness over the baseline system (T300/5208) (Union Carbide/Narmco) was accompanied by a reduction in the hot/wet compression strength. Resin systems like BP907 are much tougher, as reported by Beyers (ref. 3) and others, but their hot/wet compression strengths are unacceptable for compression panels. Because tension is less of a problem than compression, especially with available high-strength fibers, the LCPAS key technology program focused on compression.



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b. Graphite/Epoxy Panel with Soft Skin and Mechanical Fasteners

• All dimensions in inches

Figure 1. Prior Technology Compression Panels

To focus on the appropriate level of damage tolerance and structural efficiency, the baseline aluminum test panels from the current commercial programs were used. Figure 1a represents the most relevant panel. It has thick skins providing a level of shear stiffness appropriate for the future high-aspect-ratio wing. The skin carries 65% of the total load. In test this panel actually carried 833 kips, 29 kips/in. at 70 ksi (Gt/Nx = 37). Aluminum panels are critical for long-column effects (buckling/crippling/Johnson/Euler interaction), and the graphite-epoxy panels are critical for impact damage.

The prior NASA and industry test data on composite compression panels included the damage-tolerant panels of reference 4, which were noteworthy because of the high levels of structural efficiency and damage tolerance demonstrated in test. These panels were designed to meet end load and stiffness requirements of current commercial transport aircraft and to provide damage tolerance with respect to external skin damage. The strongest panel, illustrated in figure 1b, carried 509 kips, 28 kips/in. at 59 ksi (Gt/Nx = 23) after impact. This panel had several damage-tolerant features, including soft skins, closely spaced stiffeners, and mechanical fasteners. Although the 0.14-in.-thick skins, laid up (0/89/11)%, had enough shear stiffness (Gt = 640 kips/in.) for most areas of current aircraft, the LCPAS study section required almost twice as much shear stiffness (Gt = 1200 kips/in.).

The LCPAS key technology progam was structured to incorporate the proven damage-tolerant features of these prior aluminum and composite panels and to extend the data base to end loads of 30 kips/in. and shear stiffness requirements of 1200 kips/in.

The durability and damage tolerance of the potential design configuration had to be assessed before the wing mass reduction could be confirmed. For tension-loaded structure, the finite-element/fracture mechanics approach reported by Porter and Pierre (ref. 5) and others has correlated well with test data. A similar analysis procedure does not exist for impact-damaged compression panels, except possibly for through-penetration damage. Hence, testing was the primary method used to assess the structural efficiency, durability, and damage tolerance of the selected designs.

The results of the panel test program were used to redesign the wing cover panels, and a theoretical wing mass reduction was established for the final selected design. Substantiating data for the tension panels were limited to the coupons with fastener penetrations.

The airplane shown in figure 2 is a fuel-efficient, 200-passenger airplane with a high-aspect-ratio wing. Internal loads and stiffness requirements for the wing, developed in the previous NASA contracts (ref. 2), are shown in figure 3. The area of the wing selected for study on this program was the upper wing panel at the nacelle, where end loads approach 30 kips/in. and the shear stiffness requirement is Gt = 1200 kips/in. (Gt/Nx = 40).

PASSENGERS MIXED CLASS ALL TOURIST	 197 207
ENGINES	TWO CF6-6D2
WEIGHTS (lb) TOGW OEW	 268 040 176 120
ASPECT RATIO	12.0

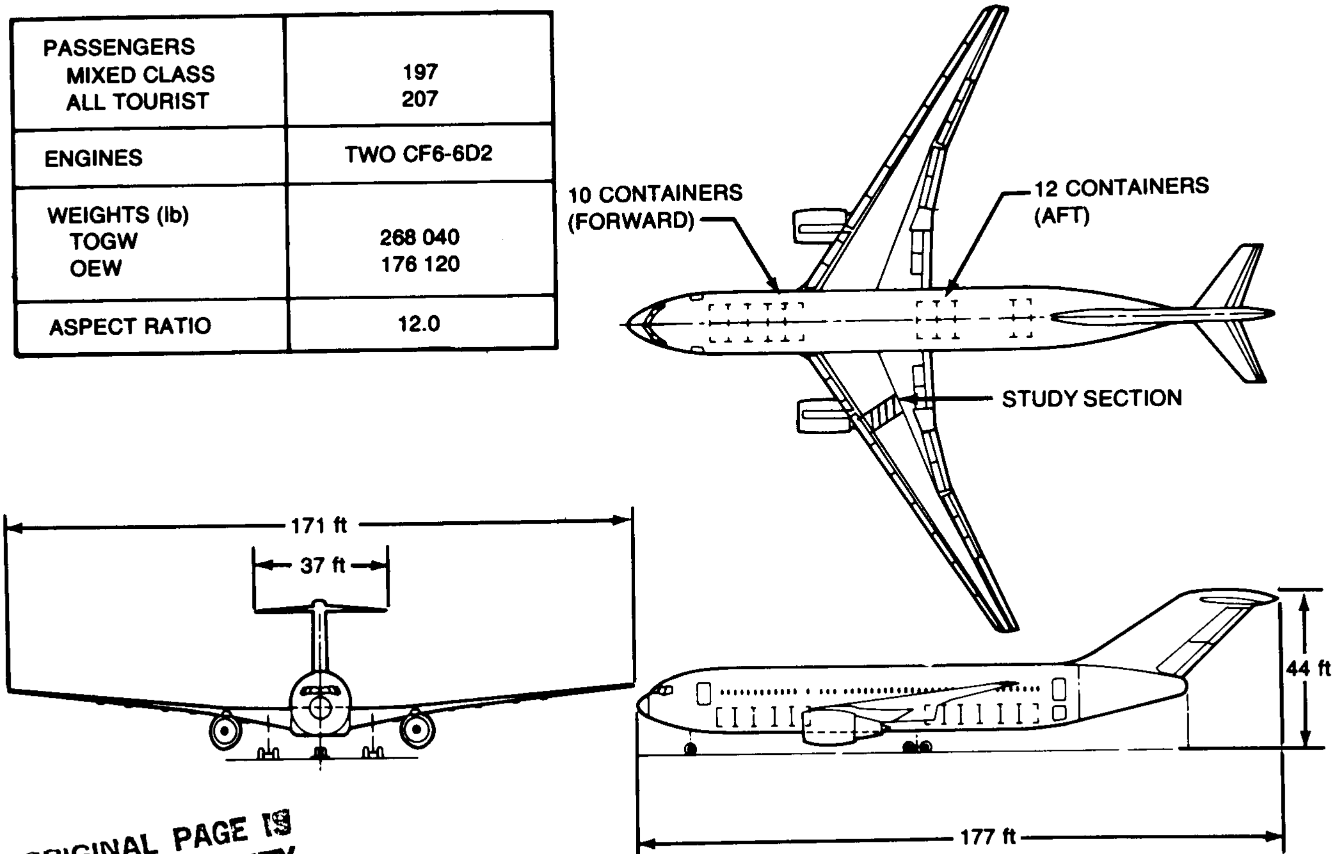

Figure 2. Reference Airplane

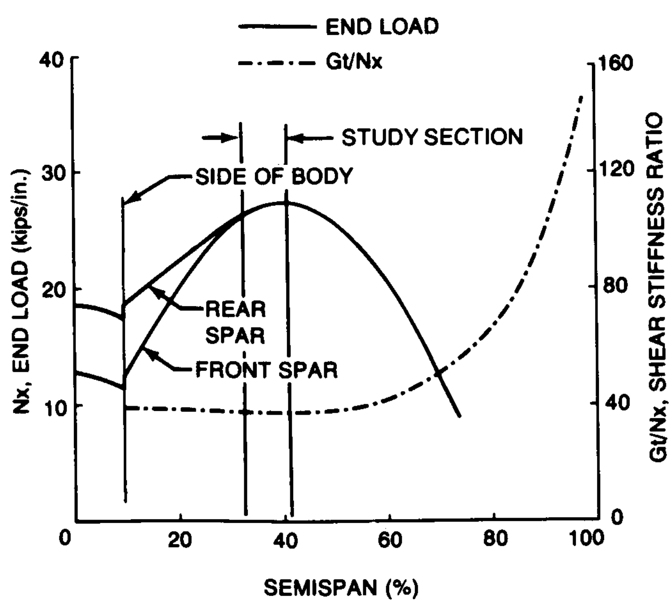

Figure 3. Internal Loads and Stiffness Requirements

CRITERIA

Design criteria established by the FAA and by Boeing require maintenance of a level of safety equivalent to that of the aluminum structure used on current large transport airplanes. This means that the composite structure must carry ultimate loads with manufacturing defects and processing anomalies such as nicks, scratches, gouges, delaminations, impact damage, and ply splices up to the size and severity allowed by the processing and assembly specifications. For example, delaminations up to 0.5 in. are allowed, provided no more than one such delamination exists in any 12- by 12-in. area. This includes delaminations caused by accidental impact that may occur during the manufacturing process.

Another requirement for composite structures, should in-service damage occur, is that the remaining structure will carry limit loads until the damage is detected by the planned maintenance program. For most of the wing structure, this level of damage should be visually detectable from some distance. The through-penetration impact assessed during the current program is an exploratory level of damage addressing this requirement.

STRUCTURAL CONCEPTS

Considering the strength, stiffness, and damage-tolerance requirements, a series of preliminary designs was completed for the upper and lower wing panels. The design studies considered various damage-tolerance features such as discrete stiffeners, mechanical fasteners, soft skins, wraps of soft material around the 0-deg-dominated areas, and closely spaced stiffeners. Graphite-epoxy experience indicates that more technology development is required for damage tolerance of the upper panel. The detailed designs and the test program were therefore oriented toward the upper surface panels. The three upper panel designs shown in figure 4 were selected for fabrication and test and represent three distinctly different concepts. All were designed to carry 30 kips/in. of compression load at 0.006 strain while providing a shear stiffness of 1200 kips/in. ($Gt/Nx = 40$).

The I-stiffened concept represents the most optimistic of the three, having a substantial percentage of 0-deg material in all three elements and a target ultimate stress of 58 ksi. The damage-tolerance features of the I-stiffened panel include: (1) stable stiffeners with excess Euler buckling capability to better carry load around damaged areas; (2) discrete stiffeners capable of being mechanically attached to the skins in case of bondline weakness; and (3) uniform channels permitting good compaction of plies in the radii and facilitating automated manufacture. The target mass reduction for the I-stiffened panel was 33% compared with the 70-ksi aluminum test panel.

The J-stiffened concept was assessed to have minimum risk of failing the damage-tolerance tests. Damage-tolerance features include: (1) embedded 0-deg material protected from impact damage by 45-deg plies on all sides; (2) wide attachment flanges on the stringers to increase the bond area between the skin and the stringers and to allow for mechanical attachment with 5/16-in.-diameter fasteners; and (3) gradual transition from (19/62/19)% skin to (56/33/11)% cap to provide minimum Poisson ratio mismatch between adjacent elements. The target stress level for the J-stiffened concept was 52 ksi, providing a 25% mass reduction.

The blade-stiffened concept was estimated to be the lowest cost of the three and provided a mass reduction potential equal to that of the low-risk J concept. Damage-tolerance features of the blade include: (1) soft (10/80/10)% skin, which is believed to have better load redistribution capability (toughness) than the (30/60/10)% skin of the I-stiffened panel or the (19/62/19)% skin of the J-stiffened panel; (2) closely spaced stringers, providing a nearby load path for redistribution of load around damaged areas; and (3) interleaving of the 0-deg and 45-deg material at the base of the blade, providing a redistribution load path between the skin and the stringer that does not rely entirely upon bondline strength.

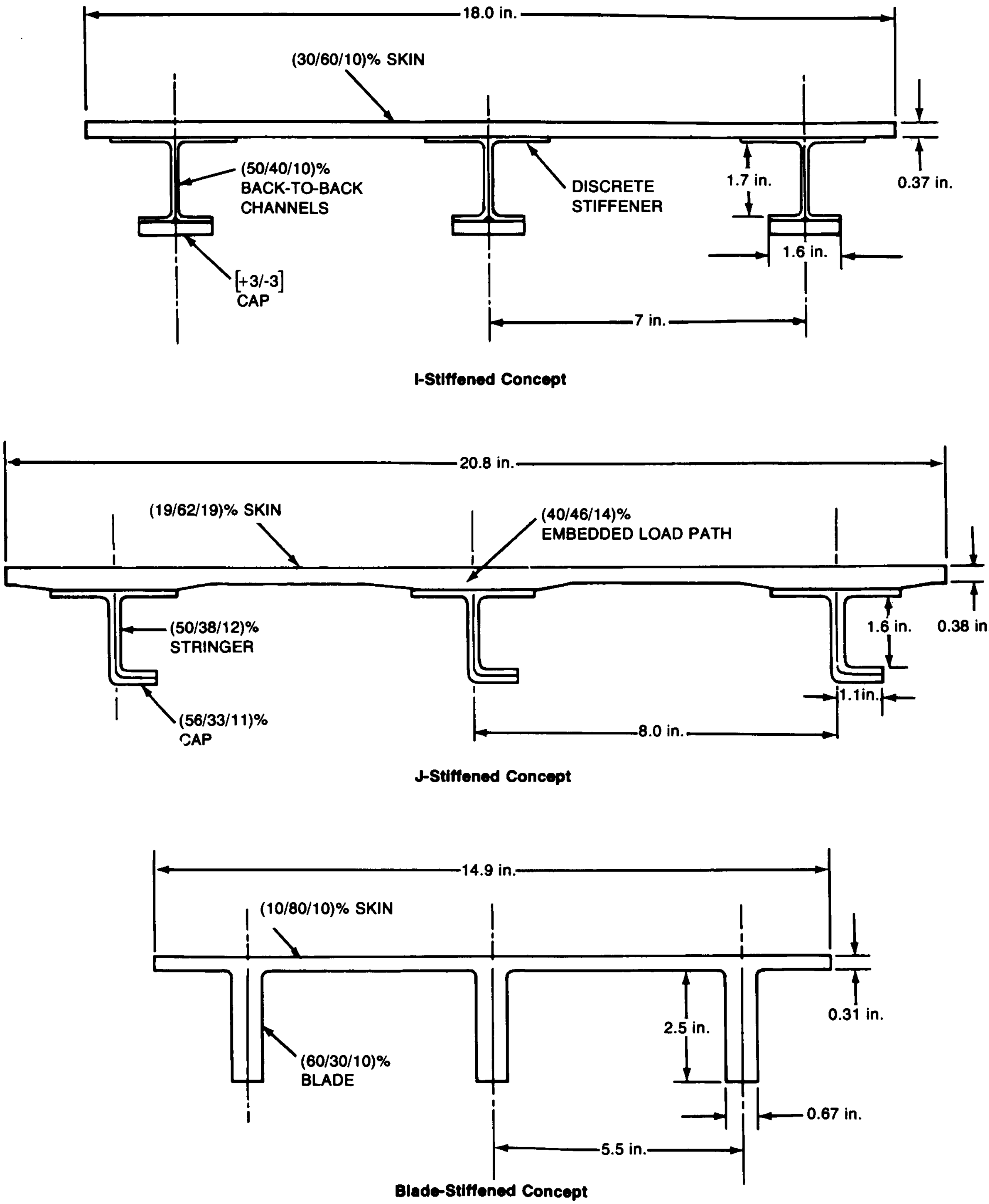

Figure 4. Selected Structural Concepts

COUPON TESTS

Following the preliminary design studies, a coupon test program (fig. 5) was formulated to address the range of laminates appearing in the upper and lower wing panel concepts. Static tension, compression, and fatigue tests were conducted at room temperature on 160 coupons cut from 11 different panels ranging from 24 to 50 plies in thickness (0.178 to 0.37 in.). The unloaded fastener was selected as the basic structural detail for the coupon test program. This provided a local stress concentration considered typical of wing structure. In general, countersunk fasteners were used for the candidate skin layups and protruding-head fasteners were used for the stringer layups. Coupons were tested to compare the current-technology material system (T300/5208) with the selected, toughened system (AS4/X2220-3) used for most of the LCPAS tests. Open-hole coupons and impact coupons per reference 6 also were tested to characterize the AS4/X2220-3 material system relative to other toughened systems selected by the other LCPAS contractors. Layups ranged from the soft, all-45-deg laminate to the stiff [+3/-3]-deg laminate.

Figure 6 summarizes the static tension data from 39 coupons with fastener-filled holes. Test results are tabulated in table 1. For the two layups providing material evaluation data, the AS4/X2220-3 material was 32% to 73% stronger than the T300/5208 material. Of the candidate stringer layups, the [+3/-3]-deg laminate was 65% stronger than the (60/30/10)% laminate. Of the candidate skin layups, in terms of failure strain, the all-45-deg laminate was much better than the (8/84/8)% laminate, and the (8/84/8)% laminate was much better than the isotropic laminate. These large differences in stringer load capability and skin strain to failure provide some guidance for design of damage-tolerant structure. Fiber orientation and material characteristics together can provide strong load paths to carry load around damaged areas, and high strain skins may be effective in redistributing load from the damage site to the adjacent structure.

In general, the coupon tension results support the target design allowables for tension panels. The rib padup tests indicated that a 10% improvement in static tension strength is available with a 20% unsymmetrical padup. Depending on the knockdown factors to be assigned for statistical variation and environmental effects, some local padups may be required on the tension material.

Figure 7 and table 2 summarize the significant compression coupon results. For the AS4/2220-3 material system, the compression strength after 140 in.-lb impact was 18% better than that of T300/5208. Filled-hole compression results are included in table 2 and figure 7. The results show that the filled-hole compression strength is less critical than compression-after-impact strength.

Cyclic loading generally has not been a problem with graphite-epoxy material systems at the current low level of strain application. To assess the effect of cyclic loading in the high-strain range, fatigue tests were run with the range of layups from all-45-deg to essentially unidirectional. The original plan was to test each fatigue coupon to two-part failure after two lifetimes of spectrum fatigue loads. However, there was absolutely no detectable damage on any of the (25/50/25)%, the (25/75/0)%, or the [+5/-5]-deg fatigue coupons after the two lifetimes. Rather than just repeat all of the filled-hole static data, the fatigue load intensity was increased to induce fatigue damage. The loads were increased in 20% increments with 120,000 flights at each level. One tension coupon and one compression coupon were tested in this manner. Failure of both coupons occurred at 240% of nominal load level after 60 lifetimes of cyclic loads. The conclusion from these tests is that fatigue still is not a problem for graphite-epoxy laminates with fastener penetration when ultimate strains are raised to 0.006.

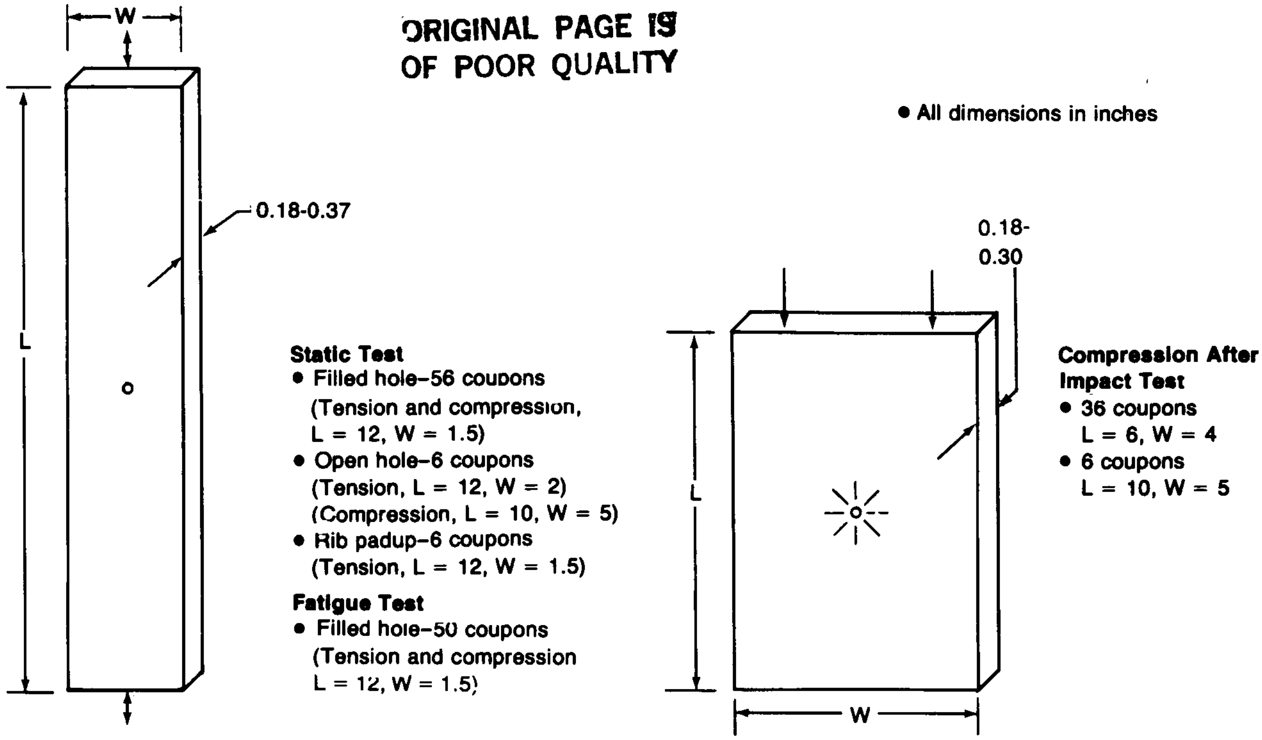

Figure 5. Coupon Program

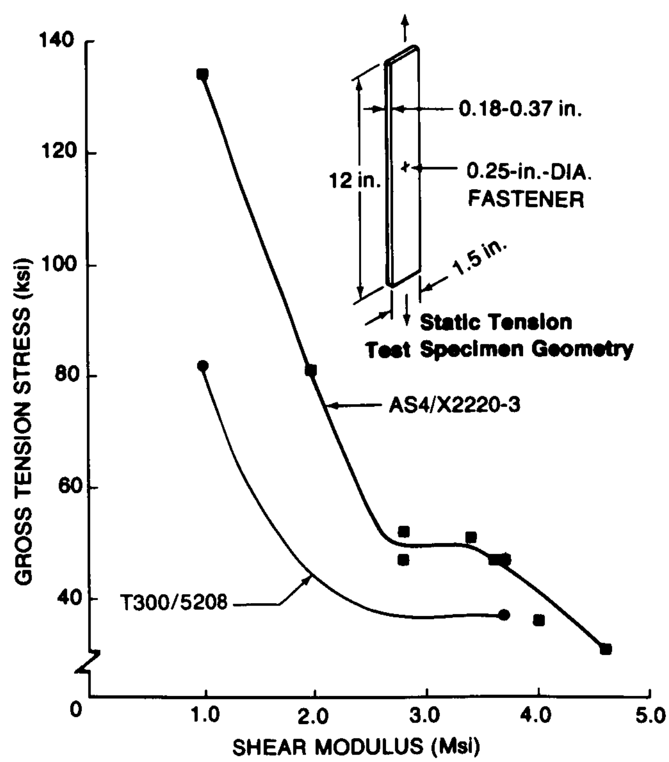

Figure 6. Tension Strength Data

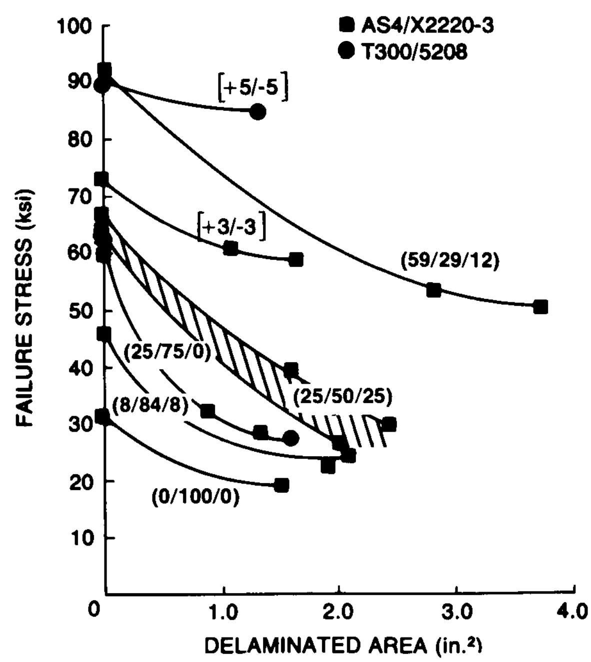

Figure 7. Compression Strength Data

Table 1. Tension Data–Laminates With Holes

LAYUP AS4/X2220-3 EXCEPT AS NOTED	FASTENER-HEAD DETAIL (0.25-in. HOLES)	NUMBER OF PLIES[a]	LOAD (kips)	NET STRESS (ksi)	NOMINAL GROSS STRESS (ksi)	INITIAL EXTENSIONAL MODULUS (Msi)	P/AE NOMINAL STRAIN (in./in.)	δ/L[b] FAILURE STRAIN (in./in.)	SHEAR MODULUS (Msi)
(0/100/0)	Countersunk shear	24	8.22	38	31	2.0	0.0150	0.101	4.6
(8/84/8)	Countersunk shear	24	9.67	46	36	4.3	0.0084	0.0120	4.0
(25/75/0)[c]	Countersunk shear	24	9.80	45	37	6.4	0.0057	0.0100[d]	3.7
(25/75/0)	Countersunk shear	24	12.7	59	47	6.4	0.0074	0.0084	3.7
(29/71/0)	Countersunk shear (padup)	28	14.7	56	47	6.9	0.0069	0.0080[e]	3.6
(33/67/0)	Countersunk shear (padup)	24	13.6	61	51	7.1	0.0072	—	3.4
(25/50/25)	Countersunk shear and 0.02-in. counterbore	24	12.5	60	47	6.7	0.0070	0.0076	2.8
(25/50/25)	Open hole	40	31.0	59	52	6.7	0.0078	0.0081[e]	2.8
(59/29/12)	Countersunk tension	34	25.3	86	67	12.2	0.0055		2.05
(59/29/12)	Protruding	34	30.6	100	81	12.2	0.0066		2.05
(60/28/12)	Countersunk tension	50	43.1	98	78	12.3	0.0063	Essentially	2.0
[+5/-5] [c]	Protruding	24	21.8	95	82	18.0	0.0045	linear strain	1.0
[+3/-3]	Protruding	24	35.7	165	134	18.0	0.0074	to failure	1.0

[a]Laminate thickness is 0.0074 in. per ply

[b]Based on head travel (6-in. gage length), except as noted

[c]T300/5208

[d]Based on extensometer (1-in. gage length)

[e]Based on extensometer (2-in. gage length)

Note: All widths are 1.5 in. except the 40-ply laminate, which is 2-in. wide

Table 2. Compression Data

LAYUP AS4/X2220-3 EXCEPT AS NOTED	STACKING SEQUENCE[a]	NOMINAL THICKNESS (in.)	IMPACT ENERGY (in.-lb)	DENT DEPTH (in.)	DELAMINATION			FAILURE LOAD (kips)	STRESS (ksi)	STRAIN (in./in.)	
					AREA (in.^{2})	WIDTH (in.)	LENGTH (in.)			NOMINAL P/AE	FAR-FIELD STRAIN GAGE
(0/100/0)	$[+45/-45]_{12}$	0.178	245	0.008	1.50	1.3	1.3	13.3	18.7	0.0094	-
		0.178	—	—	—	—	—	8.2[b]	31.0	0.0150	-
(8/84/8)	$[(+45/-45)_2/0/(+45/-45)_2/90/+45/-45]_S$	0.178	245	0.008	2.08	1.8	1.5	17.5	24.5	0.0057	-
↓	↓	0.178	300	0.021	1.94	1.6	1.7	16.5	23.2	0.0054	-
		0.178	—	—	—	—	—	12.3[b]	46.0	0.0110	-
(25/75/0)[c]	$[(+45/-45/0)_3/+45/-45/+\overline{45/-45}]_S$	0.178	-	-	-	-	-	16.9[b]	63.0	0.0099	-
(25/75/0)[c]	↓	0.178	140	0.017	1.16	1.0	1.6	19.3	27.1	0.0042	-
(25/75/0)		0.178	140	0.006	0.84	0.9	1.3	22.7	31.9	0.0050	0.0049[d]
↓		0.178	245	0.007	1.35	1.0	1.8	20.1	28.2	0.0044	-
		0.178	—	—	—	—	—	16.0[b]	60.0	0.0094	-
(25/50/25)	$[+45/0/-45/90]_{5S}$	0.296	0	-	-	-	-	93.9	63.4	0.0095	0.0114
↓	↓	0.296	0	Hole[e]	0.78	1.0	1.0	52.8	35.7	0.0053	0.0058
		0.296	240[e]	0.005	1.16	1.2	1.3	57.8	39.1	0.0058	0.0064
		0.296	360	0.008	2.44	1.7	2.0	43.8	29.6	0.0044	0.0046
	$[+45/0/-45/90]_{3S}$	0.178	245	0.008	2.03	1.6	1.7	18.7	26.3	0.0039	-
		0.178	-	-	-	-	-	17.9[b]	67.0	0.0100	-
(59/29/12)	$[(+45/90/-45/0_5)_2/+\overline{45/-45}]_S$	0.252	245	0.004	2.84	1.7	2.4	54.1	53.6	0.0044	-
↓	↓	0.252	300	0.009	3.72	2.0	2.7	51.5	51.1	0.0042	-
		0.252	-	-	-	-	-	34.7[b]	92.0	0.0076	-
(100/0/0)[c]	$[+5/-5]_{12}$	0.178	-	-	-	-	-	24.1[b]	90.0	0.0050	-
(100/0/0)[c]		0.178	200	0.005	1.30	0.4	3.0	60.6	85.1	0.0047	-
(100/0/0)	$[+3/-3]_{12}$	0.178	200	0.006	1.10	0.7	3.0	42.5	60.0	0.0033	-
↓	↓	0.178	245	0.005	1.65	0.8	3.0	41.8	58.8	0.0033	-
		0.178	-	-	-	-	-	19.4[b]	73.0	0.0040	-

[a]Refer to Table I for initial extensional modulus

[b]Filled hole data

[c]T300/5208

[d]Strain alongside damage = 0.0065

[e]Standard toughness tests

Note: All coupons were 4 x 6 in., except b coupons were 1.5 x 12 in. and 0.296 coupons were 5 x 10 in.

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STRUCTURAL ELEMENTS

Each of the selected upper panel concepts was tested first as a one-stringer element in the undamaged condition, as shown in figure 8. Before testing the formal designs with full skin bays, two developmental stringers (1I-1 and 1I-2) were tested with only 3 in. of skin. All three concepts were later tested with full skin bays to obtain undamaged baseline data (1I-3, 1J-1, and 1B-1). The J-stiffened concept was tested with and without a double row of 0.25-in. bolts through the skin and stringer. All the undamaged elements failed near the ends. The J-stiffened concept was retested after reinforcement of the ends (1J-3), and the load improvement was 6%. Further tests of undamaged I-stiffened structural elements (1I-4, 1I-5, and 1I-6) and panel (3I-9) showed that load improvements up to 12% are possible with ends reinforced by hot-bonded doublers.

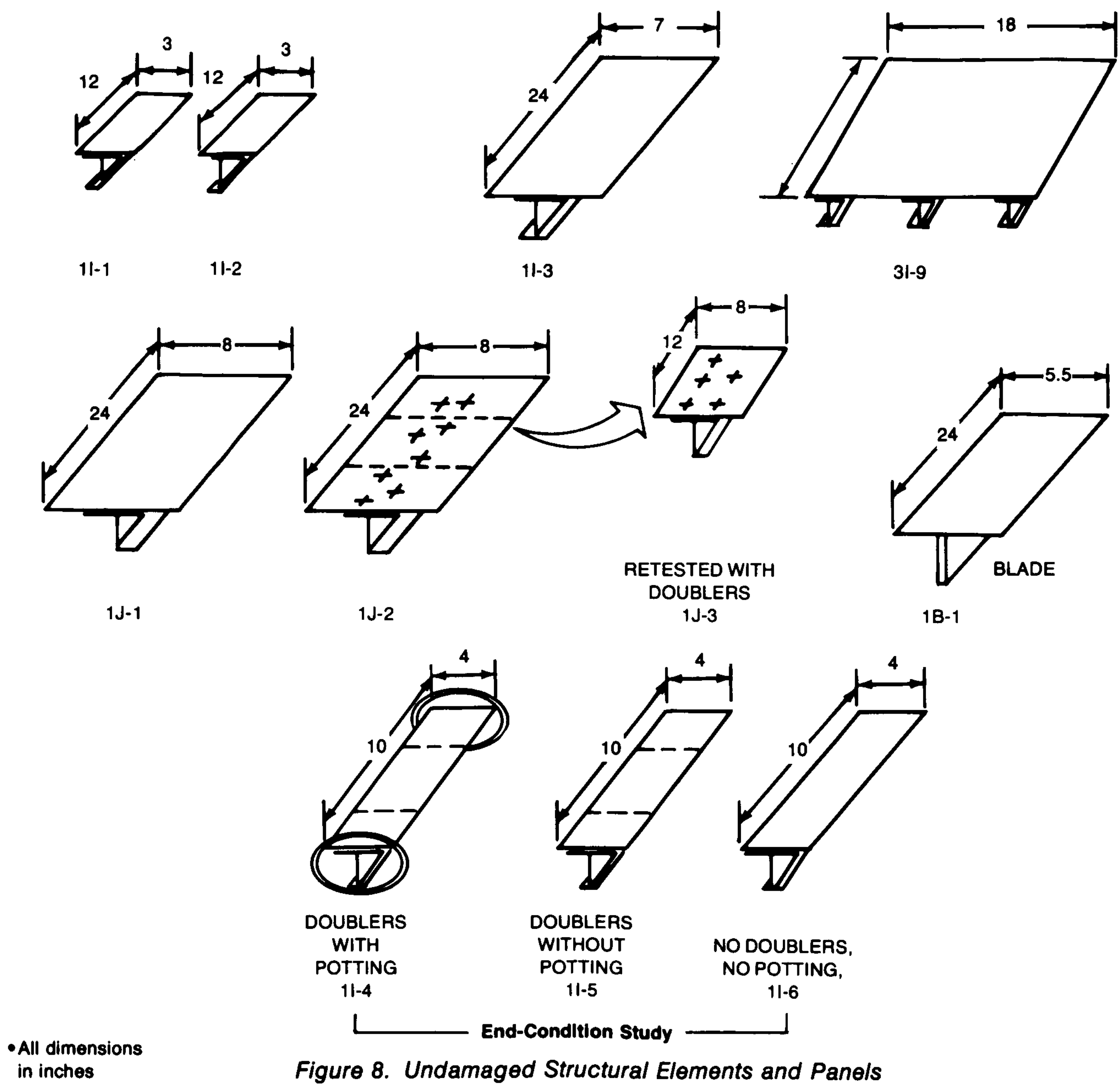

Figure 8. Undamaged Structural Elements and Panels

PANEL TESTS

The test plan for the three-stringer panels was designed to provide data on various levels of damage up to the limits allowed by the size of the panels and to correlate coupon and panel results for establishment of realistic wing design guidelines (fig. 9). Panel 3I-1 was tested to failure after nonvisible impact damage to the skin between stringers to correlate the compression-after-impact coupon results with wing structure results. Panel 3I-3 was tested to failure after nearly half of the [+3/-3]-deg stringer cap was severed. This test was essential to establish the damage tolerance of the discrete, bonded stringers with respect to severe stringer damage, even though this type of damage is considered unlikely in service.

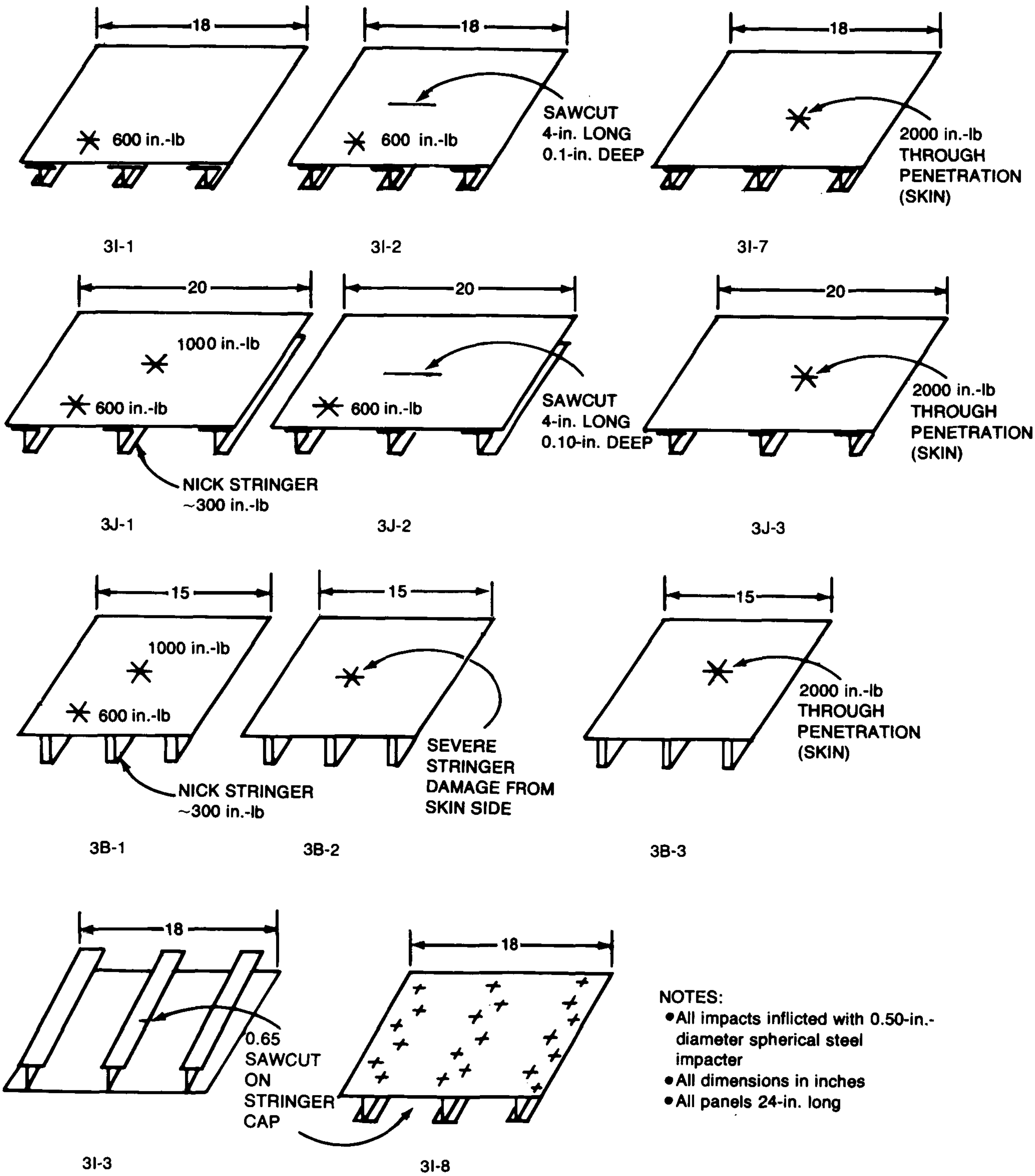

Figure 9. Damaged Compression Panels

Tests on the remaining panels were oriented toward external damage, because this type of damage is most likely to occur on the wing in service. Panels 3J-1 and 3B-1 were tested to failure after visible impact damage, and panels 3I-7, 3J-3, and 3B-3 were tested after severe impact causing through-penetration. These tests were made to find how the level and location of impact damage affected postimpact strength. Panels 3J-2 and 3B-2 were loaded to a truncated version of a flight-by-flight fatigue spectrum after sustaining nonvisible impact damage and gouges. Damage growth was monitored during approximately six lifetimes of compression fatigue cycles, with limit load cycles and nondestructive inspections after each lifetime. The gouges were increased from 0.1-in. to 0.4-in. depth (fig. 10) prior to residual strength tests. Finally, panel 3I-9 was tested without previous damage to supplement the results of the single-stringer elements tested earlier. Figure 11 shows the panel test setup.

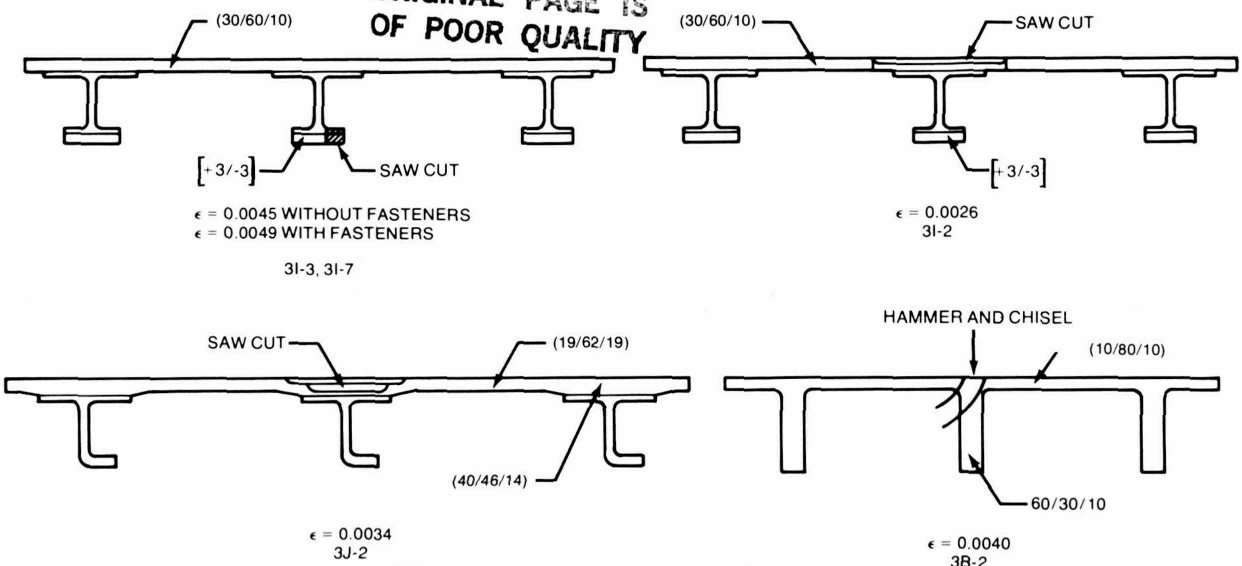

Figure 10. Severe Cuts and Impacts

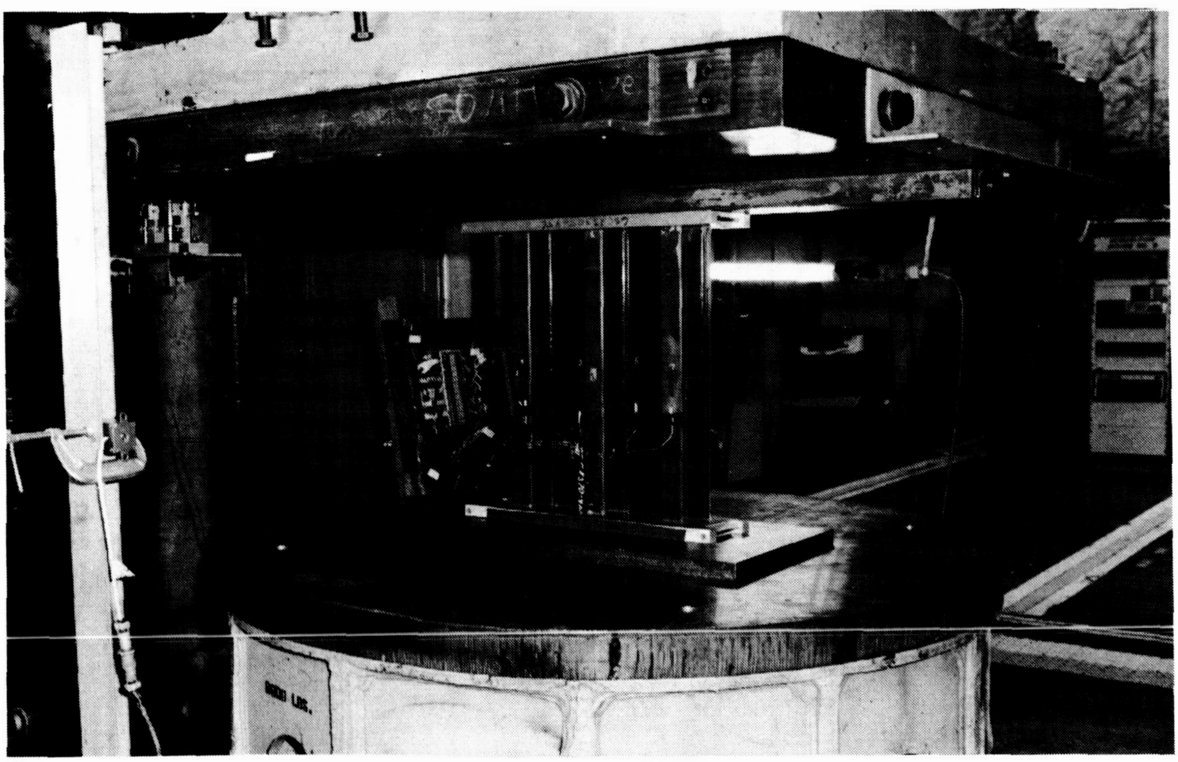

Figure 11. Panel 3I-1 in Test

Impact

Panel 3I-1 was impacted by the same blunt-nosed, 0.5-in diameter steel impactor used for the standard tests (ref. 6), but the mass was increased from 10 to 20 lb to get the same extent of damage, because the panel was more flexible than the 5- by 5-in. frame used to support the standard 7- by 10-in. panels. It required three impacts in the same spot, at 300, 450, and 600 in.-lb, to get damage slightly more severe than the damage on the isotropic 7- by 10-in. panel after a single impact at 360 in.-lb. Panel 3I-1 carried 451 kips at 45 ksi after impact at the outer skin surface halfway between stringers. Failure was through the impact site approximately 6 in. from the lower end of the 24-in.-long panel (fig. 12). The skin and all three stringers failed on a plane through the impact site. The strains, shown in figure 12, were recorded in the center of the panel approximately 6 in. from the failure plane just before failure. This represents a substantial improvement in residual strength over the strength indicated by coupon data and shows the importance of testing panels rather than coupons to get wing panel effective working strain levels.

All three panel concepts were tested in compression after being subjected to impacts that produced nonvisible damage. The first impact was on the skin between stringers. The I-stiffened panel with (30/60/10)% skins (3I-1) was tested to failure after the three impacts noted above. Far-field strains at failure were 0.0054. The J-stiffened panel with (19/62/19)% skins and (40/46/14)% planks (3J-1) was tested to limit load in compression after the first impact. Panels 3J-1 and 3B-1 were impacted a second time directly over the center stringer at 1000 in.-lb by means of a spring-loaded device having a similar impact head and comparable mass to the standard impactor. Craters up to 0.09-in. deep were caused by these skin-side impacts, and delaminations of up to 2-in. diameter (3 in.^{2}) were detected by through-transmission and pulse-echo nondestructive testing.

The third impact on each panel was applied with a sharp-edged steel bar of approximately 5-lb mass. Impact energy was approximately 300 in.-lb, based on experimental drops of other sharp-edged objects on scrap panels. These impacts were applied to the critical stringer details and represent an exploratory level of damage related to assembly of the wing and maintenance operations carried out inside the wing. Figure 13 shows the stringer damage prior to failure. Panel 3J-1 failed through the 1000-in.-lb. external impact site at 576 kips, at an average panel stress level of 44 ksi. Far-field strains were 0.0055 to 0.0058 on the skins and 0.0059 to 0.0061 on the stringer caps.

Panel 3B-1 carried 379 kips, at an average panel stress level of 38 ksi prior to failure. Far-field strains were 0.0051 to 0.0053 when the panel failed through a series of manufacturing defects remote from the intentional damage. The low failure strain of the blade-stiffened panel after nonvisible impact was caused by crooked fibers in the blades. The failure did not pass through any of the three intentional damage sites. The crooked fibers represent a correctable condition caused by improper handling of the fabrication details prior to cure.

Severe Stringer Damage

The most interesting test from a damage-tolerance standpoint was the slow, progressive failure of the I-stiffened panel (3I-3) with a saw-cut center stringer cap (fig. 14). At 170 kips, the gage on the cap near the end of the saw-cut was lost. The last recorded strain was 0.0037. The other gage near the end of the saw-cut increased from 0.0039 at 170 kips to more than 0.017 at 270 kips.

At 270 kips, there was a loud pop accompanied by sudden increases in the side stringer cap strains from 0.003 to 0.0034. The sudden loss of load in the center stringer cap was picked up by the cap gages 2, 4, and 6 in. from the saw cut. At 330 kips, there was a second pop accompanied by a sudden increase from 0.0017 to 0.0033 in skin-side strain immediately above the broken stringer. Finally, at 350 kips, the entire skin laminate buckled away from the stringers, the stringers failed, and the load dropped abruptly to zero.

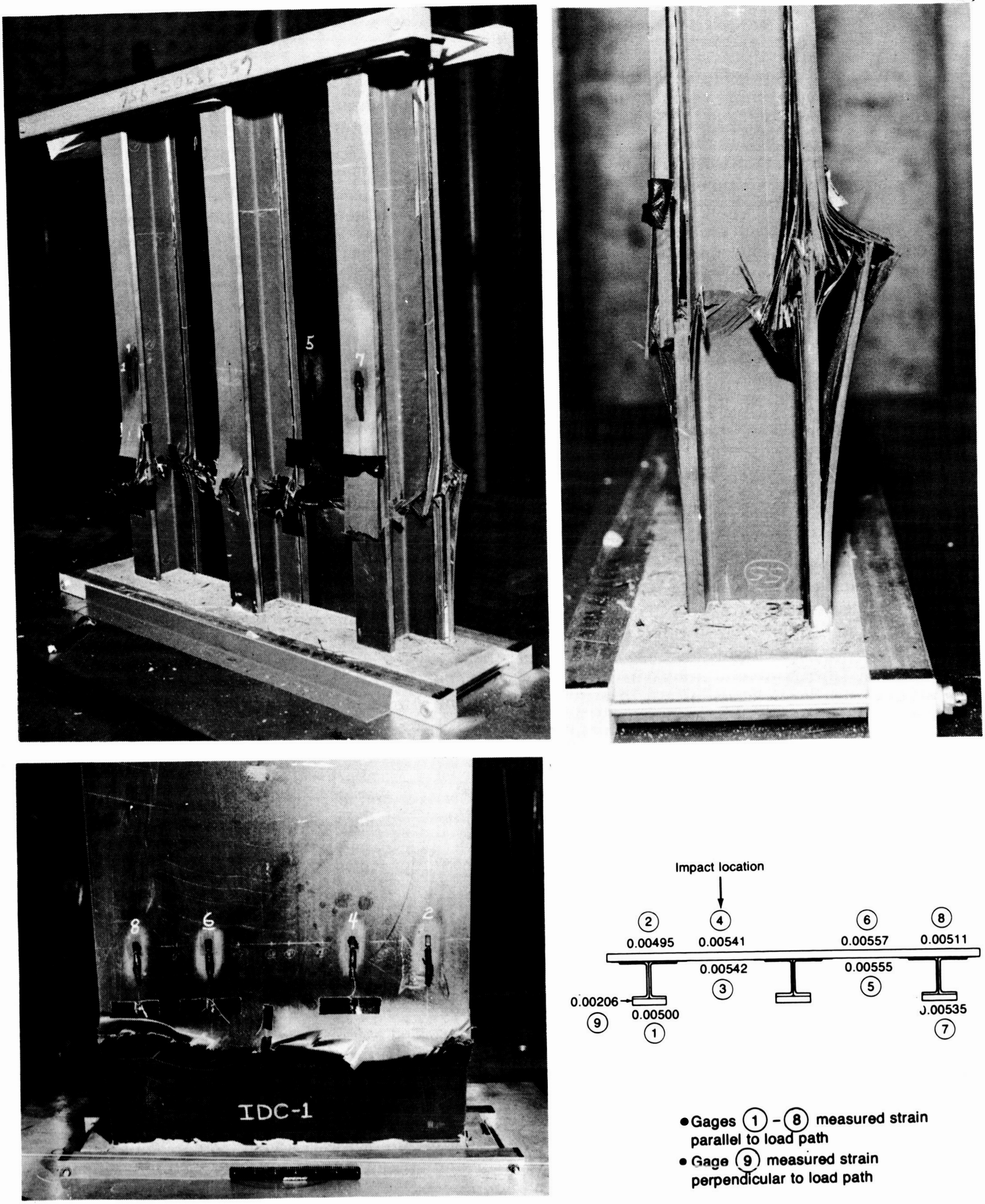

Figure 12. Failed 3I-1 With 600 in.-lb Impact

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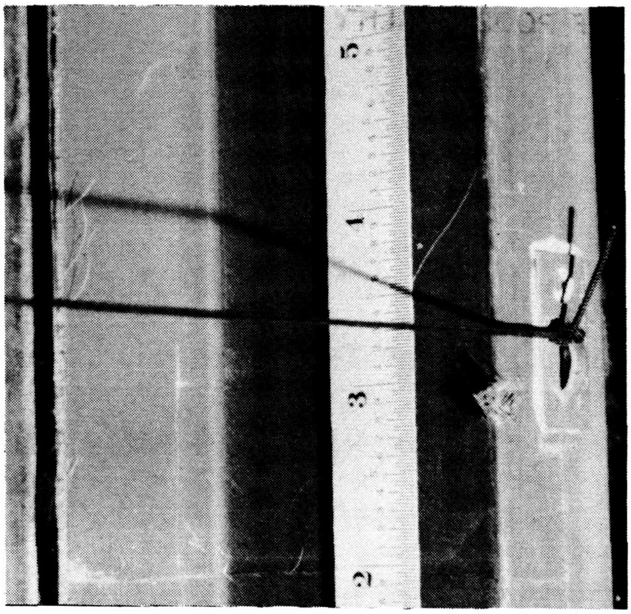

Figure 13. Stringer Impact Damage on Panel 3J-1

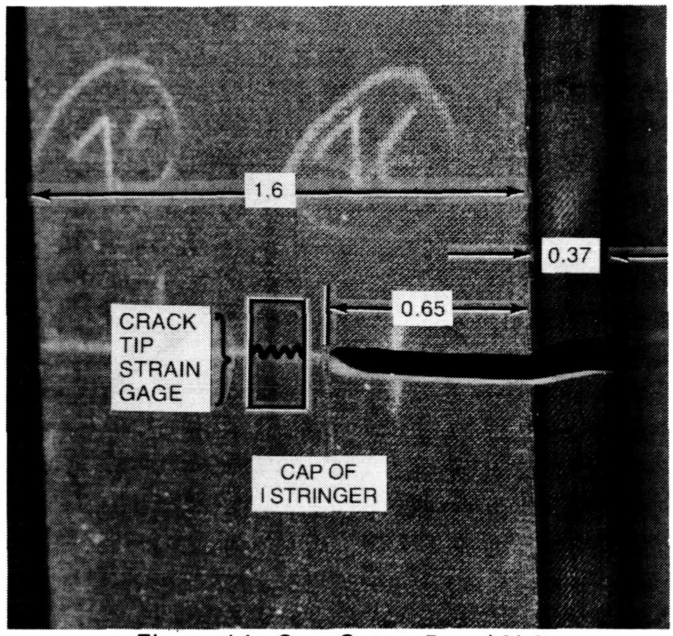

Figure 14. Saw Cut on Panel 3I-3

This progressive failure sequence demonstrated for the first time some inherent damage-tolerance of skin-stringer panels with bonded-on stringers. The load from the failed central stringer was redistributed to the side stringers. The bond lines demonstrated the capacity to absorb the shock loads associated with explosive failure of the stringer element. The shear material in the stringer web proved capable of transferring load between a failed cap and the redistribution load path, the 45-deg-dominated skin. The redistribution load paths remained intact until the far-field strains exceeded 0.004. This indicates that a wing panel having five or more stringers would carry limit load after complete failure of one stringer. This test was repeated on a similar panel with bolts reinforcing the skin/stringer bonds. The failure sequence described above was recorded on high-speed movies. The major redistribution occurred at 230 kips, but ultimate failure load increased 12% to 394 kips.

Through-Penetration Impact

All three concepts were tested to failure after impact causing through-penetration of the skin alongside the central stringer. The same 0.5-in.-diameter spherical impact head was used, and the energy level was 2000 in.-lb. The central stringer was undamaged in all cases, providing a nearby redistribution load path. Far-field strains at failure were 0.0039 on the I-stiffened panel with (30/60/10)% skins, 0.0041 on the J-stiffened panel with (19/62/19)% skins and 0.0067 on the blade-stiffened panel with (10/80/10)% skins.

Severe Skin Side Damage

Figure 10 shows the maximum damage levels sustained by LCPAS panels. The severe stringer damage was described above. The skin cuts on panels 3I-2 and 3J-2 and the skin side stringer damage on panel 3B-2 represent exploratory levels of damage addressing the FAA requirement for continued safe flight and landing following engine burst. Figures 15 and 16 show all of the LCPAS panel data relative to the final design stresses and strains.

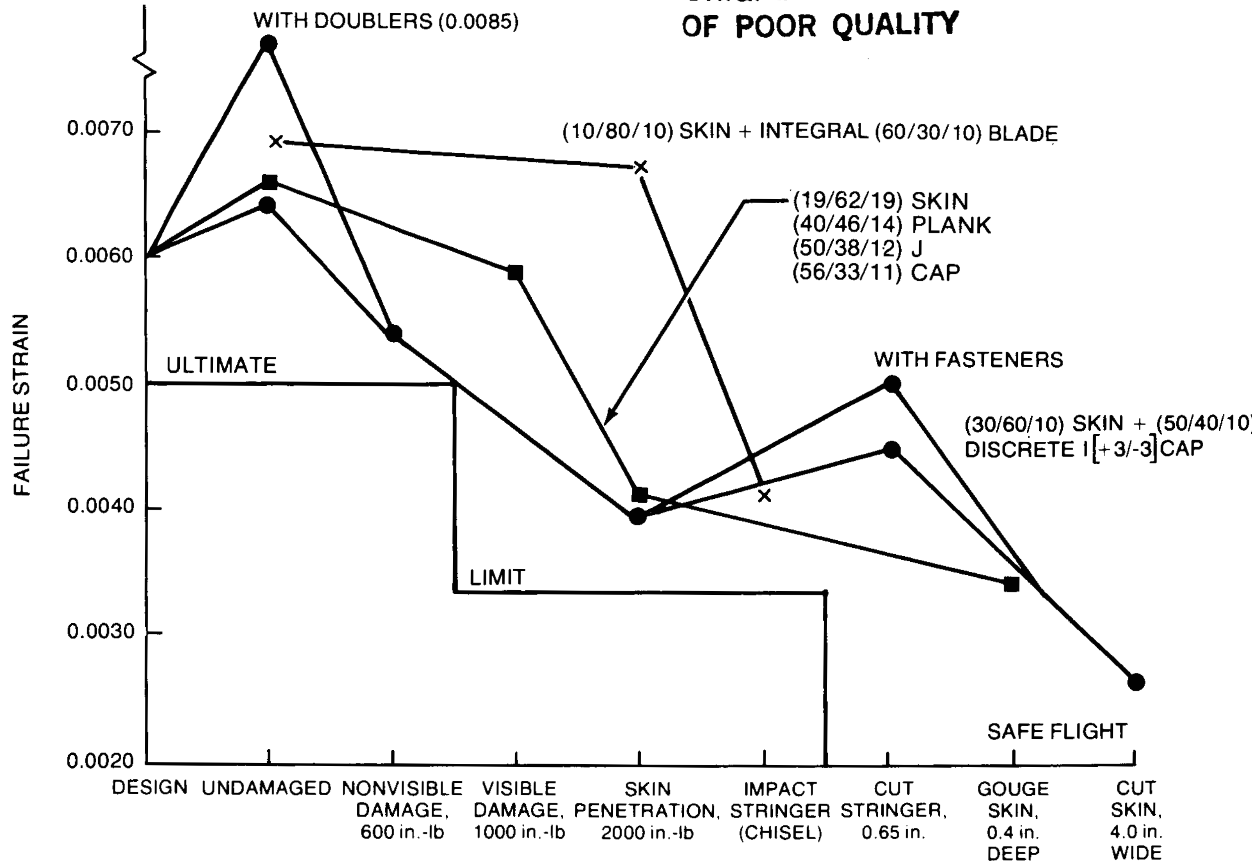

Figure 15. Panel Test Results – Ultimate Demonstrated Capability

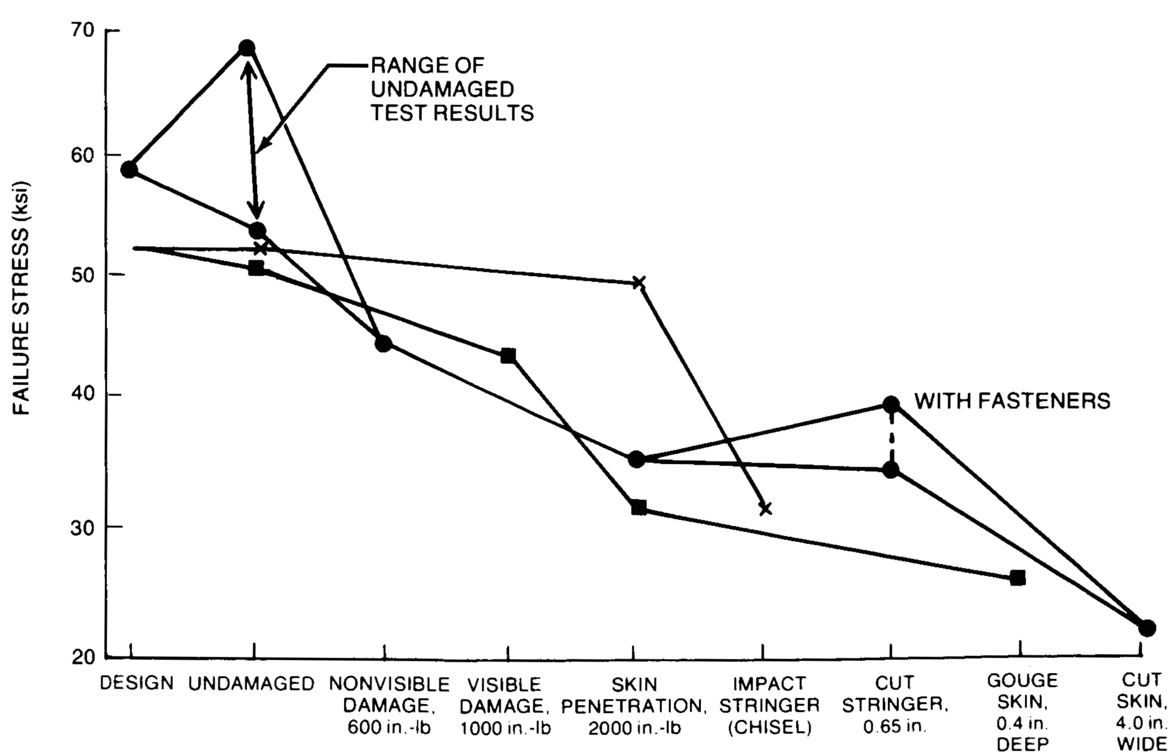

Figure 16. Panel Test Results – Stress

Summary

Tables 3 and 4 summarize the results of the LCPAS panel tests. The basic panel dimensions, design shear stiffness, and failure loads are shown. The end loads and stresses have been corrected for full skin bays, assuming the skins were fully effective. This assumption was validated by the strain-gage data, which indicated no skin buckling prior to failure. Strain gages were monitored during all the panel tests, and the average far-field strains at failure are shown for each panel. The far-field gages were on the skins and side stringers. Typically, all the far-field strains at failure were within 10% of the average. The average panel stresses are shown. The stresses are based on nominal area, and the actual cross-sectional areas were all within 3% of nominal.

Table 3. Design Data

	STRINGER PITCH (in.)	NOMINAL SKIN GAGE (in.)	SHEAR MODULUS (Msi)	$t/\bar{t}$	$P/\bar{P}$	Gt (kips/in.)
LCPAS						
I	7.00	0.370	3.20	0.705	0.580	1184
J	8.00	0.385	3.25	0.668	0.712	1251
Blade	5.46	0.311	3.90	0.503	0.292	1212
R&W*						
1	5.00	0.218	3.88	0.574	0.414	847
3	8.79	0.140	4.58	0.397	0.145	642
4	6.00	0.140	4.58	0.318	0.106	642

*Reference 4

Table 4. Panel Test Data

	PANEL NO.	DAMAGE SIZE (in.)	FAILURE LOAD (kips)	END LOAD (kips/in.)	FAILURE STRESS (ksi)	FAILURE STRAIN
Undamaged one-stringer elements	1I-5		190	36.1	69.0	0.0084
	1J-3		238	29.8	50.4	0.0066
	1B-1		174	31.9	51.6	0.0069
Undamaged panel	3I-9		539	28.0	53.3	0.0062
Barely visible impact damage 600-1000 in.-lb	3I-1	0.005	451	23.4	44.6	0.0054
	3J-1	0.09	576	25.6	43.5	0.0059
	3B-1	Defects	379	23.7	38.4	0.0053
Skin penetration (2000 in.-lb)	3I-7	0.5 diameter	358	18.5	35.4	0.0039
	3J-3	through	421	18.8	31.8	0.0041
	3B-3	skin	486	30.4	49.2	0.0067
Severe stringer damage	3I-3	0.65 cut	350	18.2	34.6	0.0045
	3I-8	0.65 cut	394	20.5	38.8	0.0049
	3B-2	chisel	298	18.8	31.8	0.0040
Skin cuts (after fatigue)	3I-2	4.0 through	223	11.6	22.1	0.0026
	3J-2	0.4 deep	350	15.6	26.4	0.0034

- End loads and failure stress based on full skin bays
- Failure strains based on far-field gages

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Figure 17 represents a summary of the LCPAS test data on undamaged structural elements and damaged panels.

With nonvisible damage, the LCPAS concepts represent an 8% improvement in compression panel efficiency compared to the best previous industry data obtained by Rhodes and Williams (ref. 4) on panels 1, 3, and 4. The LCPAS panels with nonvisible damage represent a 27% improvement over the best (undamaged) data for aluminum panels.

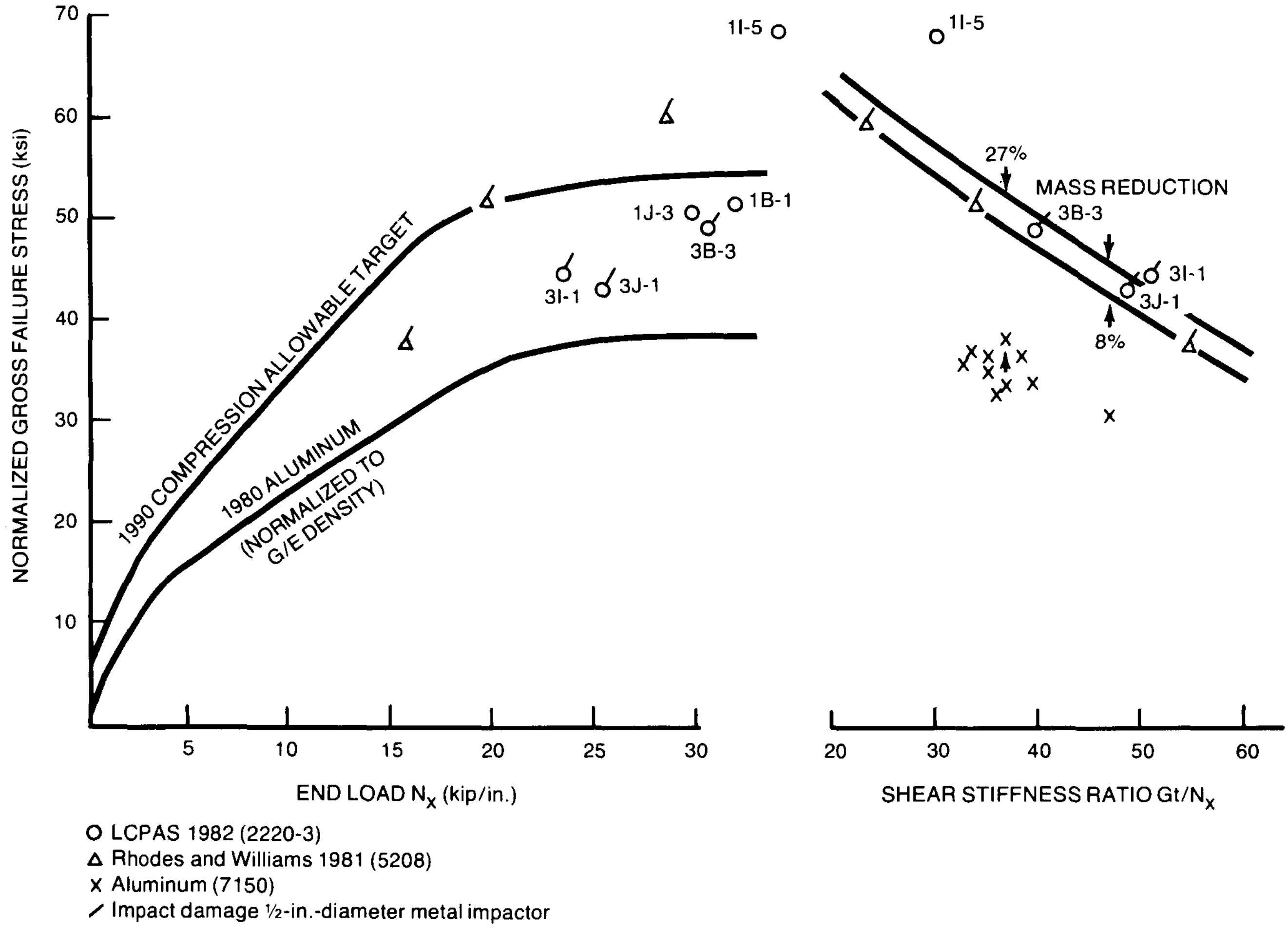

Figure 17. Compression Panel Test Summary

FINAL SIZING

Although the blade-stiffened panel can be made to carry 52 ksi as a short column with no local pressure, the demands of combined loads and residual strength favor a discrete stiffener with a substantial bending capability, as provided by the I or J concepts. Because current tooling and assembly techniques favor the I over the J concept, the I concept was selected for the final LCPAS sizing. Because the panel test data, especially for panels 3I-2 and 3B-3, indicate that soft skin configurations are more damage tolerant, the selected skin layup is (11/78/11)%. The wing size shown in figure 18 is based on maximum fiber-direction strains of 0.005 in compression and 0.006 in tension. The soft lower skins with discrete planks were maintained to provide discrete tear straps. Although the almost unidirectional laminate provided good stiffness, strength, and toughness, it would be difficult to repair with mechanical fasteners, so the percentage of 0-deg material was maintained below 67% in all parts of the structure. Although the all-45-deg laminate exhibited damage tolerance, the Poisson ratio associated with this layup might produce fatigue cracks in the metal fittings attached to the wing panels, and the typical rib chords would be highly loaded.

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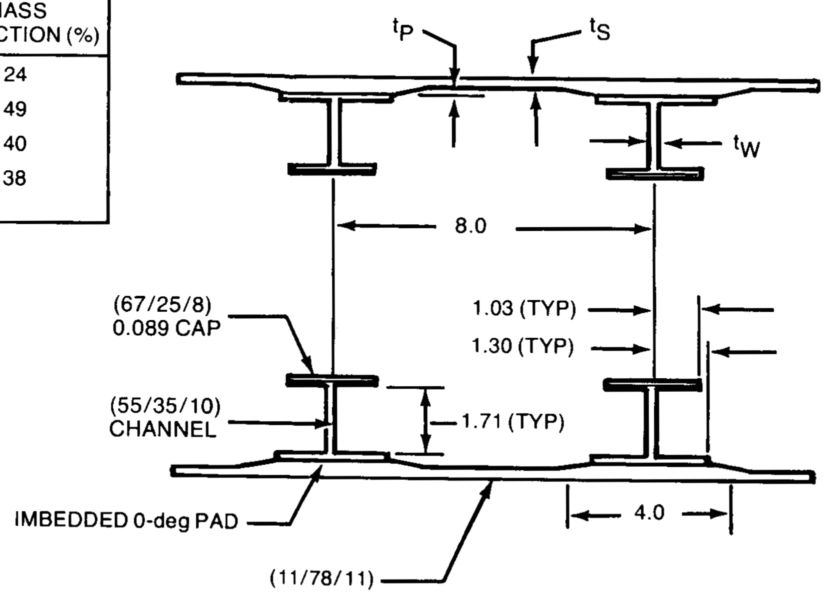
Figure 18 presents the results of the final wing sizing based on the LCPAS test results. The theoretical mass reduction for wing panels and spars is 38%.

COMPONENT	THEORETICAL MASS REDUCTION (%)
UPPER SURFACE	24
LOWER SURFACE	49
SPARS	40
TOTAL (PANELS + SPARS)	38



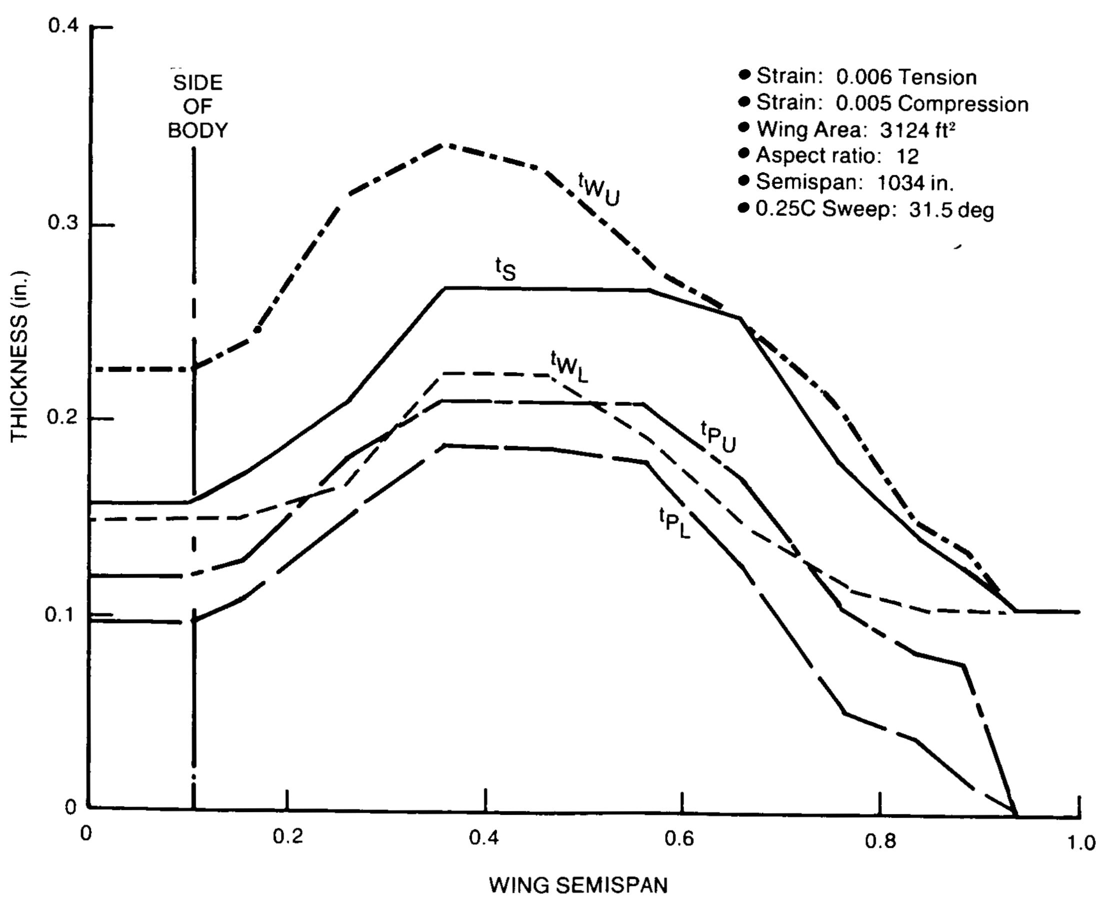


Figure 18. Wing Sizing and Mass Comparison

CONCLUSIONS

Durable, damage-tolerant, and producible commercial wingbox surface panels were designed, built, and tested. The durability of the tension and compression load paths was demonstrated by testing at the coupon level in spectral-type loading. The damage tolerance of the compression load paths was demonstrated at the three-stringer level with respect to the appropriate limit and ultimate strength criteria. The impact energy levels required to damage the wing panels were much higher than those required to damage the coupons. The bonded-on side stringers proved capable of carrying load from a broken central stringer, even when the damage to the central stringer was sudden and progressive under high compression load conditions. The test data were used to resize the wing panels and spars of the reference commercial transport airplane. The data suggest a theoretical mass reduction of 38% for the AS4/X2220-3 graphite-epoxy panels and spars relative to current aluminum.

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16. Abstract
Analysis and testing that addressed the key technology areas of durability and damage tolerance were completed for wing surface panels. The wing of a fuel-efficient, 200-passenger commercial transport airplane for 1990 delivery was sized using graphite-epoxy materials. Coupons of various layups used in the wing sizing were tested in tension, compression, and spectrum fatigue with typical fastener penetrations. The compression strength after barely visible impact damage was determined from coupon and structural element tests. One current material system and one toughened system were evaluated by coupon testing. The results of the coupon and element tests were used to design three distinctly different compression panels meeting the strength, stiffness, and damage-tolerence requirements of the upper wing panels. These three concepts were tested with various amounts of damage ranging from barely visible impact to through-penetration. The results of this program provide the key technology data required to assess the durability and damage-tolerance capability of advanced composites for use in commercial aircraft wing panel structure.

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