15 Durability and Damage Tolerance of Laminated Composites

This section addresses durability and damage tolerance of laminated composites including failure mechanisms, testing and development of material allowables, certification requirements and analysis While this section provides a good overview and general guidance on these topics, analysis criteria, methodology and material allowables are program-specific and are governed by the program Structural Design Criteria Document and program analysis guidance.

Contents

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Nomenclature

Symbol/Acronym	Description	Units
ASTM	American Society of Testing Materials	
BAI	Bending After Impact	
BMI	Bismaleimide	
BVID	Barely Visible Impact Damage	
C_1-C_{11}	Regression Constants	
CAI	Compression after Impact	
CDADT	Composite Durability and Damage Tolerance Analysis Program. Part of IDAT Tools	
CSAI	Compression Strength after Impact (same as CAI)	
CTA	Cold Temperature Ambient, typically -65°F	
D_{11}, D_{22}, D_{12}	Bending Stiffness Parameters	
D_{11S}, D_{22S}, D_{12S}	Smeared Bending Stiffness Parameters	
DADT	Durability and Damage Tolerance	
DLL	Design Limit Load	
EMD	Engineering and Manufacturing Development	
ETW	Elevated Temperature Wet, typically 220°F, 250°F or 375°F and 85% Relative Humidity	
E_x	Elastic modulus in the material x direction	psi
E_y	Elastic modulus in the material y direction	psi
FHT	Filled Hole Tension Strength	
G_{xy}	Shear modulus	psi
HFT, HRP	Types of Honeycomb Core	
IBOLT	Composite bolted joint analysis tool. Part of IDAT Tool Suite	
IDAT	Integrated detail Analysis Tool Suite	
ILS	Interlaminar Shear Stress	psi

ILT	Interlaminar Tension Stress	psi
JSSG-2006	Joint Services Specification Guide Reference 15-2	
L	Length	in
OEM	Original Equipment Manufacturer, *e.g.*, Lockheed Martin	
OHT	Open Hole Tension Strength	
OML	Outer Mold Line	
PACT	Finite plate multiple cutout analysis tool. Part of IDAT Tool Suite	
P_{xx}	Residual Strength Load	lbs
R	Stress Ratio	
RTA	Room Temperature Ambient, typically 75°F	
σ_{allow}	Allowable Stress	psi
σ_{avg}	Average Applied Stress	psi
S-N Curve	Stress-Number of Cycle Fatigue Curve	
SDC	Structural Design Criteria Document	
SFBOLT	Bolt Strength and Fatigue analysis tool. Part of IDAT Tool Suite	
SLEP	Service Life Extension Program	
SQ5	Composite Lamination Theory Analysis Tool. Part of IDAT Tool Suite	
TAI	Tension after Impact	
UNT	Unnotched Tension Strength	
W	Width	in

15.2 Introduction

Durability, damage tolerance and cyclic load effects must be taken into account when sizing composite structures. Normally these design requirements fall into two classes of requirements: durability requirements and damage tolerance requirements with each including cyclic effects. The fiber-reinforced matrix orthotropic nature of composite materials used on high performance aircraft has presented difficulties in developing closed form solutions to fatigue and fracture problems. Instead of developing equations, composite allowables dealing with these issues are test and semi-empirically based. This section addresses the durability and damage tolerance requirements which are usually levied against the composite parts of the airframe.

Per Reference 15-2 all structural parts are classified based on their criticality. Reference 15-14 provides further discussion and a flow chart to aid in this classification. The most critical parts are designated fracture critical followed by durability critical, maintenance critical and the least critical parts are normal controls. The classification governs the traceability requirements, type of analyses, impact levels, and manufacturing process controls.

All structural composite parts have a durability requirement while the fracture critical parts have the additional, more severe Damage Tolerance requirement.

15.2.1 Durability Damage and Failure Mechanisms

Durability may be defined as the ability of material to resist degradation over a spectrum of time, environment and loading. Durability is generally viewed as an economic rather than a safety of flight issue. The effect of aging, repeated loads and environmental effects can degrade the composite laminate in various ways.

To appreciate the action of fatigue in composites the interplay of the fiber and matrix in proper composite material strength must be understood. The fiber carries the tension and compression loads but must be supported by the matrix. The matrix transfers the loads between the fibers and in compression supports the fibers from micro-buckling and failure. Composite fatigue tends to involve a process of matrix cracking leading to diffuse damage and lack of proper consolidation of fiber and matrix. When out-of-plane tension or shear is the dominant loading, this can lead to interlaminar delamination and inherent stiffness and strength reduction. If the load is substantially in-plane, diffuse matrix damage and fiber-matrix disbonding will lead to fiber failure at the micro-level, eventually leading to macro-rupture failure. Figure 15.2-1 from Reference 15-1 presents various load-related fiber/matrix disbonding diffuse matrix damage and eventual interlaminar delamination.

In addition to the damage resulting from fatigue, levels of minor damage that are expected to be present in the airframe are also tested. The sources of minor damage may be small manufacturing defects, tool drops during maintenance actions, *etc*. The damage is idealized by introducing low velocity, high probability impacts depending on the applicable criteria and requiring critical environment residual strength after two service lifetimes. Structure must be sized so that these impacts cause no functional impairment (strength reductions), moisture intrusion or visible damage. In this way the durability requirements can be thought of as damage resistance. These impacts have unique stability-related failure mechanisms similar to the larger damage tolerance impact summarized in Section 15.2.3.

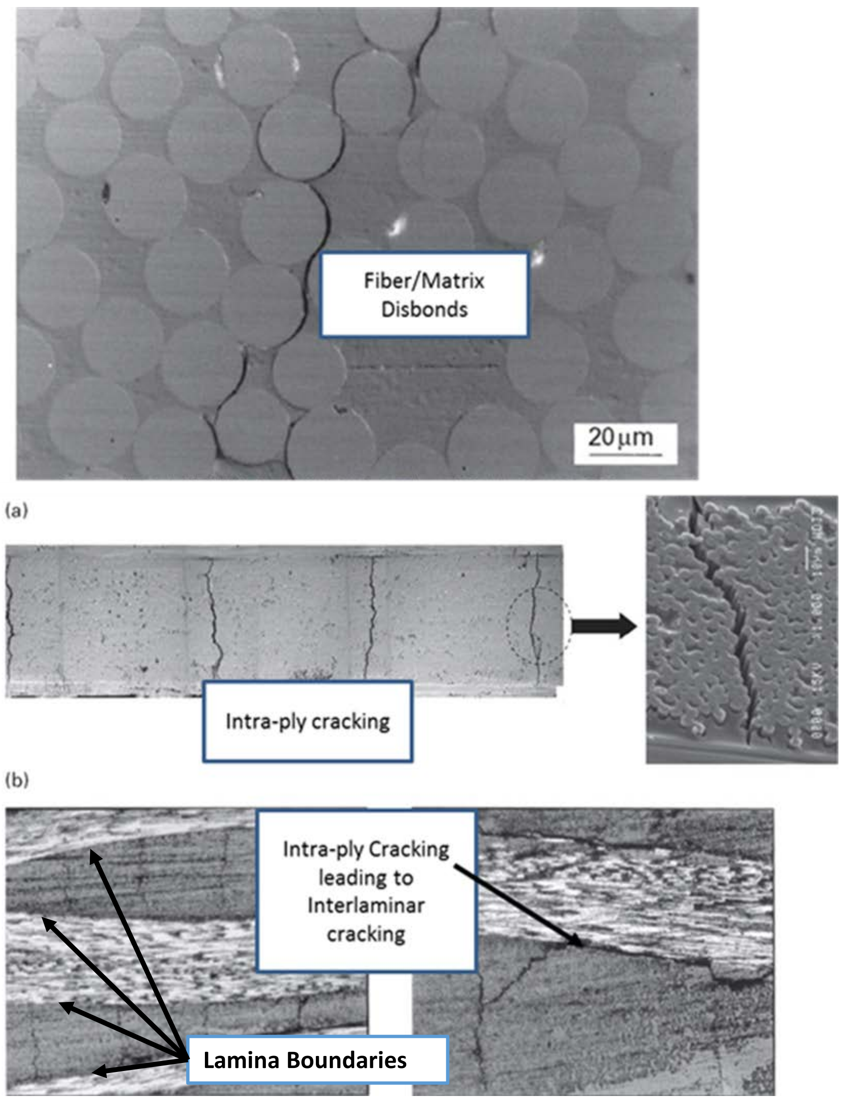


Figure 15.2-1 Progression of Matrix Cracking

15.2.2 Durability Design Requirements

Durability requirements for composite structure are met by following these procedures:

1) Utilize program-specific “static” composite allowables. The “static” allowables are formulated to protect the structure for conservative low-cycle spectrum cyclic loading, usually related to wing-root bending moment and aircraft vertical acceleration. When cyclic loading varies significantly from this exceedance set more scrutiny may be needed. See Sections 15.3 for guidance in verifying durability when developing “static” allowables. Strength margins resulting from IDAT programs such as SQ5, IBOLT and PACT, in conjunction with program specific material files, will yield acceptable durability when calculating “static” margins.
2) Develop and apply applicable high-cycle fatigue allowables for areas of high aeroacoustic loading.
3) Apply durability impact allowables to applicable areas of structure. IDAT/CDADT is used for this purpose. CDADT is program specific and users outside of those programs should seek methods group guidance for selection of appropriate variant[1]. See Section 15.8.2.
4) Verifying that minimum thickness requirements have been meet. See Section 15.3.1.3
5) Depending on program guidelines there may be durability “stand-in” requirements. These include adding open or filled-hole checks in panel acreage for the purpose of future repairability. Consult program guidelines.

15.2.3 Damage Tolerance Damage and Failure Mechanisms

Damage tolerance is, as the term suggests, the ability for composite structures to tolerate damage of varying types and perform as intended. Given the nature of composite materials there are a variety of different types of failure modes that may be present. These include delamination, microcracking, fiber breakage and others. Central to the damage tolerance requirements levied on the airframe is the threat of delaminations from impact. Composites can develop delaminations due to impacts not visible to the outer surface that can seriously degrade compression strength.

Figure 15.2-2 presents the basics for the compression mechanism that leads to failure with composite panels possessing delaminations. The buckling sublaminates are the driving force behind the failure modes. Low velocity impacts cause interlaminar shear forces exceeding interlaminar strength. The resulting delaminations are usually multilayer depending on the impact level. As compression forces are applied the resulting sublaminates will eventually buckle which causes Mode I tension prying forces at the crack fronts. If the delamination strain energy exceeds the critical strain energy release rate of that material and mode, then the delamination will grow across the whole panel resulting in collapse.

The mechanisms outlined above are complicated by the influence of curvature in the laminates. Curvatures could arise from initial geometry or from the action of bending moments applied to the laminate. Since the failure mechanism can be idealized as related to buckling, curvature in panels can be seen as a factor which may affect the mechanism. In similar fashion, delaminations in panels with curvatures from bending moments or in postbuckled panels may also be affected. Analysts should be aware of the possible need for targeted tests and additional analysis for fracture critical curved laminates in compression or postbuckling panels.

[1] The IDAT/CDADT code and materials are tailored by program and the appropriate specific program version should be used.

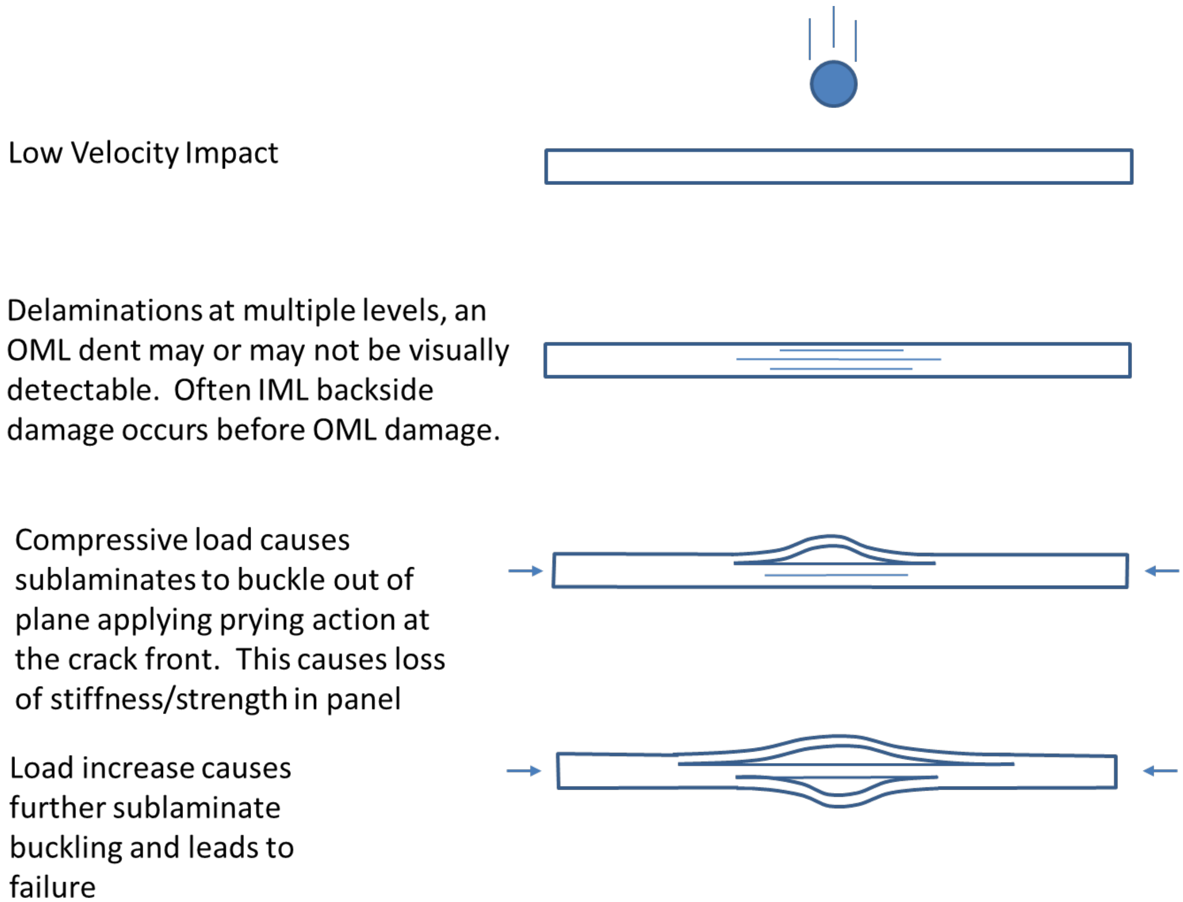


Figure 15.2-2 Composite Delamination Event

15.2.4 Damage Tolerance Design Requirements

Composite structure classified as "Fracture Critical" will require damage tolerance analysis. More severe than durability impacts, composite damage tolerance requirements are accomplished by performing a damage tolerance analysis on the applicable facture critical laminates utilizing the CDADT program, or via element/component testing. Within IDAT, versions of CDADT are available. Users should consult the methods group for program specific variants. Refer to section 15.8.2

15.3 Certification Aspects, Allowables, Analysis and Testing

15.3.1 Durability Certification Requirements

Reference 15-2, Joint Services Structures Guide (JSSG-2006) Section 3.11, presents guidelines for the durability requirements of composites. The JSSG-2006 is not a specification or requirement but rather serves as a starting point for tailored specification requirements negotiated for a structural program. These are codified in the program's Structural Design Criteria (SDC) document. From Reference 15-2:

"*3.11 The durability capability of the airframe shall be adequate to resist fatigue cracking, corrosion, thermal degradation, delamination, and wear during operation and maintenance such that the operational and maintenance* *capability of the airframe is not degraded and the service life, usage, and other provisions of 3.2.14 are not adversely affected. These requirements apply to metallic and nonmetallic structures, including composites, with appropriate distinctions and variations as indicated. Durability material properties shall be consistent and congruent with those properties of the same material, in the same component, used by the other structures disciplines."*

These durability requirements require certain tasks to be included during the development of the composite allowables. Certain allowables, normally referred to as the "Static Ultimate Allowables" must include the effects of cyclic loading, thermal degradation and environmental effects. Environmental effects include applicable temperature variations and lifetime moisture pickup based on conservative environments. The static ultimate allowables include but are not limited to the unnotched tension and compression, open and filled-hole tension and compression, interlaminar shear and tension and the various bearing and bearing/bypass failure modes. Important in the development process is the fact that the static allowables must be capable of resisting ultimate load at the critical design environment without failure and two lives of representative cyclic loading with no progressive damage. Most of the assumptions used for composite allowables for manned aircraft assume a 1.5 factor between ultimate and limit load. Factors less than 1.5 would have a profound impact on the allowables process. This is discussed in Section 15.3.1.1

In addition to the standard static ultimate allowables, the JSSG-2006 suggests that a certain amount of statistically likely impact damage should be included in the allowables mix; usually one level of impact energy is specified for no visible damage and a slightly greater one for no functional impairment, *i.e.*, strength. There is no ultimate strength requirement but similar to the "static ultimate allowables" is the requirement for residual strength at the critical environment after two representative service lifetimes. The residual strength load (P_{xx}) for these allowables varies among the various composite airframe programs but is generally in the 105% to 120% of Limit Load range. The residual strength load level, P_{xx}, is related to the probability of reaching a certain percentage of limit load. Current JSSG-2006language, Reference 15-2, requires the $_{xx}$ load level that occurs one in 20 lifetimes.

There can be other analyses as required by the program Structures Design Criteria document or program-specific guidance. Typically this might include a requirement for ensuring composites are repairable. To meet this requirement, the composites are analyzed for open or filled-hole tension or compression in structural acreage. Structural acreage is defined as areas of laminates in centers of panels or webs away from joints, holes or other stress concentrations.

15.3.1.1 Durability Allowables Development

The requirements for the airframe to "*resist fatigue cracking, corrosion, thermal degradation, delamination and wear*" need to be verified through a building block development program. Typically the low level verification is performed by applying a low-cycle spectrum and constant amplitude loading to the various test types, *i.e.*, unnotched, open hole, filled hole, *etc*. with conservative spectrum loading to verify that no adverse effects occur. Test methods for these can be found in Reference 15-4. As the modes of failure for a coupon go from fiber dominated to matrix dominated the chances of fatigue life reduction are increased. Table 15.3-1 presents a qualitative comparison of the different failure modes and the sensitivity to cyclic effects.

Table 15.3-1 Sensitivity to Cyclic Effects by Failure Mode Type

Type	Test Type	Level of Cyclic Effects	Load Level Basis
Fiber Dominated	Unnotched Tension (UNT) Open Hole Tension (OHT) Filled Hole Tension (FHT)	Low	Ultimate
Mixed Fiber/Matrix	Unnotched Compression (UNC) Open Hole Compression (OHC) Filled Hole Compression (FHC)	Medium	Ultimate
Matrix Dominated, Static Based	Interlaminar Shear (ILS) Interlaminar Tension (ILT)	High	Ultimate
Matrix Dominated, Impact Based	Strength after Durability Impact	High	Residual Strength, $P_{xx,}$
Adhesive	Lap Shear, Peel	High	Ultimate

Typically the OHC and OHT allowables are studied to verify that no cyclic effects need to be included in the static allowables. Steps in the verification process:

1) Perform static ultimate allowables tests at RTA and critical environments, *i.e.*, elevated/wet and cold/dry
2) Develop a conservative spectrum with approximately 50% truncation. Unlike metals, a conservative spectrum is developed which is the most conservative maneuvering spectrum found on the aircraft. This conservative spectrum eliminates the need to develop part/site-specific spectra.
3) Verify the required residual load level to be performed after each lifetime of cycling. The residual load level referred to as P_{xx}, is usually a level negotiated with the customer, between 105% and 120% DLL. Per current JSSG-2006guidelines this corresponds to a once-in-20-lifetime loading.
4) Perform two lives of spectrum fatigue loading with reference load at 2/3 of the typical failure load.
5) If the cyclic coupon survives P_{xx} after two lives of spectrum fatigue the applicable static ultimate allowable is considered to not require any cyclic reductions. The allowables are published and when ultimate margins checks are performed the margins area also positive for cyclic loading.
6) If two lives at 2/3 average-static-failure-load cannot be accomplished, additional cyclic tests at RTA need to be performed to determine the acceptable P_{xx} after two lives. Reference loads are reduced until two lives are achieved and an applicable P_{xx} is found. The previously developed static ultimate allowables are then reduced by applying a factor of [(Reference Load)$_{successful}$] / [Typical Failure Load/1.5].

Figure 15.3-1, taken from Reference 15-3, presents open-hole tension and compression static and 2-lifetime stress allowables data for a sample material. The dashed lines represent the limit/ultimate ratio (2/3) applied to the static failure stresses. If two lifetimes of spectrum fatigue can be applied with a reference load greater than these values, the statically based allowables do not have to be adjusted for cyclic effects. As can be seen in this figure, the cyclic failure stresses are higher than 2/3 of the typical failure stresses, and no cyclic adjustment need be made for this material.

Figure 15.3-2, compiled from data contained in Reference 15-5, presents constant amplitude bearing stress cyclic data for AS4/3501 tape. In the same fashion as the previous curve it can be seen that approximately 10,000 limit load (2/3 typical failure) cycles can be performed before failure. While no spectrum load exists, the capability of performing this number of limit load cycles should allow static derived allowables to be used with no cyclic knockdown.

The JSSG-2006 requirement that the airframe will be free of cracking, delamination, *etc.* during the life of the airframe is borne out of the observation that slow crack growth of composites is not a practical design concept. The process of fiber/matrix disbonding, microcracking, diffuse damage, delaminations and final failure often yield little observable effects for substantial cycling followed by rapid damage growth and abrupt failure. Figure 15.3-3 from

Reference 15-6 presents a plot of hole elongation versus number of cycles. Figure 15.3-4 from Reference 15-7 presents a similar chart for open-hole tension loading.

The sensitivity for fatigue of matrix-dominated interlaminar stresses may be seen by comparing the data in Figures 15.3-5 and 15.3-6 from Reference 15-8. Figure 15.3-5 presents S-N data for open-hole tension-compression constant amplitude fatigue. Note the shallow slope of the curve. Compare this to Figure 15.3-6 which presents S-N Interlaminar Tension (ILT) data. Note the slope is much steeper than the open-hole fatigue data and, just as important, note how scatter in the ILT data is much greater. These two trends of increased S-N slope and increased scatter are typical of interlaminar cyclic behaviors.

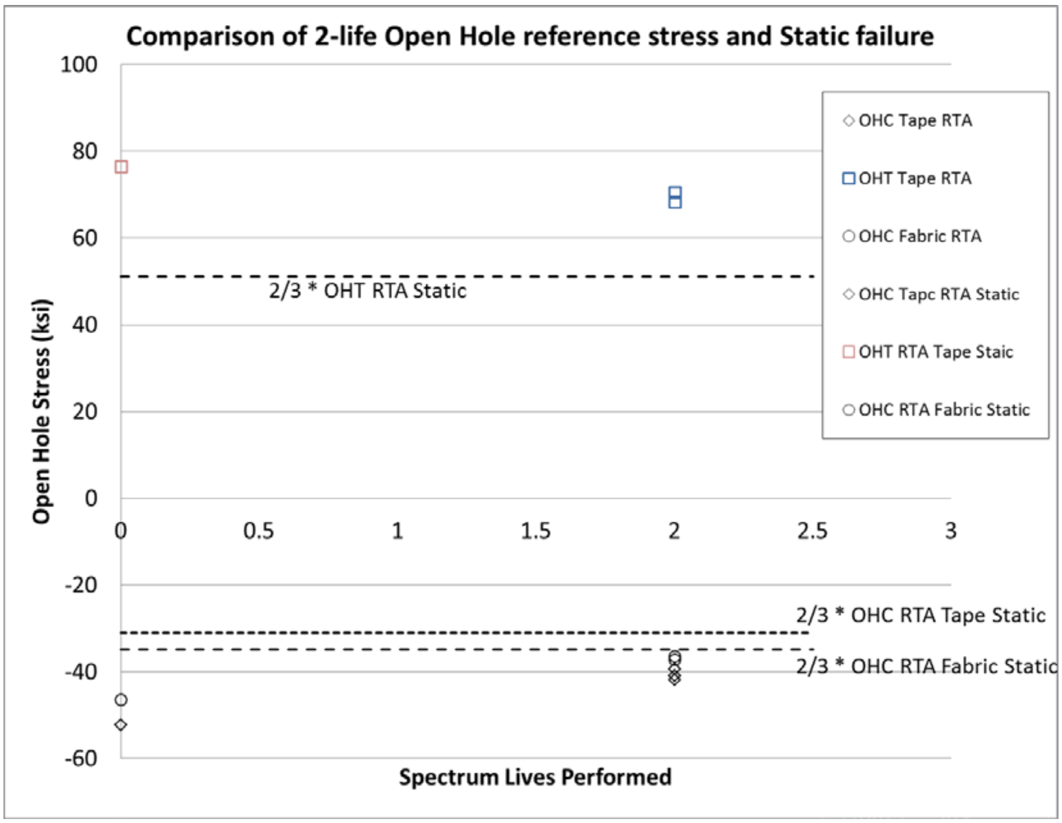


Figure 15.3-1 Comparison of Two-Life Open Hole Reference Stress and Static Failure

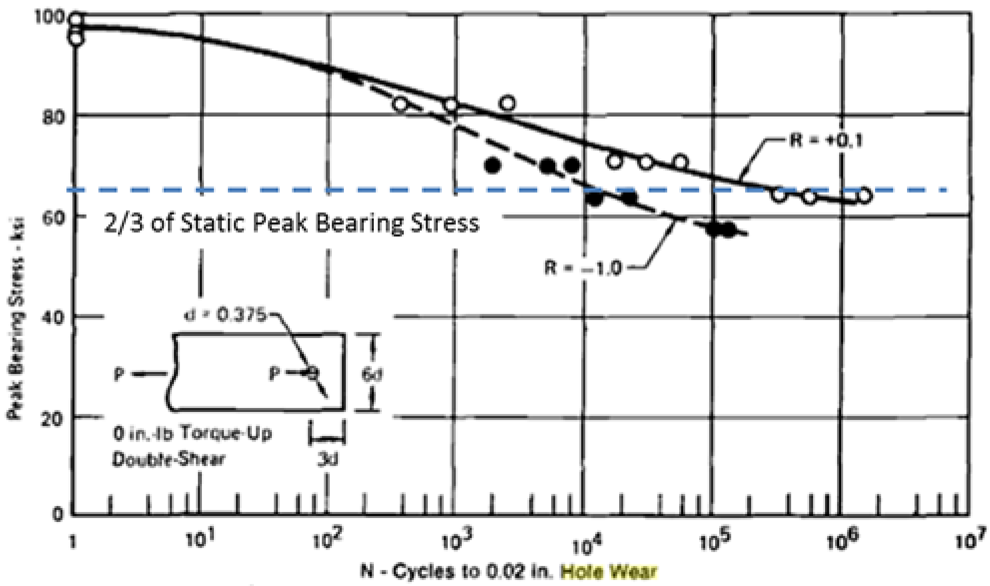


FIG. 6—*RTD baseline joint fatigue life.*

Figure 15.3-2 Example Allowable Bearing Stress versus Number of Cycles

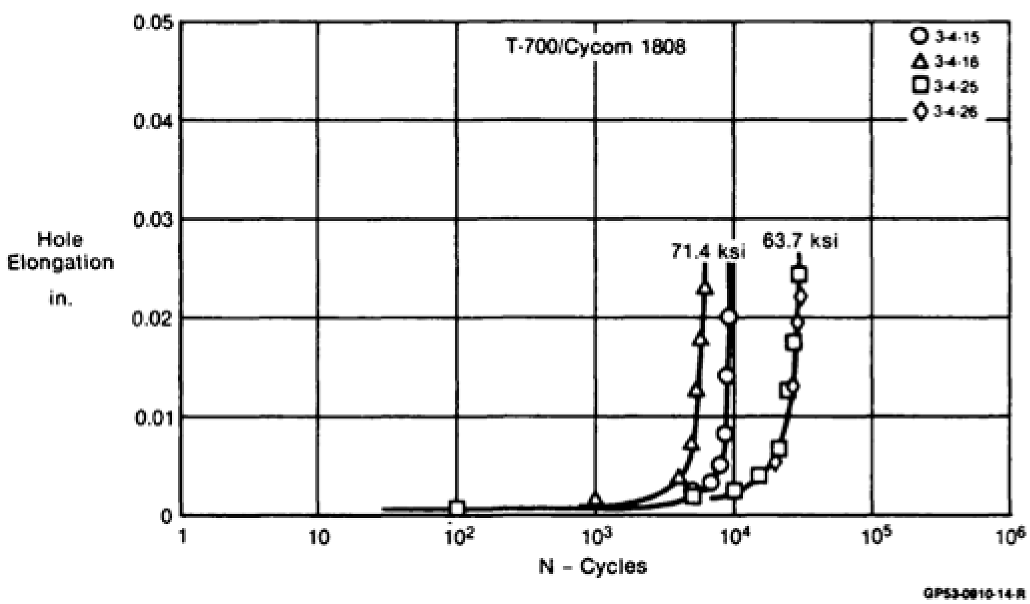
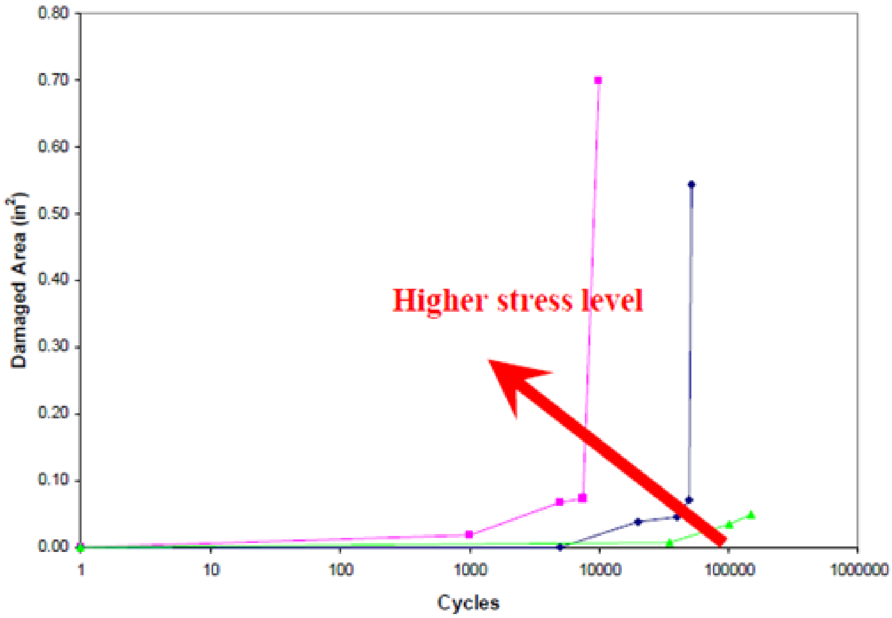


Figure 83. Pure Bearing Fatigue Hole Elongation Measurements: R = −1

Figure 15.3-3 Example of Hole Elongation versus Number of Cycles for Pure Bearing



(b) Damage Growth

Figure 15.3-4 Example of RTA Open Hole Tension Fatigue Damage Area versus Number of Cycles (AS4-Plain Weave Fabric, Quasi-isotropic)

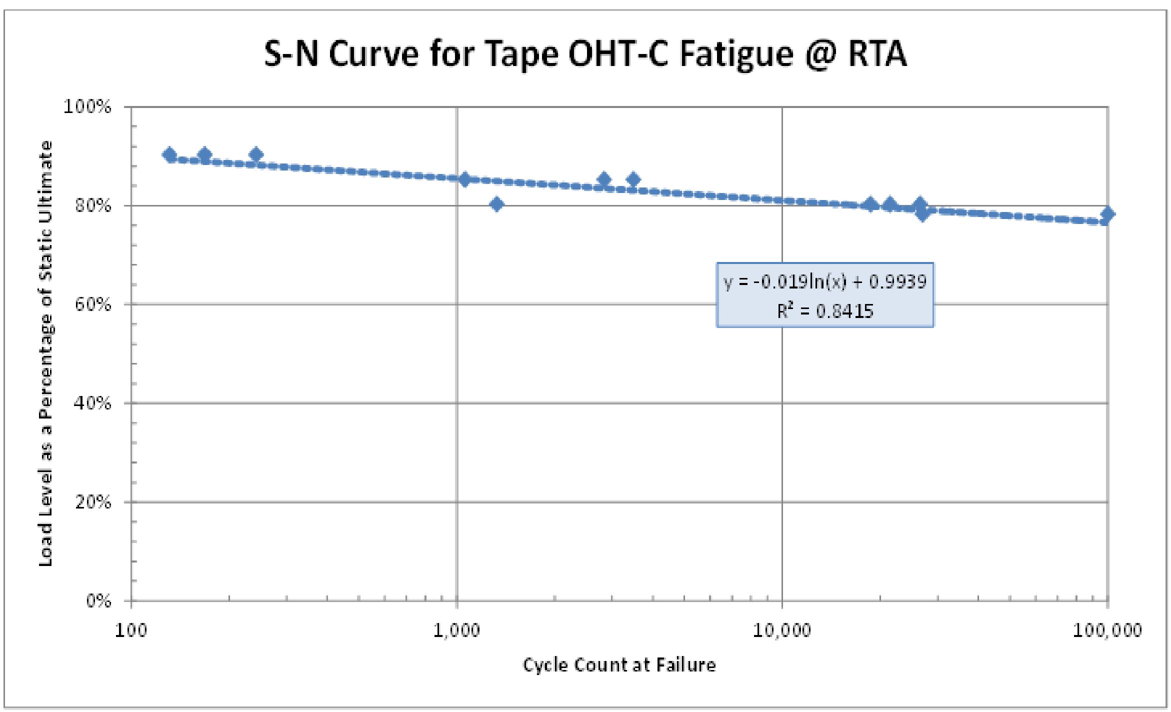


Figure 15.3-5 S-N Curve for Open Hole Tension, IM7/M65 Tape (RTA, R=-1.0)

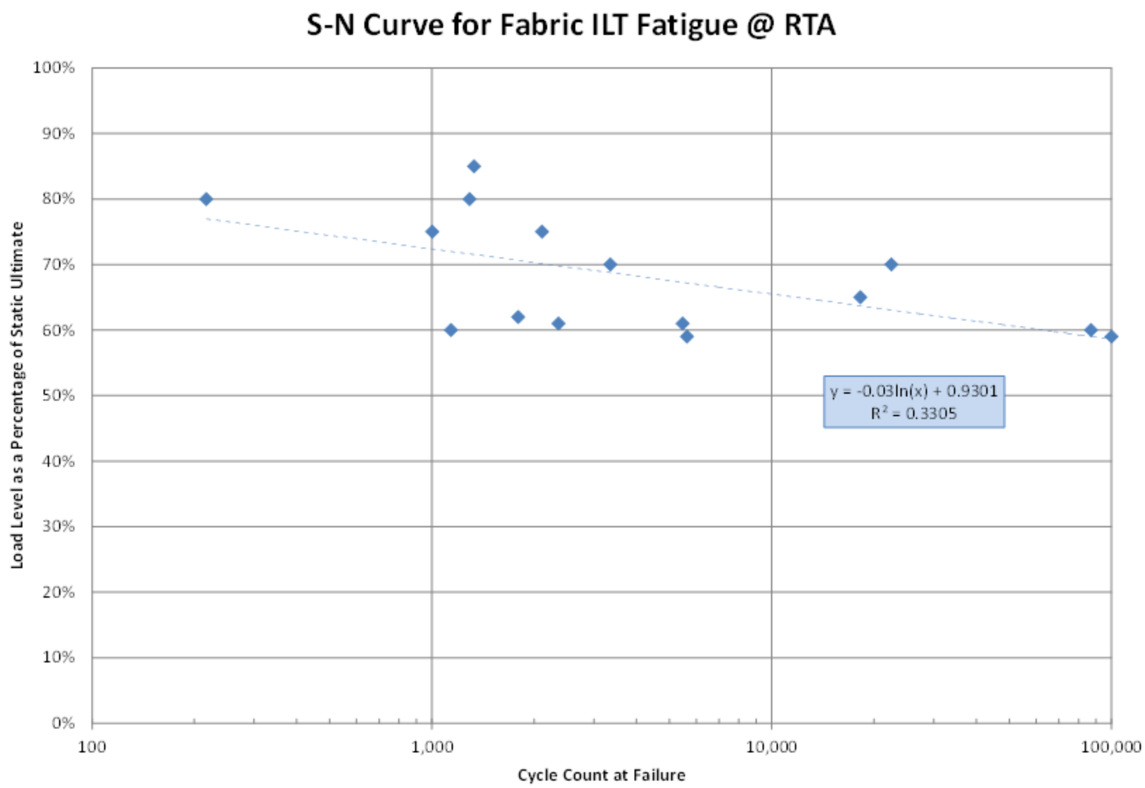


Figure 15.3-6 S-N Curve for Interlaminar Tension, IM7/M65 Fabric (RTA, R=-1.0)

15.3.1.2 Coupon Spectrum Considerations

There may be exceptions to the process for determining low-cycle verified static ultimate allowables. The vast majority of the airframe is correctly verified by using a conservative maneuvering spectrum applied to the static ultimate allowables. However there may exist structural details which are found to require additional scrutiny. These may involve spectrum effects not related to maneuver, such as door open/close events, pressure cycling, buffet or others. Door closing details can be especially severe as door closing preloads may impart pervasive tension preload stresses superimposed on pressurization and maneuver stresses. Additional point design cyclic tests may be required to certify these details. A building block approach should be used to address these.

Another consideration is the number of test lives performed to verify that no cyclic effects are applicable. The initial contract may require two lives. However, during the likely Service Live Extension Program (SLEP) additional lives may be required for verification. Additional allowables testing for additional lives will be costly and there always the chance that the same composite material formulation may not still be produced. During the Engineering and Manufacturing Development (EMD) phase considerations should be given to performing up to six lives of spectrum fatigue to prepare for future allowables adjustment. In similar fashion the baseline spectrum severity may increase beyond the original developed spectrum. During baseline spectrum updates care should be taken to ensure that the original composite test spectrum is still conservative.

15.3.1.3 Durability Impact Requirements

The myriad of possible low energy durability impact requirements from the JSSG-2006 is presented in Figure 15.3-7. Note the tool impacts are split between high and low probability of impact. In addition there are guides for hail and runway debris damage. This table will most likely be copied directly into a specification for all future aircraft composite structures programs. The amount of actual testing to cover all circumstances will be subject to negotiation.

The basic durability impact requirements are:

1) No visible damage with a 4 ft-lb impact using 0.5" diameter impactor. This can be thought of as the damage resistance aspect of the requirement.
2) No functional impairment for 6 ft-lb impacts using 0.5" diameter impactor; this would include no negative strength margins at a negotiated residual strength load level, P_{xx}, after two design lives of operation.
3) No water intrusion in high probability impact zones and no water intrusion in low impact probability zones after field repair. This requires that water intrusion testing be performed on impacted panels during and after stress cycling. Keep in mind that fuel leakage is a functional impairment and subject to higher pressures than water intrusion.
4) Hail damage must be considered.
5) Small runway debris must be considered.

Some clarifications/definitions based on past program experience:

- Low Probability Zone: An internal area/bay closed out and with infrequent access.
- High Probability Zone: External mold line, bay that will opened routinely, all areas that are not Low Probability.

JSSG-2006

Figures and Tables

Table VII. Low Energy Impact (Tool Impact)

Zone	Damage Source	Damage Level	Requirements in Addition to Paragraph 3.11.1
1 High Probability of Impact	- 0.5 in. diameter solid impactor - Low Velocity - Normal to Surface	Impact energy smaller of 6 ft-lbs or visible damage (0.1 in. deep) with min. of 4 ft-lbs.	• No functional impairment or structural repair required for two design lifetimes and no water intrusion • No Visible Damage from a single 4 ft-lb impact
2 Low Probability of Impact	Same as Zone 1	Impact energy smaller of 6 ft-lbs or visible damage (0.1 in. deep)	• No functional impairment after two design lifetimes and no water intrusion after field repair if damage is visible

Table VIII. Low Energy Impact (Hail and Runway Debris)

Zone	Damage Source	Damage Level	Requirements in Addition to Paragraph 3.11.1
All vertical and upward facing horizontal surfaces	Hail • 0.8 in. dia. • Sp. Gr. = 0.9 • 90 ft/sec • Normal to horizontal surfaces • 45 degree to vertical surfaces	Uniform Density 0.8 in. on center	• No functional impairment or structural repair required for two design lifetimes • No Visible Damage
Structure in path of debris	Runway debris • 0.5 in. dia. • Sp. Gr. = 3.0 • Velocity appropriate to system	N/A	• No functional impairment after two design lifetimes and no water intrusion after field repair if damage is visible

Figure 15.3-7 Impact Levels, Table VII and VIII from JSSG-2006, (Reference 15-2)

These requirements drive the minimum gages required for the composite structure in different zones. The actual implementation of these rules would be to develop and perform a rather extensive test matrix of impact levels and thicknesses to determine minimum gages for the different zones, including different combinations of carbon tape, carbon fabric (CSW and plain weave) and glass fabric. Thickness will be different for low probability versus high probability zones. Laminate thickness will vary whether the laminate is monolithic or bonded as a face sheet to honeycomb core. While actual values will depend on location, material system and product form, an approximate median minimum thickness is 0.05". Table 15.3-2 illustrates the sensitivity to some of these considerations.

Table 15.3-2 Factors Affecting Average Minimum Gage Thickness

Circumstance	Effect on Minimum Thickness
High Probability Zone	Increased Minimum Gage
Low Probability Zone	Decreased Minimum Gage
Fuel Boundary	Increased Minimum Gage
All CSW Fabric	Increased Minimum Gage
All S-2 Glass	Decreased Minimum Gage

Historically, the "no functional impairment" requirement of Item 2, of the basic durability impact requirements, has led to the most differences in various programs. The residual strength load level has varied from P_{xx} of 105% DLL to 120 %DLL. Fortunately for toughened bismaleimides (BMI) and epoxies the allowable compression strength of these durability impacts is rather high and probably will not present any severe adjustments to laminate thicknesses.

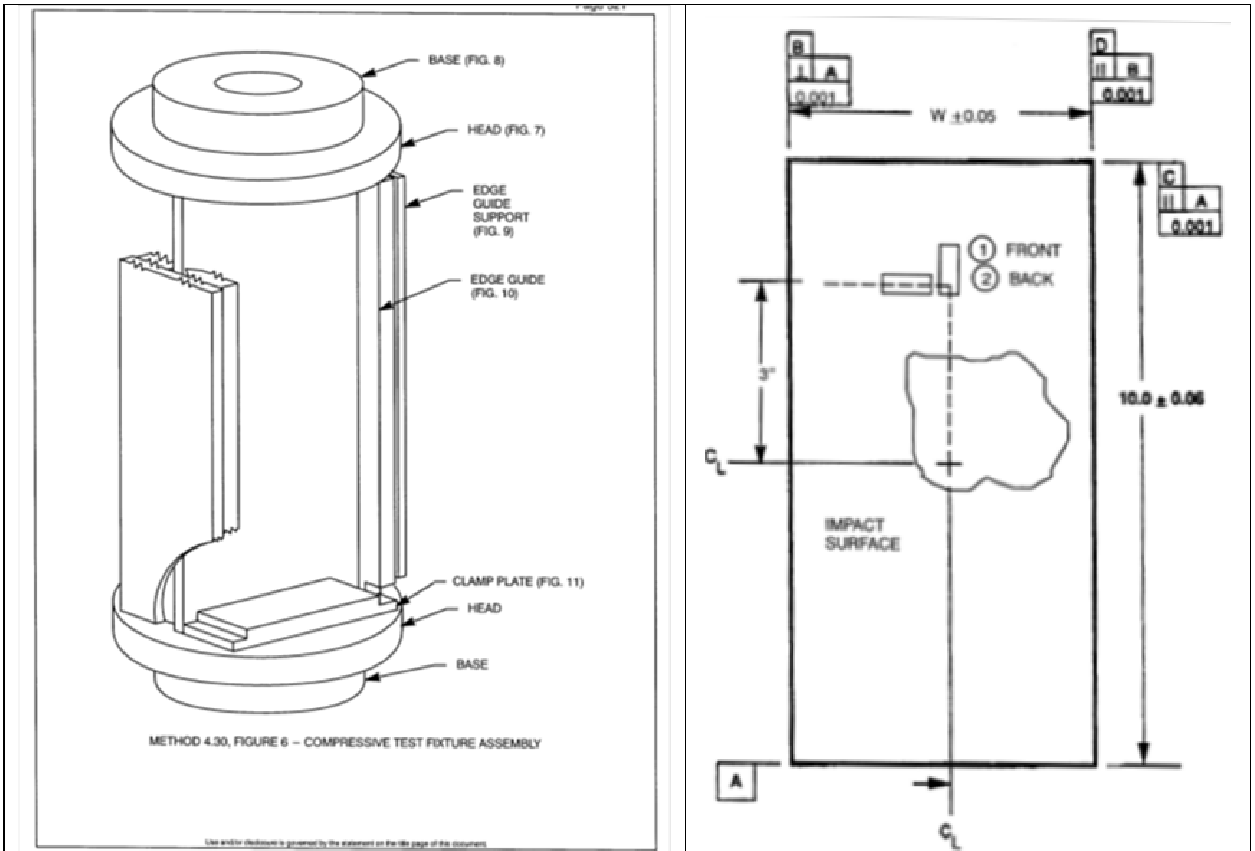
Normally, as the name implies, the JSSG-2006 is a guide for requirements contained in a negotiated contract. Consult the program Structural Design Criteria document for program specifics.

PM4056 Table 2.3-2 provides minimum thicknesses for both monolithic composites and sandwich face sheets required to meet impact damage requirements for low and high probability zones. Additional program specific testing should be conducted.

15.3.1.4 Durability Impact Allowables Testing

Legacy durability allowables were developed from test methods provided in Reference 15-4. The specification defines the coupon configurations, test fixture and testing procedures.

The lower energy levels of the 6 ft-lb impacts and the 0.5" diameter impactor tend to result in small delaminations (< 1" diameter) which, when tested, possess fairly high failure stresses. Figure 15.3-8 presents a diagram of the fixture. To minimize coupon buckling a relatively narrow coupon with lateral restraints is used. Note that the sides apply out-of-plane constraints but not rotational constraints for the buckling mitigation. Figure 15.3-8 also presents the coupon dimensions and a table with the coupon widths to be used for different thicknesses. Note that the coupon is machined with an initial width of 7" so that during impact the panel is of reasonable size. After impact the coupons is then machined down to the final tested coupon width.



Coupon Thickness (plies)		Coupon Width	Coupon Length	Impacting Size
Tape	Fabric	(in)	(in)	WxL (in)
16-24	16	2.5	10	7x10
32	--	3.0	10	7x10
--	24	3.5	10	7x10

Figure 15.3-8 Durability Impact Compression Test Fixture, Specimen and Dimensions (Reference 15-4)

Post processing of the strain data from the failed coupon must be performed in order to assure that buckling did not occur and cause an early failure of the coupon. Back to back strain gages should be plotted and evaluated for bending divergence.

15.3.1.5 High Cycle Composite Allowables

Aeroacoustic loadings apply vibrations at much higher frequencies than maneuvering-based low-cycle fatigue. These high-cycle fatigue effects can be quantified by utilizing a cantilever coupon with tuned properties which can be used on a shaker table and driven at their natural frequencies to obtain S-N data. Figure 15.3-9 presents an example of the “butterfly” cantilever high-cycle fatigue coupon. Note that the coupon can be mechanically fastened, co-cured or bonded to representative substructure. Tip weights may be used to tune for various mode shapes. Sample S-N data is presented in Figure 15.3-10. Note that the preferred method is to obtain data to at least 1.0×10^7 cycles.

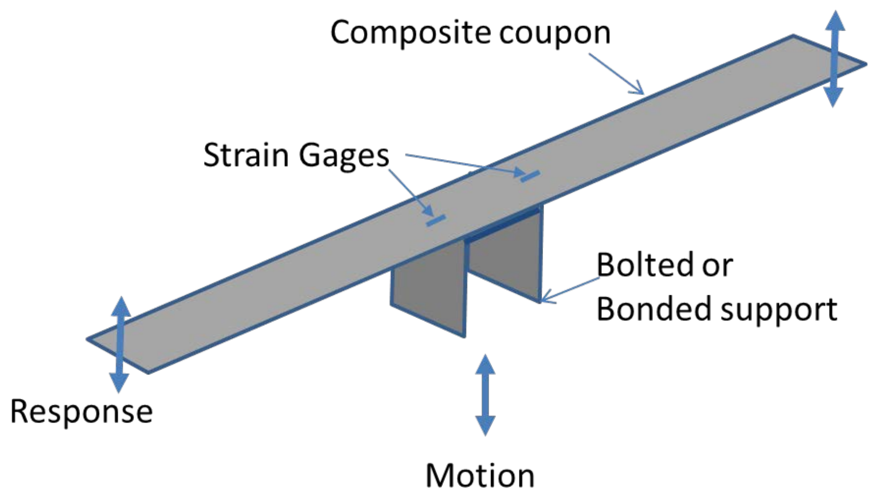


Figure 15.3-9 High-cycle "Butterfly" Coupon

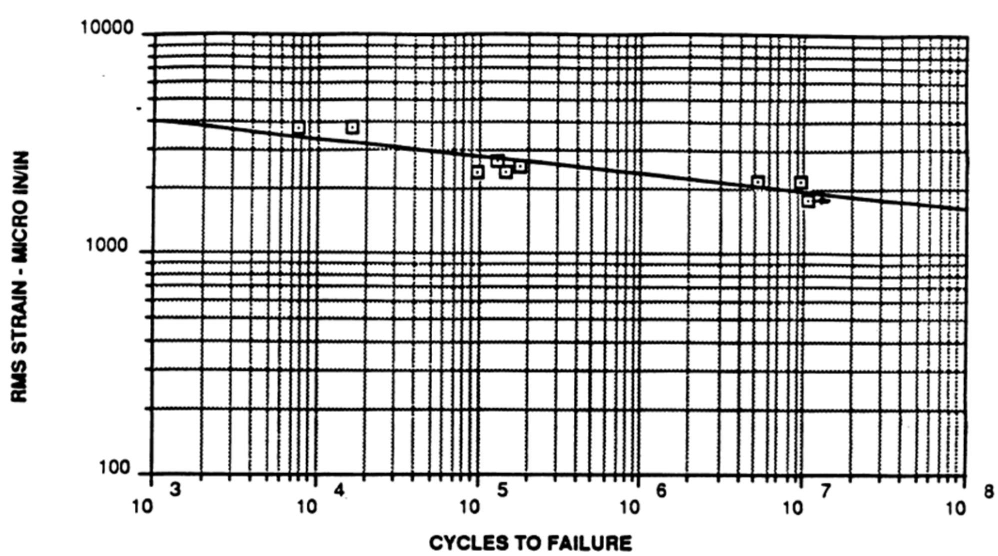


Figure 15.3-10 Desired Form of High-Cycle Fatigue Data

15.3.2 Fracture and Damage Tolerance Certification and Allowables

15.3.2.1 Fracture and Damage Tolerance Requirements

As with the durability requirements, JSSG-2006 (Reference 15-2) outlines requirements for damage tolerance. Usually the actual contracted requirements, contained in the program Structural Design Criteria document are far more specific than the JSSG-2006 contents. Common specific requirements will include:

- Specific types and severities of damage to be tolerated.
- Required service life and specific residual load level to be attained.
- Whether the allowables will require statistics in their formation or will rely on average values.

Specific types and Severities of Damage:

A typical description of the types and severities of damage to be considered is:

- Manufacturing defects of a certain size, *i.e.*, a single planar delamination. These could be from 1" to 2" in diameter.
- Damage defined as BVID (Barely Visible Impact Damage).
- Significant scratches, typically 0.02" deep and from 2 to 4 inches long.

The most problematic requirement is the second item; the intent is that the structure should be able to sustain damage not visible from a close examination up to a prescribed limit, typically 100 ft-lb. Primarily a CAI (compression strength after impact) concern, this requirement among all others has elicited the most discussion and tailoring. In the past the vagueness of the "Barely Visible" requirement resulted in a level of subjectivity that eventually drove more specific requirements. Further complicating the issue is that dent depths relax over time and exposure to environments.

On legacy programs, the Navy and Air Force did not have a common philosophy for CAI. Figure 15.3-11 presents the F-18E/F CAI requirements, taken from Reference 15-9, versus the F-22 requirements, taken from Reference 15-10, for stiff wing skin type laminate families of similar thickness. Note the layup percentages for the F-22 are 50/40/10 while the F-18 is a similar 48/48/4. The F18E/F requirement was for 0.5" diameter impactor while the F-22 was for 1" diameter. The residual load level for the F-18 was 115% DLL and 105% for F-22. Since the requirement was for "barely visible damage" an additional level of subjectivity also existed. The figure illustrates the wide range of results possible under the BVID requirement. The allowables are written at the maximum energies for each program and are normally formed to be able to compare to ultimate stresses. Hence the allowables to be compared to ultimate load are:

$$\sigma_{allow} = \sigma_{avg} \cdot \frac{150}{P_{xx}}$$

where
σ_{allow} is the allowable stress (psi)
σ_{avg} is the typical damage tolerance compression strength (psi)
P_{xx} is the residual load level (%DLL)

Current tailored JSSG-2006 guidelines have strived to reconcile the differences and provide more standardization. The 1.0" diameter impact is now the standard impactor. The "barely visible" requirement has been replaced by more definite requirements such as a simple 0.1" dent or an impact energy which results in a lower plateau of strength. An example of this lower end strength concept may be seen in Figure 15.3-12 where between 60 ft-lb and 100 ft-lb the relationship between strength and impact energy become insensitive.

If a simple dent depth is the driver for determining required impact energy a sizeable set of trial impact tests will be required. These consist of panels of the desired material type and product form mounted to impact fixtures, Reference 15-10 provides a good example of the process. Various levels of impact energy are applied to the panel and thickness measurements are taken. Impact energy values for the 0.1" depth can then be applied to actual panels to be tested for allowable strength.

Service Life and Residual Load Level

Typically two services lives after the application of damage are required. The residual load level is usually the higher of limit load or a probability based level, *i.e.*, probability of 0.05 over the life the aircraft. Some customers will insist on higher levels.

Allowables Formation using Statistics, Lower Bound or Mean Data

There will be much discussion as to whether the CAI allowables should include statistical effects to develop a knockdown, use the lower limit of the existing data or use a mean value. Normally the mean value of the data is used. The reasoning is that risk is mitigated by the unlikely event of a confluence of several unlikely events. Those being: (1) Impact in a very low margin area. (2) Impact at an energy just below the visible threshold thus escaping visual detection. (3) Impact occurring during the early life of the airframe and being cycled throughout the life of the aircraft, *i.e.* actually experiencing one full life of fatigue.

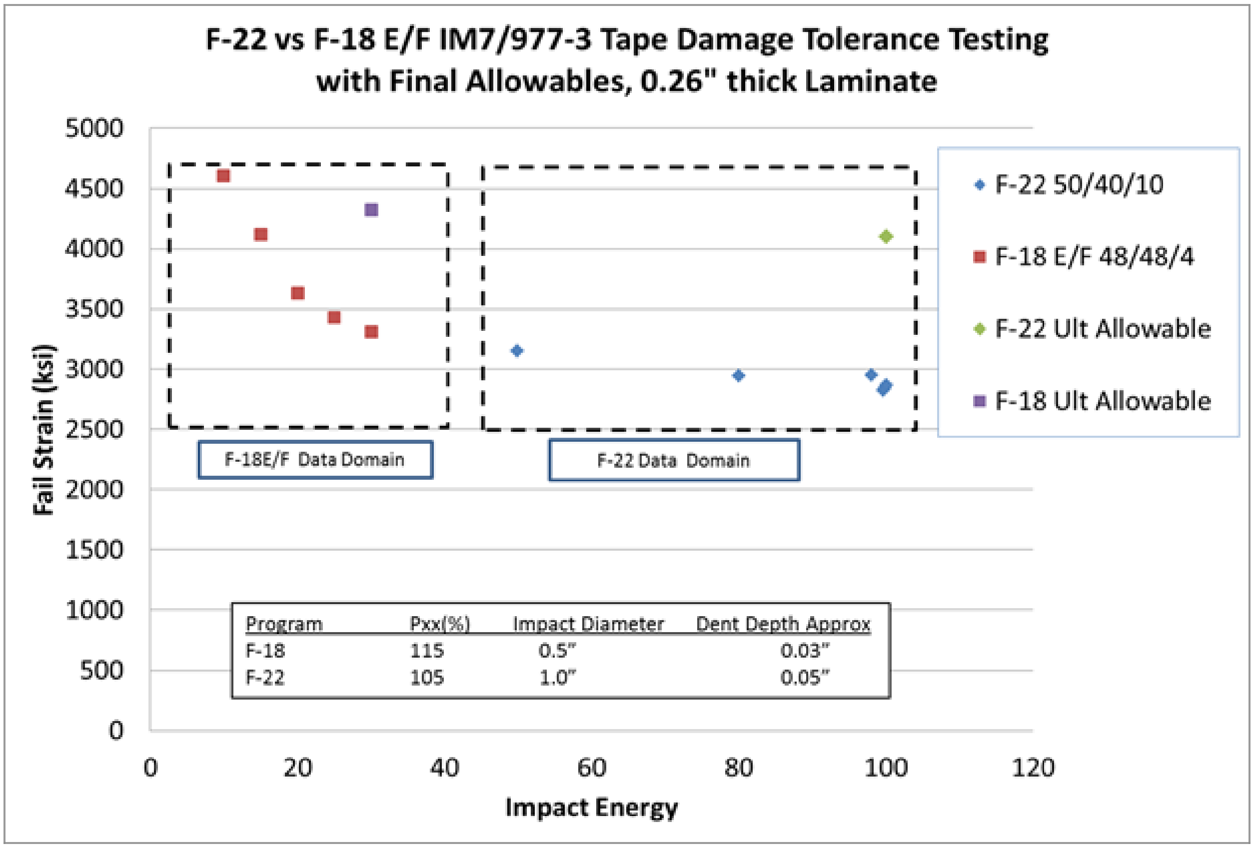


Figure 15.3-11 Comparison of F-18E/F and F-22 Compression after Impact Requirements

15.3.2.2 Coupon Spectrum Considerations

Spectrum considerations are identical to those discussed for durability requirements in section 15.3.1.2.

15.3.2.3 Damage Tolerance Allowables Development

The Compression Strength After Impact (CSAI), also referred to as Compression After Impact (CAI), allowables are at the heart of a composite damage tolerance allowables program. Damage allowables must take into account the material designation, product form, thickness, ply angles, impact energy, environment and cyclic effects. While some will claim to present a fracture mechanics based analytical method for predicting the compression strength of laminates with impact damage, the methods have not been found to be accurate across the design space need for production composite analysis. Current allowables used for legacy fighters involve using a complex regression equation that attempts to fit a large body of CAI testing. While both tension and compression strength after impact analysis is required, laminate residual tension strength is far greater and virtually never becomes the critical margin of safety.

The effects of impacts on composite laminates are a function of the fiber / resin system toughness, the thickness of the laminates and the in-situ panel support. Typical composites used on aircraft are made from toughened resin systems that are less sensitive to impact induced damage than early composite materials. However, potential for impact induced damage exists and is addressed by design requirements for durability and damage tolerance.

Typical trends for uniaxial Compression After Impact (CAI) strength as a function of impact energy are shown in Figure 15.3-12. As impact energy is increased, delamination sizes generally increases and failure stress decreases. The plot exposes several trends which are important

1) <u>Delaminations from impact can result in significant reductions in compression strength.</u> This trend is more pronounced when laminates are sufficiently thick (approximately 0.2 in thick or greater) and supported to preclude general instability failure. In the case of a clamped-edge 24-ply (0.13 inch thick) laminate the 6ft-lb specimens retain 55% of the unnotched compression strength which is better than the reduction in strength required by the need for repairabilty, as illustrated by the ¼ in. open hole curve.
2) <u>Strength with delaminations can be less than strength with equivalent sized holes.</u> Above 10 ft-lbs the strength is less than the strength with a 0.25 in. open hole. Above an impact of 25 ft-lb the strength predicted with a hole diameter equal the diameter of the impact delamination is greater than the equivalent-size delamination. The buckling sublaminates which impart significant prying and mode-I crack growth result in large reductions in compression strength.
3) <u>Above a certain level impact, a through-hole will result.</u> In this example any impact above 60 ft-lbs will result in a through-hole surrounded by considerable delamination. At this point the lower strength limit of the damaged panel is found. This figure demonstrates the lower limit runout (above 60 ft-lb) that occurs as impact energy is increased. Using that definition (lowest possible strength) the allowable would be written at 60 ft-lb.

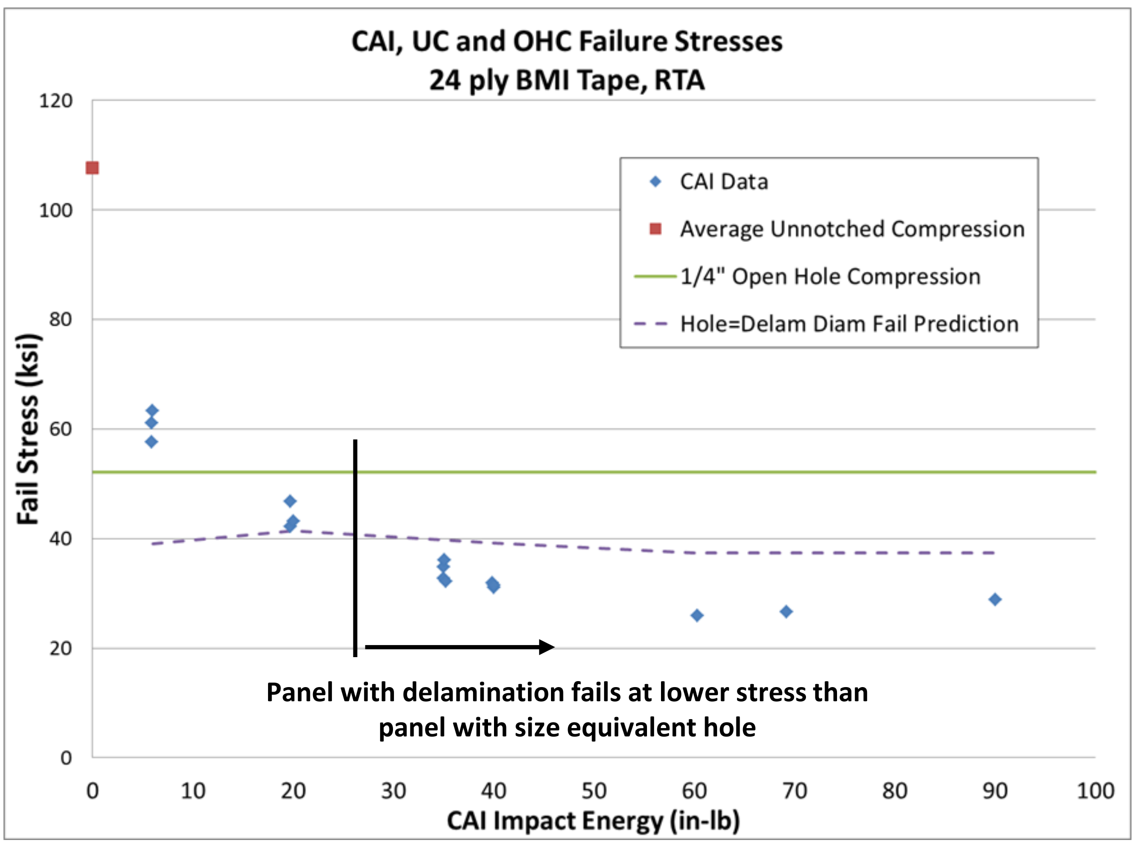


Figure 15.3-12 Example of Compression after Impact (CAI), Unnotched Compression (UNC), and Open Hole Compression (OHC) Failure Stresses

The test data that has been presented is for flat panels with uniaxial loading for a single part thickness. Note that the damage area due to impact is a function of the material as well as the part thickness. Additionally, curved panels or initial geometry will complicate the failure mechanism. Point testing may be required to certify low margin areas where significant curvature exist or where loads are generating complex in-plane and bending loads, *i.e.*, postbuckled panels.

15.3.2.4 Compression after Impact Strength Regression Equations Development

Legacy CAI allowables were developed using regression equations applied to a robust set of test data. Typically a variety of laminate thicknesses, laminate stacks and energy impacts are required. Here is a suggested list of considerations when planning a test matrix for CAI allowables.

1) Choose thicknesses that span the need for your project, typically low end damage tolerance thickness will be approximately 0.085". Typical high end is 0.26" but certainly may be greater. Keep in mind that typically above 0.26" thickness the energy requirement will always maximize to 100 ft-lb and at some point strength will increase as greater thickness resists ILS fracture resulting in smaller delaminations.
2) Laminate families should populate most of the %0 versus %45 carpet plots. If regression is used to quantify CAI allowables it is undesirable to extrapolate.
3) Bending after impact tests (BAI) are used to adjust allowables for through-thickness gradients in stress.
4) Room temperature ambient (RTA), Cold Temperature Ambient (CTA) and Elevated Temperature/Wet (ETW) will be required to verify environmental effects.
5) Cyclic effects will have to be considered. Spectrum loaded CAI will need to be performed with a conservative spectrum. The correct negotiated residual stress needs to be applied after each lifetime of cycling.

The basic form of an example CAI regression stress prediction equation is

Stress = (OHC typical predictor) • (thickness adjustment) • (impact energy adjustment) • (laminate stack adjustment)

The equations are used for both damage tolerance impacts and durability 6 ft-lb impacts. For the durability impact, the impact energy adjustment is eliminated as the impact energy is always 6 ft-lb. The laminate stack adjustment can adjust for laminate in-plane orthotropic aspects (E_x, E_y, G_{xy}) and in some cases stacking details which result in varying bending properties (D_{11}, D_{22}, D_{12}).

More specifically a symbolic equation would be:

$f = \left(C_1 \cdot E_x + C_2 \cdot E_y + C_3 \cdot G_{xy}\right) \cdot$	Open Hole Regression
$t^{C_4} \cdot$	Thickness Adjustment
$Energy^{C_5} \cdot$	Impact Energy Adjustment
$\left(C_6 \cdot E_x + C_2 \cdot E_y + C_8 \cdot G_{xy}\right) \cdot$	Additional Laminate Adjustment
$\left[\left(\frac{D_{11}}{D_{11s}}\right)^{C_9} + \left(\frac{D_{22}}{D_{22s}}\right)^{C_{10}} + \left(\frac{D_{12}}{D_{12s}}\right)^{C_{11}}\right]$	Bending Stiffness Adjustment

where
f is the stress (psi), no environmental effects are represented yet, assume RTA testing dominates
C_1 through C_{11} are regression constants
E_x, E_y and G_{xy} are the laminate in-plane moduli (psi)
D_{11}, D_{22}, and D_{12} are the laminate bending constants
D_{11s}, D_{22s}, and D_{12s} are "smeared" bending terms defined as the D_{xx} terms calculated with a single lamina at 0 deg having the properties of the representative laminate stiffness and the thickness of the laminate.

The D_{xxs} terms give a measure of the biasing of certain piles to the outer surfaces of the laminate. For example for a 25/50/25 tape laminate where the 0 degree fibers are concentrated on the outside surface, the D_{11} term will be higher than the same laminate where the 0 degree fibers were uniformly distributed (the D_{11s} term).

Constants C_1 thru C_{11} would be solved by regression. This of course implies that a robust set of test data is available for the regression task.
The next steps are those which transform the RTA prediction stress to a program allowable stress:
1) The regression values should be tied to the desired threat level. This could be the subjective visibility or a set dent depth. The threat level will be applied as an impact energy level (Energy). Trial impact panels are usually performed to determine the correct impact energy level for the analysis.
2) Apply appropriate environmental reduction factors, *i.e.*, elevated temperature/wet.
3) Cyclic effects testing must be performed to quantify the reduction for required life testing.

An example of uniaxial CAI test data for relative failure stress versus impact energy for different thickness and layups is presented in Figure 15.3-13, taken from reference 15-3. Note the characteristic decreasing strength with higher impact energy and sizeable scatter in the data. An allowables line has been developed using 0.06" dent depth as the requirement. Note for this material as thickness increases the allowable also increases. This is not a general trend and can be the opposite.

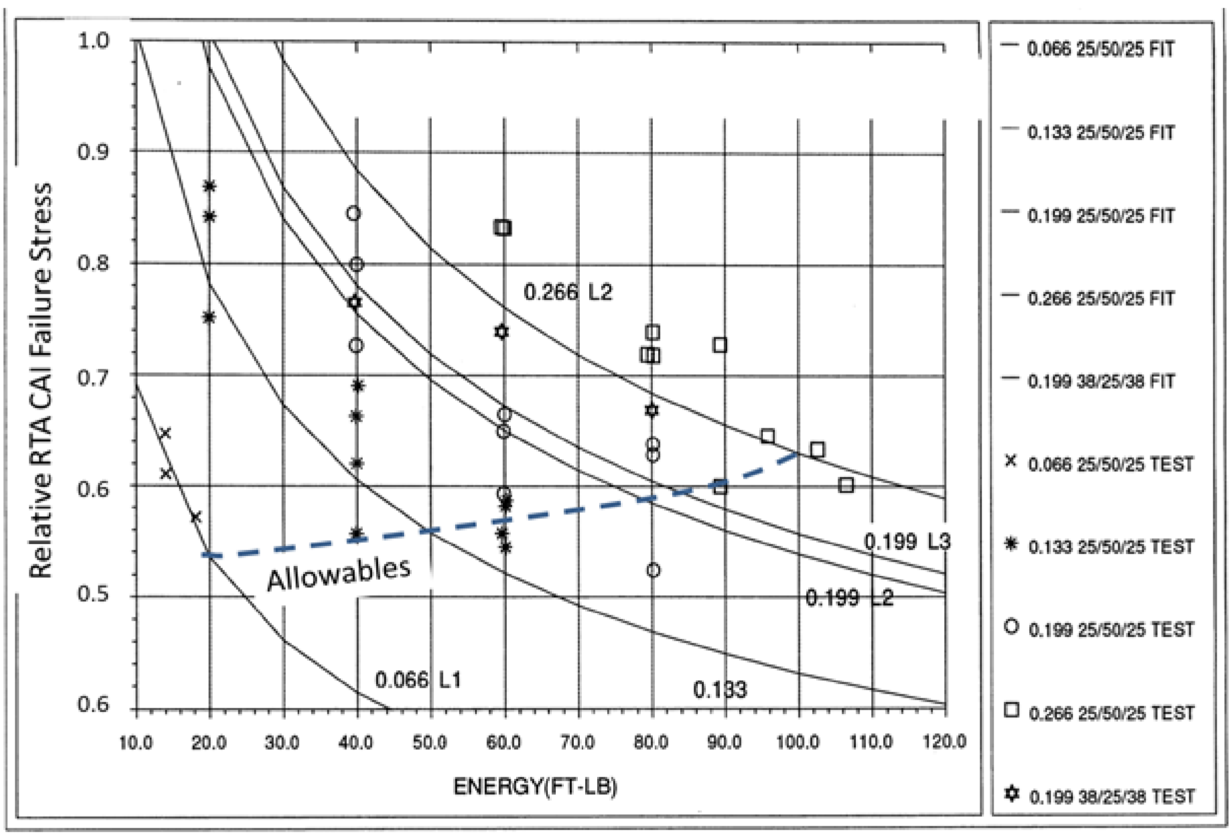


Figure 15.3-13 IM7/5250-4 Carbon/BMI Fabric Compression after Impact (CAI) Failure Stress, Predictions, and Allowables

These uniaxial allowables need to be adjusted for real world internal loads such as bending effects and biaxial loading.

Bending after Impact (BAI) stresses are taken into account by performing pure bending tests for plates of varying thickness. As is common for other modes of failure, thin plates will fail at a higher maximum bending stress than the uniform stress in an identical plate experiencing uniaxial load. As the plate becomes thicker, the bending failure stress approaches that found in the uniaxial case. The bending factor is the bending stress at failure divided by the uniaxial failure stress for an identical panel with identical impact energy and maximum principal stress direction. The form of the interaction is a bilinear curve as presented in Figure 15.3-14. This is the factor which is used in the IDAT/CDADT program.

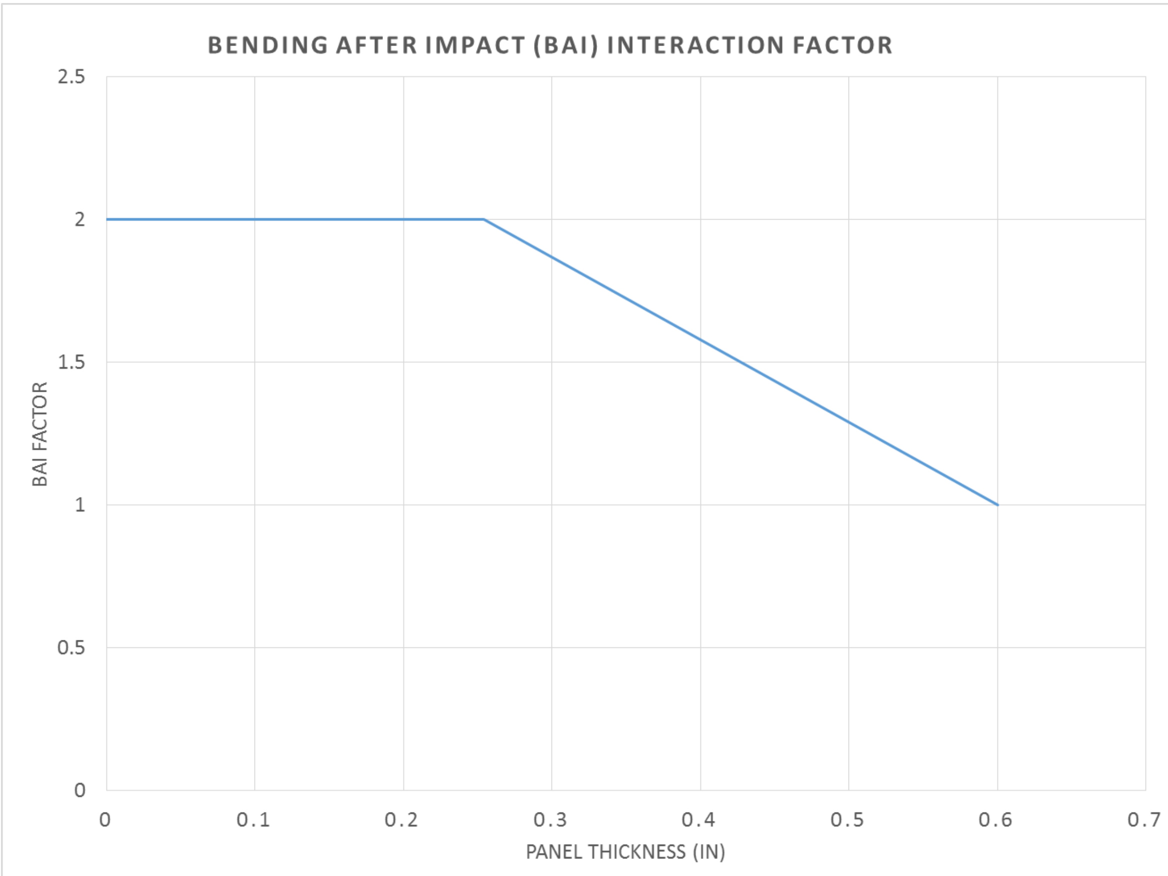


Figure 15.3-14 Bending After Impact Factor Typical Implementation

The margin of safety for axial and bending interaction can be calculated from the following equation

$$M.S. = \frac{\varepsilon_{uniaxial-CAI}}{\varepsilon_{axial} + \varepsilon_{bending}} - 1$$

Where
ε_{axial} is the strain due to the in-plane loading (με)
$\varepsilon_{bending}$ is the outer surface fiber strain due to bending divided by the bending factor (με)

Additionally the effects of biaxial stresses must be considered. Note that much controversy surrounds the effects of biaxial loads on the strength of delaminated composite panels. Little test data exists that demonstrates the effects of biaxial loads. Biaxial effects in CDADT are taken into account by utilizing a postbuckling elliptical plate Mode I[2] crack growth method. The method predicts the strain energy release rate, G_I, values for the uniaxial stress state and

[2] Mode I opening mode: a tensile stress normal to the plane of the crack

the biaxial stress state. This biaxial failure ratio is then applied to the failure prediction from the regression equation. The predicted method has the effect of increasing the failure stress for transverse tension stresses and decreasing the failure stress for a biaxial compression state. Conceptually the biaxial effect should be idealized as a quantity which trends the buckling stress for an elliptical plate, *i.e.*, as buckling stress is increased the critical failure stress increases, as biaxial compression reduces buckling stress there is a reduction in the critical failure stress.

15.3.2.5 Damage Tolerance Impact Allowables Testing

There are several test methods for determining damage tolerance of composite panels with impact delaminations. The most common non-proprietary test is the ASTM D7137 (Reference 15-12) presented in Figure 15.3-15, which uses a 4" wide a 6" long coupon. The fixture works well for smaller delamination sizes and is a commonly used fixture by composite material manufacturers. Vendors will publish results using this fixture which are useful for comparing damage tolerance properties between different composite materials. However the 4" wide coupon was judged too narrow for certain higher thicknesses and impact energies as finite width effects would be too severe. Major airframe OEM's have determined the coupon to be too small to fully characterize the requirements which are contained in their contracted specifications. For example a 0.26" thick Carbon/Epoxy panel with the maximum required 100 ft-lb impact could easily develop a 2" wide delamination. LM damage tolerance allowables were developed from test methods outlined in Reference 15-12 and utilize a fixture capable of testing a 10" wide coupon.

D 7137/D 7137M – 05$^{\epsilon 1}$

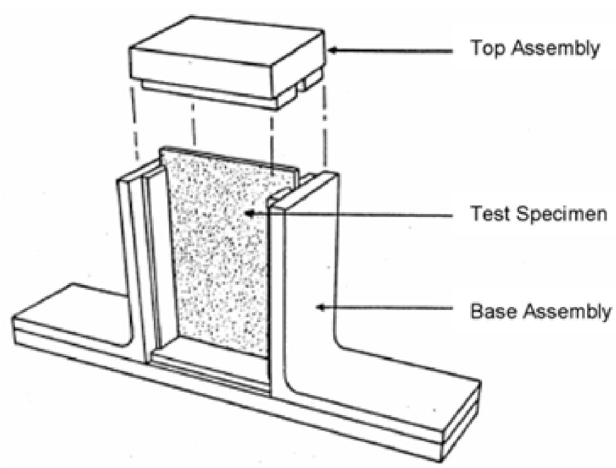


FIG. 1 Schematic of Compressive Residual Strength Support Fixture with Specimen in Place

Figure 15.3-15 Damage Tolerance Impact Compression Test Fixture, ASTM D7137 with 4" W x 6" L Test Coupon (Reference 15-12)

Figure 15.3-16 presents the damage tolerance test fixture which receives the 10"W x 12" L uniaxial compression loaded damage tolerance coupon. The coupon is sandwiched between a front and rear plate. Note the windows which are machined into the plate. The windows allow for strain gage placement and allow the impact area sublaminates to freely buckling out of plane. Several window sizes are available depending on delamination size. Thinner laminates will have small delaminations allowing smaller windows; the smaller window also precludes panel buckling.

The restraining process and windows in the clamping plate allow buckling sublaminates to fail locally while preventing overall panel buckling. Figure 15.3-17 presents the uniform strain and abrupt failure of an acceptable compression coupon. Figure 15.3-18 presents the early buckling which applies non-uniform loads to the impact zone and tends to result in early failure not representative of a correct failure mode. A review of strain gage values is always prudent to assure correct coupon performance.

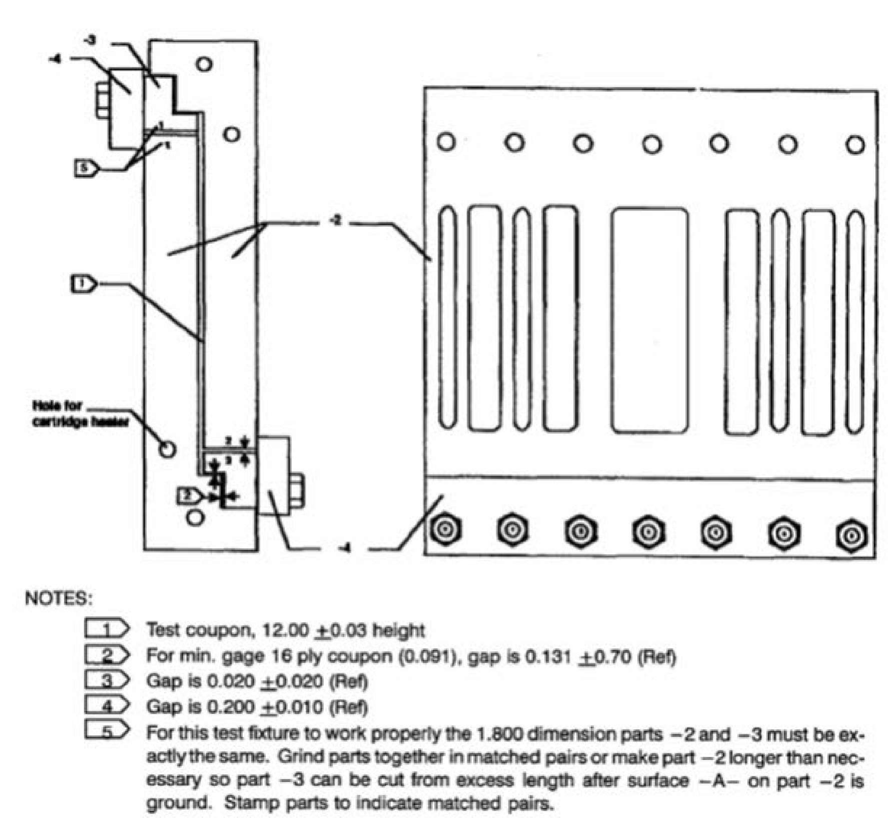


Figure 15.3-16 Compression End-Loaded Damage Tolerance Test Fixture, Reference 15-4

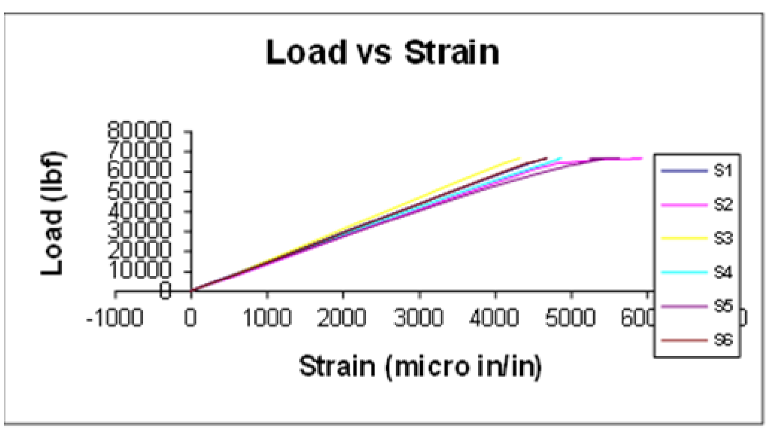


Figure 15.3-17 Acceptable Strain Gage Readings During Compression After Impact Testing

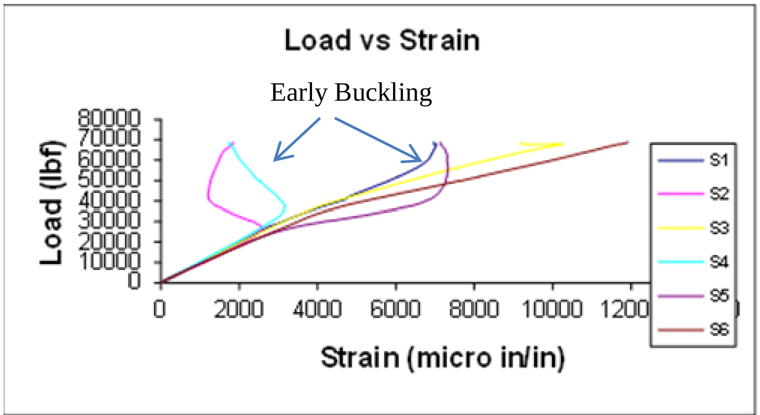


Figure 15.3-18 Unacceptable Strain Gage Readings During Compression After Impact Testing

15.4 General Design Guidelines

PM4056 Section 2.3.2.3 provides some guidelines for use in design of robust damage resistant structures. In general the guidelines for design details to aide durability and damage tolerance are consistent with those for all static modes of failure.

- Balanced symmetric laminates are desirable
- Avoid grouping like ply orientations together to minimize matrix microcracking
- Use the least strength critical ply at the OML surface to minimize the degradation in strength in the event of damage.
- Use bi-directional woven fabric (see next item) for the outer surface plies to increase surface toughness and increase abrasion resistance.
- Add a ply of a ductile material or fiberglass on the back side of graphite composite laminates to limit back side damage in the event of impact and reduce handling damage
- Fasteners are often introduced into cocured or bonded structure where unavoidable peel stresses are developed. These fasteners keep opening-mode prying stresses lower and are often used in flanges of highly loaded cocured stiffeners or runouts in bonded structure. While a correctly spanned building block program should address this issue, the installation of “anti-peel” fasteners is often prudent practice given the unknowns that may affect structure during the service life.

The last three suggestions can be satisfied by using a bi-directional woven fiberglass ply on the outer surfaces of critical parts. The fiberglass also helps in prevention of corrosion of adjacent metallic structure.

15.5 Composite Durability Analysis

Analysis of composite structure for durability is performed using static ultimate applied loads with material properties which include the cyclic effects, thus a low cycle, spectrum-based durability type approach is not necessary. The details of the actual analysis approach used is program specific and is governed by the program Structural Design Criteria Document.

Durability impact analysis is performed in the IDAT/CDADT program at the same time as the Damage Tolerance analysis is performed. Alternately, some programs may adopt an IDAT/IBOLT approach using a filled hole tension/filled hole compression analysis in lieu of the IDAT/CDADT analysis. Consult program specific guidance.

15.6 Composite Damage Tolerance Analysis

IDAT/CDADT performs the durability and damage tolerance impact analysis necessary for aircraft parts. Various material system/product form combinations have allowables available.

This analysis methodology takes the following items into consideration:

- Durability impact allowables are based on 6 ft-lb with 0.5" diameter impactor.
- Damage tolerance impact allowables are presented at various energy levels up to 100 ft-lb in addition to subjective 5 ft. visibility allowables (approximately 0.06" dent depth).
- The allowables are based on monolithic laminates. Laminates over honeycomb core will behave differently depending on the density and type of the core. In some case smaller delaminations will occur with larger core/skin disbonds. These allowables are generally obtained from element testing in a custom building block program. Refer to Section 15.7.5.
- Note that S-2 glass/BMI allowables are not included. These systems typically exist bonded to honeycomb and are based on custom empirical tests based on actual designs used in a structures program. Consult the Program allowables representative for guidance.
- Allowables are usually based on a conservative (fighter) maneuvering spectrum. If the spectrum deviates substantially from this standard, custom adjustments may be needed.
- Ensure that the aircraft program is observing the correct damage resistance criteria, *i.e.*, minimum gage requirements. Occasionally damage resistance requirements more severe than the standard 6 ft-lb impact criterion will be required such as no damage from a 10 ft-lb impact. Requirements outside of those listed in the JSSG-2006 should be studied very carefully prior to acceptance as they may impose harsh requirements on the airframe.

15.7 Durability and Damage Tolerance Issues Associated with Various Structural Details

15.7.1 Panel Acreage and Holes/Corners

Typically the durability and damage tolerance allowables are to the applied to the acreage, *i.e.*, unsupported areas between spar/rib caps and other hard points. The existence of substructure providing support will absorb impact energy and preclude the development of delamination, thus no analysis is required over substructure . Applying the compression-based allowables to the edges of cutouts where stress concentrations occur can be problematic for several reasons.

- Stress concentrations around cutouts tend to be fairly localized, while durability impact delaminations required for failure are on the order of at least 0.75" in diameter. There needs to be an area of stressed structure at least as large as the impact delamination for the delamination failure mechanism to occur. It is not correct to simply compare durability stress allowables to the peak compression stresses in the cutout. As an alternative, some programs elect to compare the durability allowables to the stresses at a distance of some fraction of the hole diameter, usually half the hole radius.
- Damage tolerance impacts are larger than durability impacts, usually at least 1.25" in diameter. As with durability impacts it would not be correct to simply compare damage tolerance allowables to the peak compression stresses in a cutout area. Consult program guidelines for applying damage tolerance allowables at cutouts.
- Delamination damage from impact is a complex phenomenon involving impactor type, impact energy, material properties and boundary conditions. Impacting free edges including free edges of cutouts will result in a different response than a centered impact on a supported monolithic panel representing acreage. Often the increased flexibility of free edges results in increased elastic deflection at the impactor with little or no damage. Trying to develop a "Barely Visible Impact Delamination" can be especially tricky. Often what happens is that impact energy is increased with no delamination until at some level the impact causes fiber damage and a through-hole. There may be no impact level capable to producing delamination damage

capable of being predicted by Lockheed Martin methods. This same phenomenon is why doing damage tolerance on full scale components is risky, the chance of impacting at greater than the BVID level is quite high and could result in larger than desired damage and lower than applicable strength. Performing trial impacts on representative component structure to quantify correct impact energy can reduce risk but will also increase costs

15.7.2 Bolted Joints

Bolted composite joints have several possible failure modes from cyclic loading, fastener shank/head/tail and the composite adherends; however, most have critical modes of failure in the fastener as opposed to the adherends. Much like the notched modes (open-hole, filled-hole, *etc*.) pure bearing fatigue is tested with a conservative spectrum to determine if static ultimate allowables may be used without cyclic reductions. Current design practice does not apply delamination-related strength reductions to bolted bearing strength. There may be reductions applied for effects of defects during salvage activities, but that is a separate issue which is addressed in PM4056 Section 19. For composite bolted joints, the IDAT program SFBOLT is a semi-empirical program used to predict bolt fatigue life.

15.7.3 Bonded Structure

Bonded structure can be problematic as the disbonding of bond lines due to impacts is difficult to predict. The latest guidelines in JSSG-2006 require bonded joints to have at least limit load residual strength in the presence of a complete disbond (Reference JSSG 2006 Section 3.10.5). This guidance refers to a fail-safe damage tolerance approach, which require bonded structure to be designed for multi-load-path or single-load-path fail-safety. This results in a requirement for the use of crack arrest design features, such as fasteners, in order to limit the failed bond ligaments to lengths that can sustain the required residual strength.

It is recommended that structure be designed with a sufficient number of crack arrest fasteners such that no inspections are required. That requires sufficient fasteners to carry residual load, P_{xx}, without failure of the fasteners or any failure modes of the disbonded laminate, including pull-through, bearing/bypass and interrivet buckling. Program design criteria may provide further clarification and guidance.

15.7.4 Co-cured Structure

Co-cured structures often have stiffeners, spars and ribs integrally cured to the skins. While not considered acreage, impacts directly over substructure can result in loss of substructure integrity particularly at run-out or areas of discontinuity. Trial impacts during certification can be useful in showing that the delaminations will be small or that peel fasteners or fittings are required. Custom tests are usually performed during the building block testing. A cut-up and test program for part-family certification should include trial impacts over cocured substructure to assure that damage is limited and does not result in large scale skin/stiffener separation and hence significant strength loss. It is important to investigate the effects on full-sized parts, supported in a representative manner, perhaps as part of a full-scale component test. The substructure stiffness is a big contributor to the behavior and the resulting damage.

15.7.5 Honeycomb Structure

Impacts on composite skins bonded to core behave differently than impacts applied to monolithic skins. The core may be honeycomb or a solid core such as syncore. Damage may be a combination of skin delamination, skin/core disbonding and core crushing. Due to the supportive nature of the core on skin, small delaminations and induced disbonds result in different allowable compression strengths than would result from monolithic laminates. An allowables program that includes core will usually require point tests for representative skin/core combinations. The allowables program will need to include the different combination of core types and densities and applicable facesheet materials and thicknesses. Strength as well as minimum-gage thickness for water intrusion resistance should be quantified.

Some of the glass cores such as HFT and HRP can be quite brittle and results in large core fractures. The Kevlar and Nomex cores are more flexible and can result in smaller damage. All core/skin type combinations need to have trial impact, water intrusion and strength tests to fully develop the allowables.

15.8 Analysis Procedures

15.8.1 Lamina and Laminate Allowables

Allowables for unnotched and notched failure modes should include maneuver cyclic load effects. Unlike metals there is no spectrum-dependent cycle-by-cycle analysis for durability or damage tolerance. The composite allowables should already be reduced to account for these effects. If the applicable usage spectrum differs greatly from the maneuver spectrum, special point-specific allowables may need to be generated. Consult your program structural certification group.

15.8.2 Durability and Damage Tolerance Allowables

Several of the programs contained in the analysis tools suite IDAT have programs for the determination of allowables for specific applications. Program-specific Structural Design Criteria documents and analysis guidance should also be consulted before starting an analysis.

IDAT contains the CDADT module which allows calculation of the durability and damage compression and tension after impact allowables. See the CDADT Users Guide, Reference 15-13 for more detail. There are program specific versions which contain several customer mandated differences, *i.e.*, differing residual strength, P_{xx}, values. Special new versions may be required based on customer requirements.

15.8.2.1 IDAT/CDADT and Durability Allowables Calculation

Figure 15.8-1 presents the IDAT/CDADT main dialog. Under “Criticality” choose the “Maintenance Critical Criteria” option to calculate durability impact allowables. The impact levels are set by the CDADT program based on the criticality level. The impact level for “Maintenance Critical” is 6 ft-lbs. See Section 15.3.1.3 for discussion.

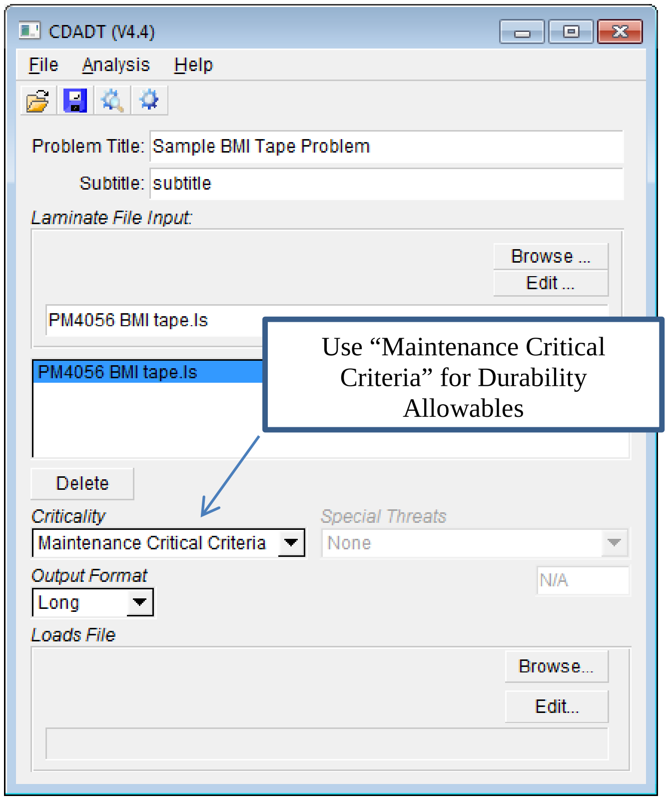
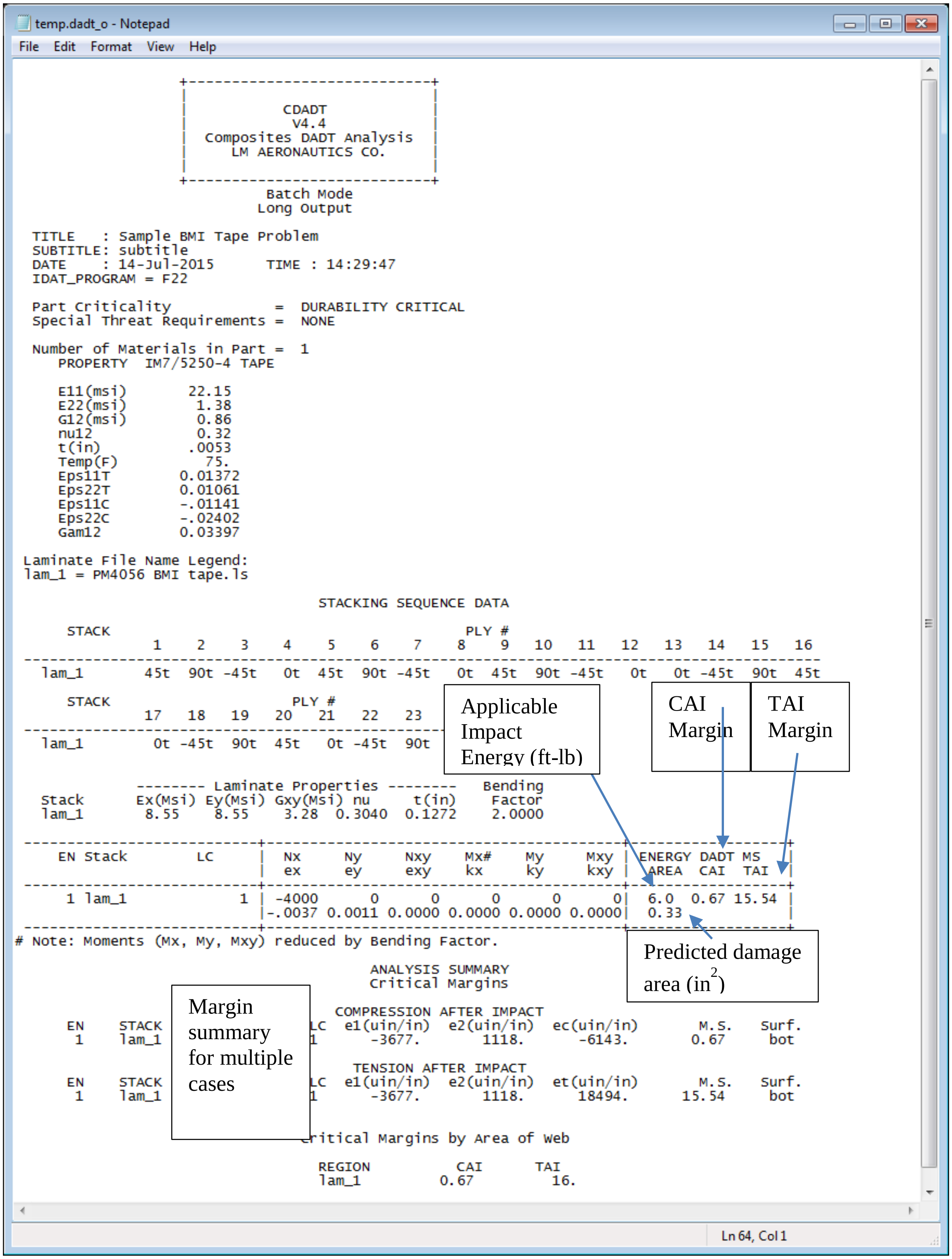


Figure 15.8-1 CDADT Main Dialog for Durability Analysis

Figure 15.8-2 presents the output from the CDADT program for a 24 ply BMI tape quasi-orthotropic laminate. The applicable impact energy (6 ft-lb), Compression after Impact (CAI) and Tension after Impact (TAI) margins are listed. Note that for this problem the allowable durability strain in compression would be -6143 µin/in.



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temp.dadt_o - Notepad
File  Edit  Format  View  Help

                    +-----------------------------+
                    |                             |
                    |            CDADT            |
                    |            V4.4             |
                    |   Composites DADT Analysis  |
                    |     LM AERONAUTICS CO.      |
                    |                             |
                    +-----------------------------+
                             Batch Mode
                            Long Output

  TITLE    : Sample BMI Tape Problem
  SUBTITLE: subtitle
  DATE     : 14-Jul-2015       TIME : 14:29:47
  IDAT_PROGRAM = F22

  Part Criticality             =  DURABILITY CRITICAL
  Special Threat Requirements =  NONE

  Number of Materials in Part =  1
     PROPERTY   IM7/5250-4 TAPE

     E11(msi)        22.15
     E22(msi)         1.38
     G12(msi)         0.86
     nu12             0.32
     t(in)           .0053
     Temp(F)           75.
     Eps11T         0.01372
     Eps22T         0.01061
     Eps11C         -.01141
     Eps22C         -.02402
     Gam12          0.03397

 Laminate File Name Legend:
 lam_1 = PM4056 BMI tape.ls

                                   STACKING SEQUENCE DATA

      STACK                                        PLY #
                1    2    3    4    5    6    7    8    9   10   11   12   13   14   15   16
 -------------------------------------------------------------------------------------------
   lam_1       45t  90t -45t   0t  45t  90t -45t   0t  45t  90t -45t   0t   0t -45t  90t  45t

      STACK                          PLY #
               17   18   19   20   21   22   23
 -----------------------------------------------
   lam_1        0t -45t  90t  45t   0t -45t  90t

              -------- Laminate Properties --------   Bending
   Stack      Ex(Msi) Ey(Msi) Gxy(Msi) nu     t(in)    Factor
   lam_1       8.55    8.55    3.28  0.3040  0.1272    2.0000

 --------------------------+-----------------------------------------------+--------------+
     EN Stack          LC  |  Nx       Ny       Nxy      Mx#      My      Mxy | ENERGY DADT MS |
                           |  ex       ey       exy      kx       ky      kxy |  AREA  CAI  TAI |
 --------------------------+-----------------------------------------------+--------------+
      1 lam_1           1  | -4000      0        0        0        0        0|  6.0  0.67 15.54 |
                           |-.0037 0.0011 0.0000 0.0000 0.0000 0.0000|  0.33             |
 --------------------------+-----------------------------------------------+--------------+
# Note: Moments (Mx, My, Mxy) reduced by Bending Factor.

                                         ANALYSIS SUMMARY
                                         Critical Margins

                                   COMPRESSION AFTER IMPACT
      EN     STACK              LC  e1(uin/in)  e2(uin/in)  ec(uin/in)       M.S.   Surf.
       1     lam_1              1     -3677.       1118.      -6143.         0.67    bot

                                     TENSION AFTER IMPACT
      EN     STACK              LC  e1(uin/in)  e2(uin/in)  et(uin/in)       M.S.   Surf.
       1     lam_1              1     -3677.       1118.      18494.        15.54    bot

                                 Critical Margins by Area of Web

                                   REGION         CAI        TAI
                                   lam_1         0.67        16.

Ln 64, Col 1
```


Figure 15.8-2 CDADT Output for 24 ply BMI Tape Quasi-Orthotropic Laminate for Durability Analysis

15.8.2.2 IDAT/CDADT and Damage Tolerance Allowables Calculation

Figure 15.8-3 presents the IDAT/CDADT main dialog. Under “Criticality” choose the “Fracture Critical Criteria” option to calculate damage tolerance impact allowables.

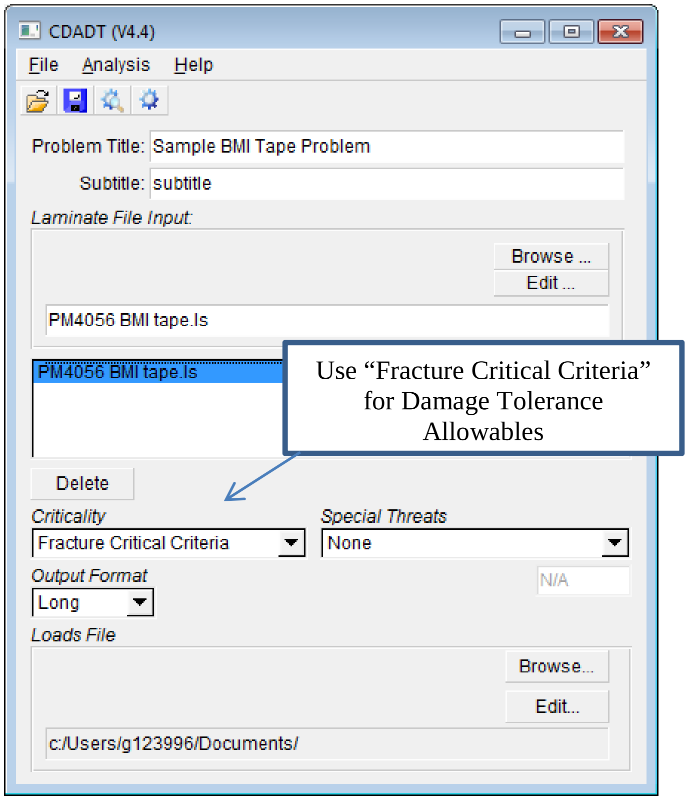
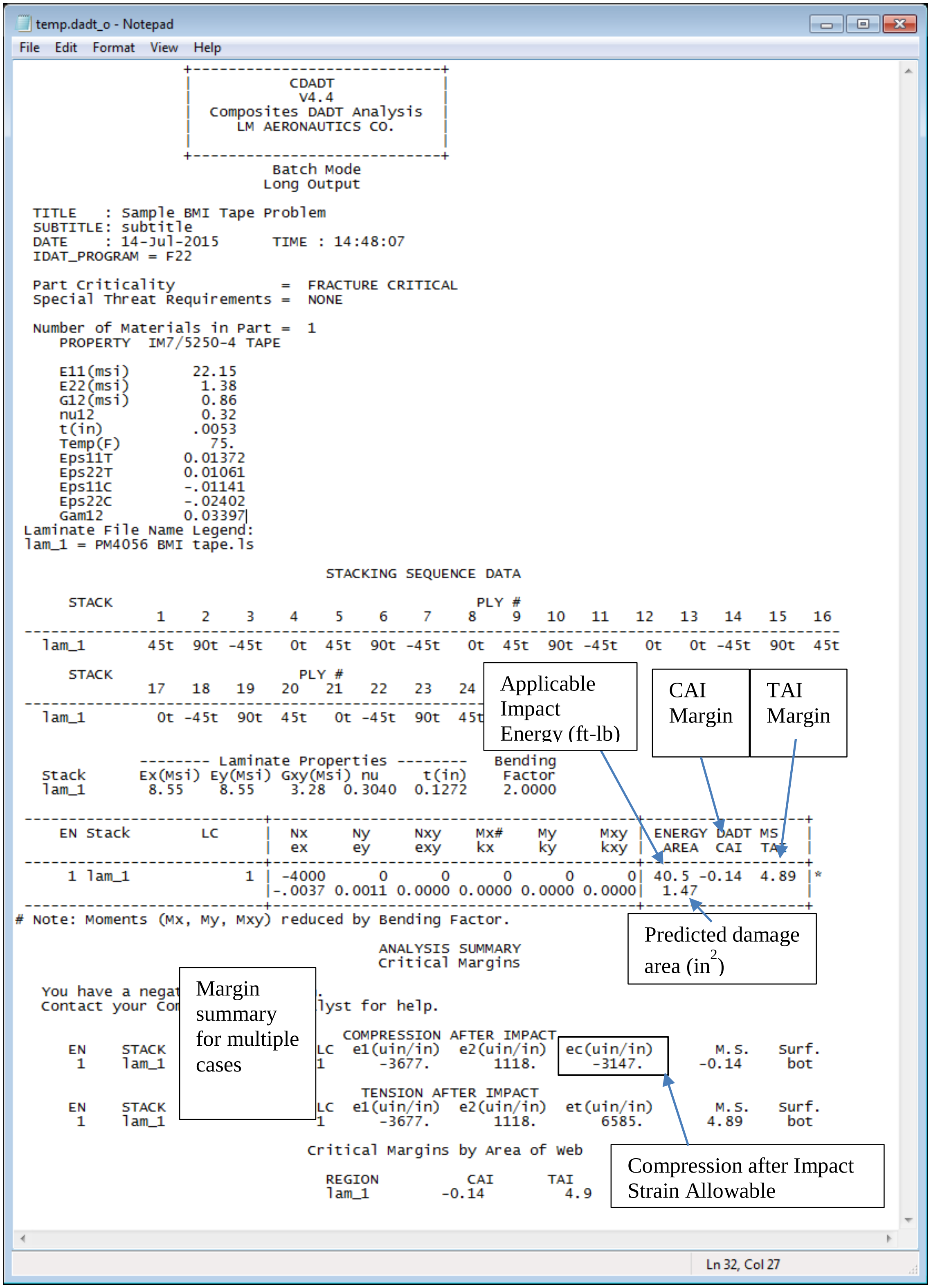


Figure 15.8-3 CDADT Main Dialog for Damage Tolerance Analysis

If fracture critical is selected, the option of using special threats is given. Options include setting a maximum energy level cap, specifying an energy level, or specifying the damage area. Specifying the damage area is especially useful for evaluating strength if the damage area of a part is known. PM4056 Section 19.6.1.1 discusses the use of this option.

Figure 15.8-4 presents the output from the CDADT program for a 24 ply BMI tape quasi-orthotropic laminate. The applicable impact energy (40.5 ft-lb) represents the energy need to develop a barely visible impact and is built into the program. Compression after Impact (CAI) and Tension after Impact (TAI) margins are listed. Note that for this problem the allowable damage tolerance strain in compression would be -3147 µin/in.



```
temp.dadt_o - Notepad
File  Edit  Format  View  Help
                     +-----------------------------+
                     |           CDADT             |
                     |           V4.4              |
                     |   Composites DADT Analysis  |
                     |      LM AERONAUTICS CO.     |
                     |                             |
                     +-----------------------------+
                              Batch Mode
                             Long Output

  TITLE    : Sample BMI Tape Problem
  SUBTITLE: subtitle
  DATE     : 14-Jul-2015        TIME : 14:48:07
  IDAT_PROGRAM = F22

  Part Criticality            =  FRACTURE CRITICAL
  Special Threat Requirements =  NONE

  Number of Materials in Part =  1
     PROPERTY  IM7/5250-4 TAPE

     E11(msi)       22.15
     E22(msi)        1.38
     G12(msi)        0.86
     nu12            0.32
     t(in)          .0053
     Temp(F)          75.
     Eps11T        0.01372
     Eps22T        0.01061
     Eps11C        -.01141
     Eps22C        -.02402
     Gam12         0.03397
 Laminate File Name Legend:
 lam_1 = PM4056 BMI tape.ls

                                 STACKING SEQUENCE DATA

      STACK                                        PLY #
                1    2    3    4    5    6    7    8    9   10   11   12   13   14   15   16
 -------------------------------------------------------------------------------------------
   lam_1       45t  90t -45t   0t  45t  90t -45t   0t  45t  90t -45t   0t   0t -45t  90t  45t

      STACK                        PLY #
               17   18   19   20   21   22   23   24
 ---------------------------------------------------
   lam_1        0t -45t  90t  45t   0t -45t  90t  45t

             -------- Laminate Properties --------   Bending
   Stack     Ex(Msi) Ey(Msi) Gxy(Msi) nu     t(in)    Factor
   lam_1      8.55    8.55    3.28  0.3040  0.1272   2.0000

 -------------------------+---------------------------------------------+---------------+
     EN Stack         LC  |  Nx     Ny     Nxy    Mx#    My     Mxy    | ENERGY DADT MS |
                          |  ex     ey     exy    kx     ky     kxy    |  AREA  CAI  TAI|
 -------------------------+---------------------------------------------+---------------+
      1 lam_1           1 | -4000   0      0      0      0        0    | 40.5 -0.14 4.89|*
                          |-.0037 0.0011 0.0000 0.0000 0.0000 0.0000   |  1.47          |
 -------------------------+---------------------------------------------+---------------+
# Note: Moments (Mx, My, Mxy) reduced by Bending Factor.

                                   ANALYSIS SUMMARY
                                   Critical Margins

   You have a negat                 .
   Contact your Con                 lyst for help.

                              COMPRESSION AFTER IMPACT
     EN     STACK              LC  e1(uin/in)  e2(uin/in)  ec(uin/in)      M.S.    Surf.
      1     lam_1              1     -3677.       1118.      -3147.       -0.14     bot

                                TENSION AFTER IMPACT
     EN     STACK              LC  e1(uin/in)  e2(uin/in)  et(uin/in)      M.S.    Surf.
      1     lam_1              1     -3677.       1118.       6585.        4.89     bot

                           Critical Margins by Area of Web

                             REGION          CAI      TAI
                             lam_1          -0.14     4.9

                                                              Ln 32, Col 27
```


Figure 15.8-4 CDADT Output 24 ply BMI Tape Quasi-orthotropic Laminate for Damage Tolerance Analysis.

The program includes a prediction of damage area. The damage area predictor is useful when performing MRB activities during the disposition of delaminations. For example figure 15.8-5 presents the CDADT main dialog box for the example shown in Figure 15.8-4 but with “Special Threats” set to Damage Area. In this case, suppose a delamination of 1 in^2 was measured and the strength is desired.

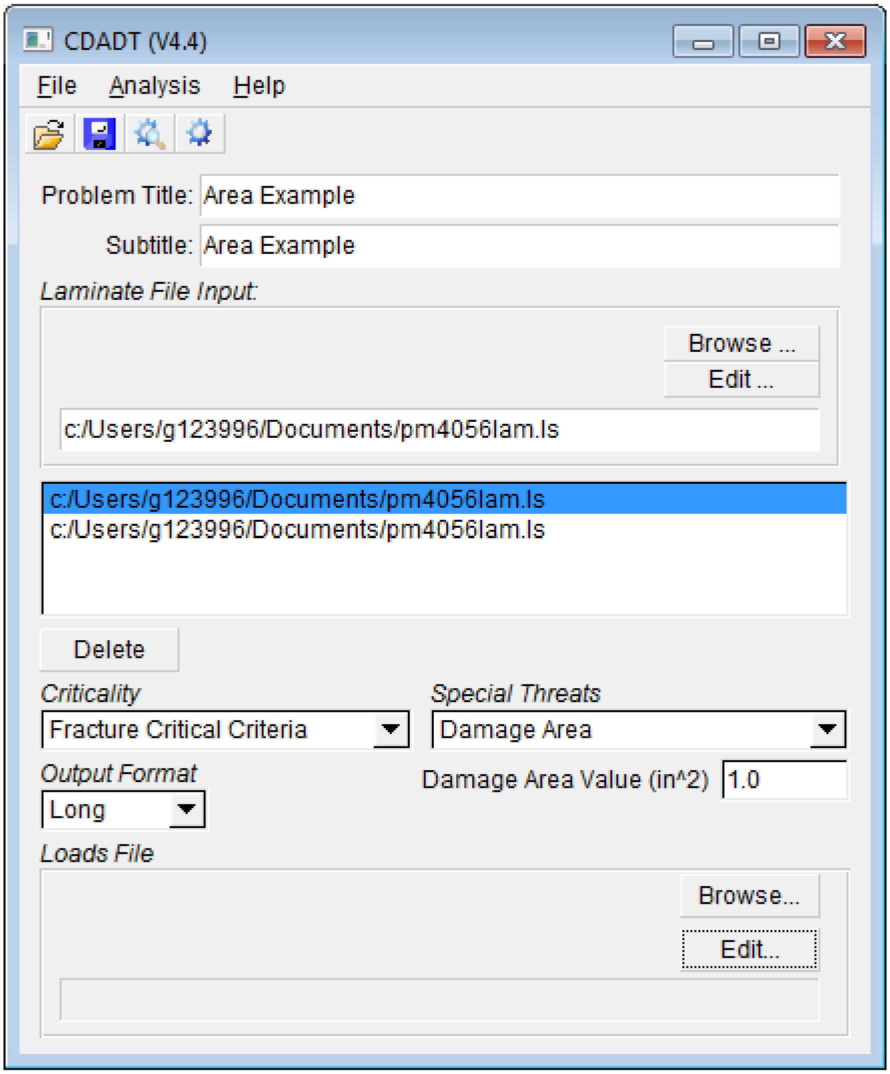


Figure 15.8-5 Main Dialog with Damage Area Special Threat

Figure 15.8-6 presents the results of the analysis with defined damage area. With the new damage area the compression failure strain allowable has improved from -3147 µin/in (Figure 15.8-4) to -3871 µin/in. Remember if MRB activity is being performed the margin needs to be written against ultimate load rather than the residual strength load.

temp.dadt_o - Notepad

File Edit Format View Help

```
                    +-----------------------------+
                    |                             |
                    |           CDADT             |
                    |            V4.4             |
                    |  Composites DADT Analysis   |
                    |     LM AERONAUTICS CO.      |
                    |                             |
                    +-----------------------------+
                             Batch Mode
                            Long Output
  TITLE    : Area Example
  SUBTITLE: Area Example
  DATE     : 23-Sep-2015       TIME : 16:30:58
  IDAT_PROGRAM = F22

  Part Criticality            =  FRACTURE CRITICAL
  Special Threat Requirement  =  SPECIFIED DAMAGE AREA
  Damage Area                 =   1.00      (in^2)

  Number of Materials in Part =  1
     PROPERTY  IM7/5250-4 TAPE

     E11(msi)        22.15
     E22(msi)         1.38
     G12(msi)         0.86
     nu12             0.32
     t(in)           .0053
     Temp(F)           75.
     Eps11T         0.01372
     Eps22T         0.01061
     Eps11C         -.01141
     Eps22C         -.02402
     Gam12          0.03397

 Laminate File Name Legend:
 lam_1 = c:/Users/g123996/Documents/pm4056lam.ls
                                  STACKING SEQUENCE DATA

      STACK                                        PLY #
                1    2    3    4    5    6    7    8    9   10   11   12   13   14   15   16
 --------------------------------------------------------------------------------------------
   lam_1       45t  90t -45t   0t  45t  90t -45t   0t  45t  90t -45t   0t   0t -45t  90t  45t

      STACK                        PLY #
               17   18   19   20   21   22   23   24
 -----------------------------------------------------
   lam_1        0t -45t  90t  45t   0t -45t  90t  45t


              -------- Laminate Properties --------   Bending
   Stack      Ex(Msi) Ey(Msi) Gxy(Msi) nu      t(in)    Factor
   lam_1       8.55    8.55    3.28  0.3040  0.1272    2.0000

 ---------------------------+------------------------------------------+-------------------+
     EN Stack          LC   |   Nx      Ny      Nxy     Mx#     My     Mxy |  ENERGY DADT MS  |
                            |   ex      ey      exy     kx      ky     kxy |   AREA  CAI  TAI |
 ---------------------------+------------------------------------------+-------------------+
      1 lam_1            1  |  -4000     0       0       0       0       0| 24.8  0.05  6.68 |
                            |-.0037 0.0011 0.0000 0.0000 0.0000 0.0000|  1.00             |
 ---------------------------+------------------------------------------+-------------------+
# Note: Moments (Mx, My, Mxy) reduced by Bending Factor.

                                         ANALYSIS SUMMARY
                                         Critical Margins

                                     COMPRESSION AFTER IMPACT
      EN      STACK                LC  e1(uin/in)  e2(uin/in)  ec(uin/in)
       1      lam_1                1     -3677.       1118.      -3871.        0.05      bot

                                       TENSION AFTER IMPACT
      EN      STACK                LC  e1(uin/in)  e2(uin/in)  et(uin/in)      M.S.    Surf.
       1      lam_1                1     -3677.       1118.       8590.        6.68      bot


                                   Critical Margins by Area of Web

                                     REGION          CAI       TAI
                                     lam_1         0.53E-01     6.7
```

Ln 11, Col 40

Defined Damage Area = 1

New compression Allowable

Figure 15.8-6 Results for the Special Thread-Defined Damage Area Example

15.9 Example Calculations

Reserved for future use

15.10 Unix/PC-Based Calculation

As mentioned throughout this section, there are a number of in-house developed, computer based programs for use in durability and damage tolerance analysis of composites. Table 15.10-1 summarizes the programs available in the IDAT tool suite.

Table 15.10-1 Summary of Computer Tools for Use in Composite Durability and Damage Tolerance Analysis

Tool	Applicability
IDAT/IBOLT	¼ in Open and Filled Hole, Tension and Compression
IDAT/CDADT	Durability Impact (4 and 6 ft-lbs); Damage Tolerance Impact (up to 100 ft-lbs)
IDAT/SFBOLT	Fatigue Life of Bolts