

APPLICATIONS OF A DAMAGE TOLERANCE ANALYSIS METHODOLOGY IN AIRCRAFT DESIGN AND PRODUCTION

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ABSTRACT

Objectives of customer mandated aircraft structural integrity initiatives in design are to guide material selection, to incorporate fracture resistant concepts in the design, to utilize damage tolerance based allowables and planned inspection procedures necessary to enhance the safety and reliability of manned flight vehicles. However, validated fracture analysis tools for composite structure are needed to accomplish these objectives in a timely and economical manner. This paper briefly describes the development, validation, and application of a damage tolerance methodology for composite airframe structures.

A closed-form analysis code, entitled SUBLAM was developed to predict the critical biaxial strain state necessary to cause sublaminar buckling-induced delamination extension in an impact damaged composite laminate. An embedded elliptical delamination separating a thin sublaminar from a thick parent laminate is modelled. Predicted failure strains were correlated against a variety of experimental data that included results from compression after impact coupon and element tests. An integrated analysis package has been developed to predict damage tolerance based margins-of-safety (MS) using NASTRAN generated loads and element information. Damage tolerance aspects of new concepts are quickly and cost-effectively determined without the need for excessive testing.

INTRODUCTION

Emerging military aircraft contain higher percentages of advanced composite materials in flight critical primary structure applications to meet demanding weight and performance goals. Examples include the B-2 stealth bomber, the F-22 advanced tactical fighter, the F-117A stealth fighter, and the V-22 tilt-rotor. These new aircraft must meet demanding requirements for durability and damage tolerance. There is a need for validated analysis tools that can be used in a production environment to assess the effects of delamination damage on the residual strength of composite structure in a timely and efficient manner. In addition, analysis tools are also needed to aid in the disposition of discrepant parts during manufacture, and to develop accept/reject criteria necessary for quality control. In this paper, applications of

a damage tolerance methodology in the design and analysis of composite airframe structure will be emphasized. The development of the fracture analysis model will be briefly discussed.

The certification of airframe structures for durability and damage tolerance has been the impetus for the development and application of fracture analysis methods at General Dynamics to improve product reliability, safety, and supportability. The goal has been to develop and validate efficient, closed-form analysis tools that are readily available to the designer and the analyst. A damage tolerance analysis (DTA) methodology for composite structures was developed and validated under a General Dynamics IR&D task. The methodology is embodied in a computer program entitled SUBLAM. SUBLAM calculates the critical biaxial strains required to cause sublaminar buckling-induced delamination extension in a composite plate.

Applications of this damage tolerance methodology in the design and manufacture of military aircraft are described. The new military damage tolerance requirements and their relationship with the analysis methodology are discussed. The use of SUBLAM to establish damage tolerance based compression strain allowables, optimize laminate stacking sequences, size structure, and develop quality control criteria for composite structure is addressed.

METHODOLOGY DEVELOPMENT

Background

Sublaminar buckling induced delamination growth produced by compressive loading has been recognized as a worst case failure mode in composite structures (ref. 1). In this failure mode, delamination damage causes the sublaminates to buckle under compressive loading. As the compressive loading increases, strain energy stored in the buckled sublaminar also increases until reaching a critical level (denoted as G_c , the critical strain-energy release rate). At this critical strain level, the delamination is assumed to propagate when $G \geq G_c$. Since failure occurs at the initiation of delamination growth, this failure mode is treated as a static problem. This behavior has been experimentally observed in a large number of impact damaged component fatigue tests at General Dynamics and elsewhere (ref. 1).

The finite element modeling (FEM) technique has been extensively utilized to address local instability-induced fracture behavior in composite materials. However, the application of complex two- and three-dimensional FEM's to solve fracture problems is not ideally suited for the aircraft industry due to time and cost constraints. In addition, the number of parts that must be efficiently analyzed during design is staggering. Thus, a fast, closed-form delamination analysis tool was developed to aid the analyst in assessing the damage tolerance of composite structure.

The SUBLAM computer program was formulated in 1988 and 1989 using the model developed by Chai and Babcock (ref. 2) as the baseline. The model treats the problem illustrated in Figure 1 in which a thin sublaminde disbonds from a thick parent laminate. The buckled sublaminde is assumed to be elliptically shaped with clamped boundary conditions. The elliptical shape can be varied by changing the aspect ratio, defined as (a/b).

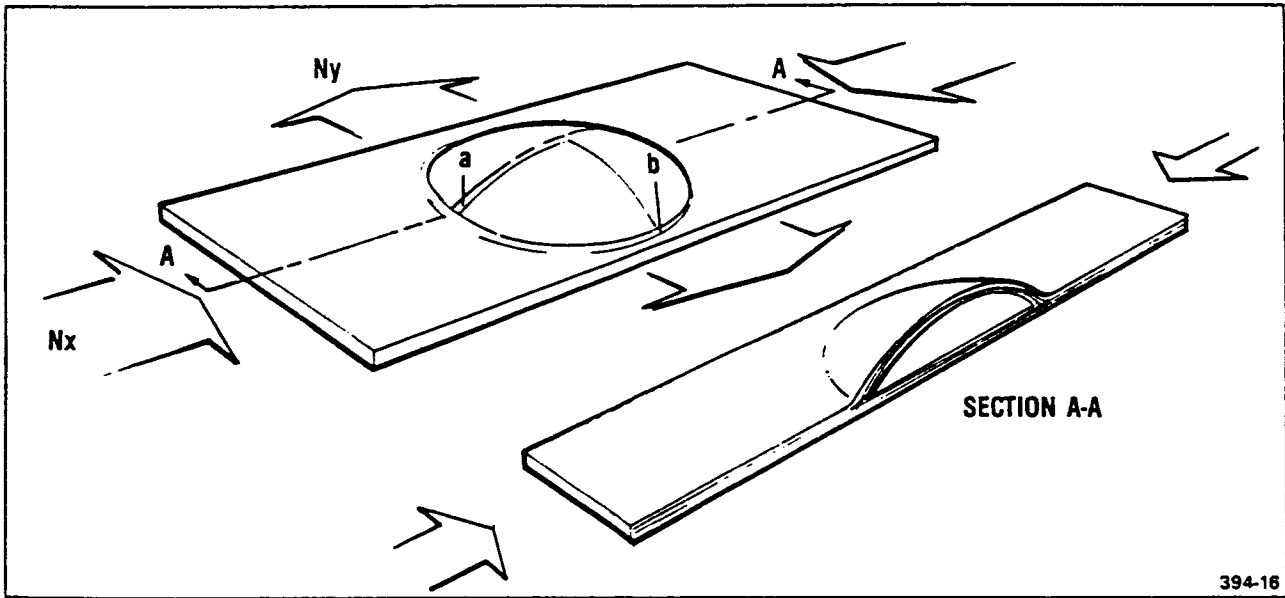


Figure 1 SUBLAM Models A Delaminated Region In A Biaxially Loaded Composite Plate

A single-term Rayleigh-Ritz solution technique is used to describe the buckled shape of the sublaminde. Through differentiation of the total potential energy with respect to the delamination ellipse major and minor dimensions, "a" and "b" respectively, the total strain-energy release rate (G) is calculated as an average around the perimeter of the delamination. When G_a or G_b is $\geq G_c$ (the critical strain-energy release rate), delamination growth is assumed to occur in a self-similar fashion. Thus, the biaxial strains required to cause buckling induced delamination growth in a composite plate are calculated. Predictions are made at every ply interface and can be performed for different in-plane loadings (N_x , N_y , and N_{xy}), flaw aspect ratios, and sizes. Program inputs include laminate stacking sequence, critical G , flaw sizes and aspect ratios, and lamina mechanical properties. Additions to Chai and Babcock's model were incorporated in SUBLAM. Both the sublaminde and the parent laminate are represented as homogeneous anisotropic layers. To predict failure strains produced by low velocity impact damage (LVID), the extensional and bending stiffness terms (A_{ij} and D_{ij}) were assumed to be degraded using a rule-of-mixtures approach.

SUBLAM was integrated into an overall damage tolerance analysis (DTA) methodology that utilizes information from the NASTRAN finite element model of the aircraft structure. Several codes used to extract load and stacking sequence data, as well as format results for post-processing were packaged into a large command file program entitled "DTC". The application of these codes in structural design and analysis are described in the following section.

Validation

General Dynamics has conducted numerous experiments to correlate and validate fracture analysis models. Durability and damage tolerance (DADT) testing has been conducted using coupon, element, and component size articles, as well as full-scale flight hardware. This effort has created a sizeable amount of data on the impact damage problem and its effects on the residual compressive strength of composite structures.

SUBLAM predicted failure strains were correlated against a variety of experimental data. Predictions were made for coupons, single-bay box-beam components, and stiffened panels subjected to compression after impact (CAI) testing. Additional stiffened panel tests data were obtained from contractor reports (ref. 1). Different laminate ply-percentages, layups, thicknesses, panel geometries, and material systems were used in the correlation. SUBLAM generally predicted conservative values, as shown in Figure 2. In some cases, the G_c was unknown, and therefore was estimated.

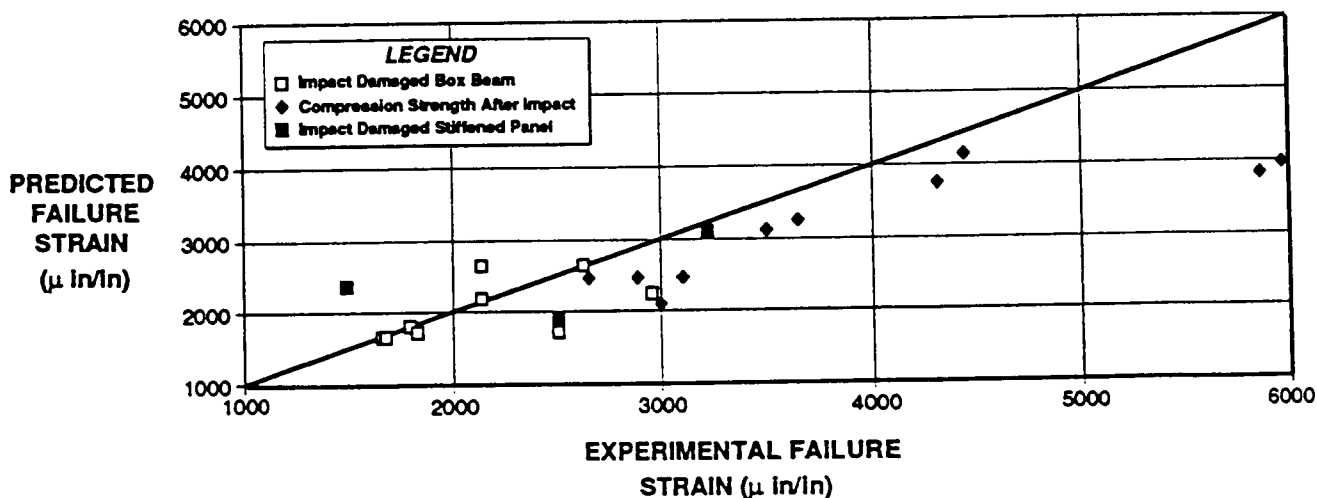


Figure 2 SUBLAM Predictions Correlate Well With Experimental Results

DAMAGE TOLERANCE REQUIREMENTS

Air Force and Navy damage tolerance specifications for composites dictated the approach taken during development, validation, and implementation of General Dynamics' damage tolerance analysis methodology. General requirements and the implications of these specifications on the methodology are described below.

Specific damage tolerance requirements and design criteria for organic matrix composite materials were not specified during design and development of various military aircraft introduced in the 1970's and early 80's, such as the F-16, F-15, and F/A-18. Instead, composite structures were certified on an ad hoc basis, usually via full-scale component testing. Typical manufacturing and in-service threats were identified, but a consensus for design purposes was not established. These early composite airframe structures are primarily damage tolerant due to the use of low design strain allowables. Strain allowables for carbon/epoxy composites are usually established by low bolted joint strength, global buckling, flutter, stiffness, and strength considerations. Fatigue has not been an issue since the operational strain levels are safely below the fatigue threshold for graphite/epoxy composite materials.

New and emerging aircraft fabricated using advanced composite materials in fracture critical airframe applications are designed to meet new damage tolerance requirements for organic matrix composites such as those contained in Air Force Guide Specification 87221A (ref. 3). Damage tolerance requirements can vary considerably depending on the customer, i.e., Air Force, Navy, Army, and FAA. Although the philosophy of designing fracture critical structure to safely tolerate the presence of damage produced either during manufacture or from an in-service event for a period of time while maintaining a specified residual strength is uniformly embraced by both government and industry.

Delamination produced by low-velocity impact damage (LVID) has been demonstrated to represent a worst case threat in terms of reductions in static and fatigue residual strength (ref. 1). Single and multiple delaminations are not as serious as impact damage. Thus, certification specifications address delamination damage produced by LVID as a worst case threat for design purposes. For instance, (dependent on the customer) fracture critical, primary load path structures are designed to sustain design ultimate load in the presence of visible impact damage with no progressive flaw growth for two lifetimes (Navy), or to sustain the once-per 20 lifetimes maximum spectrum load in the presence of damage produced by the energy necessary to create a 0.1 inch deep dent to a maximum of 100 ft-lb of energy after two design lifetimes (Air Force). The impact damage is assumed to exist in the worst possible location and orientation. Impact damage is typically produced by either a 0.5 or a 1.0 inch diameter hemispherical steel impactor.

For the composite material systems used date, the above criteria has been satisfied using a static analysis. Damage tolerance testing has verified that this approach for the certification of composite airframe

structure is sound. Composite structure designed to sustain maximum design flight loads in conjunction with end-of-life environmental properties will tolerate the presence of low-velocity impact induced delamination damage without threat of subcritical growth. This approach to the design and certification of composite airframe structures for damage tolerance is considered to be conservative.

APPLICATIONS IN DESIGN AND ANALYSIS

Design Information

Our overall objective during design is to guide material selection, to incorporate fracture resistant concepts in the design, and to establish damage tolerance based strain allowables and planned inspection procedures necessary to enhance the safety, reliability, and supportability of manned flight vehicles. The use of a damage tolerance analysis methodology to accomplish these goals is described. The importance of addressing damage tolerance early in the design process is emphasized. Otherwise, cost and schedule can be adversely impacted due to drawing and/or tooling changes necessary to meet customer damage tolerance requirements.

The demand for tougher, more damage tolerant composite materials has led suppliers in recent years to introduce a variety of new material systems, including thermoplastic, toughened epoxy, and toughened bismaleimide resins. The toughness issue is still debated throughout the aerospace industry. As in metallic materials, the price paid for increased toughness has been at the expense of strength. Specifically, for composites the hot/wet compression strength is decreased. The material selection process for new flight vehicles has become one of the most challenging, divisive, important, and sometimes political decisions made during the development stage. There are sometimes misconceptions concerning the importance of toughness in aircraft design. Historically speaking, damage tolerance strain allowables have been a factor in the sizing of less than 10 percent of the total number of composite parts on some recent aircraft programs. In other words, damage tolerant materials are desirable, but not at the expense of weight and cost. SUBLAM can be used in trade studies to predict failure strains for different material systems. Thus, fracture toughness requirements can be methodically assessed, and help in making good material selections.

During preliminary design, SUBLAM can be used to optimize stacking sequences for enhanced damage tolerance and to establish compression strain allowables. For buckling critical structure, the optimum stacking sequence can conflict with the best layup for enhanced damage tolerance. Laminates with grouped, major load bearing plies placed near the surface are prone to delaminate due to large stiffness changes through the thickness (creating high energy interfaces). As shown in Figure 3, the damage tolerance based compression strain allowable is increased by "softening" (incorporating angle-ply material) the outside laminae in the direction of the primary compression load. In addition, the benefits of subsymmetry on laminate design in reducing transverse shear are also predicted to increase the damage tolerance

strain allowable. If strength deficiencies are predicted, solutions may include rearranging the existing stacking sequence (increasing the strain allowable), or by the addition of plies (reducing the applied strain). It has been shown analytically and verified through test that the compression after impact strain allowable (for a laminate with fixed ply percentages) is highly dependent on the stacking sequence.

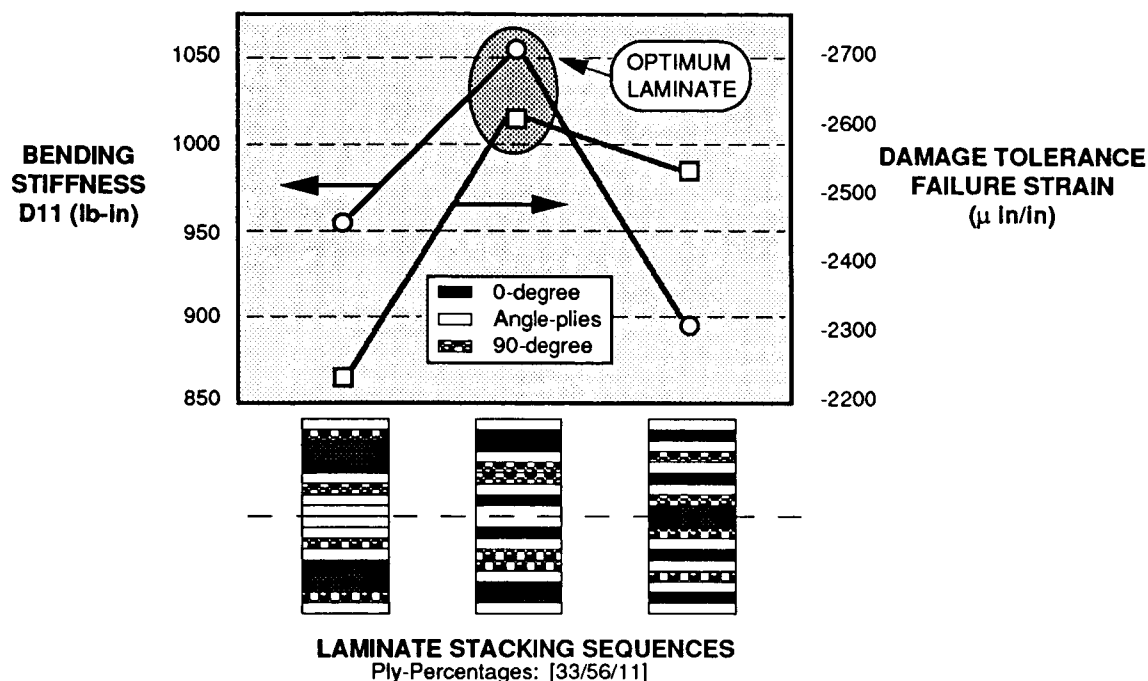


Figure 3 SUBLAM Is Used In Optimization Trade Studies

Damage Tolerance Analysis

Validated analysis tools used for the design and analysis of damage tolerant composite structure have been successfully implemented at General Dynamics. The application of these tools to certify primary composite airframe structure for damage tolerance by analysis is described. The analysis procedure is highly automated. A flowchart of the basic processing steps is shown in Figure 4. The first task is to obtain mid-plane loads, element identification, and material properties from the finite element model. Data is typically extracted via the NASTRAN portable file used for post-processing. Models of the entire aircraft or a component thereof can be selected. Next, the element and load information is matched to a stacking sequence from the engineering drawing. If the correct laminate stiffness data is included in the finite element model, the laminate stiffness matrices can be obtained directly. A conservative assumed flaw size is typically selected based on delamination area associated with visible impact damage. The optimum value for G_c is determined via correlations with experimental data. In

the absence of an empirically correlated G_c value, the mode I fracture toughness, measured using the double cantilever beam test is utilized. Not surprisingly, experience has shown that use of the mode I G_c in SUBLAM typically results in unconservative predictions. Damage tolerance based margins-of-safety are calculated for each element by comparing the SUBLAM predicted critical principal strain to that from the FEM at 120% of design limit load (or 125% of the maximum spectrum stress, whichever is greater). Both tabular listings and computer files for post-processing are created to expedite the data review process. As an example, a contour plot of damage tolerance based margins-of-safety for a horizontal empennage skin at static design ultimate load is shown in Figure 5. Hot spots are easily identified for more detailed analysis if required. In some cases when a negative MS is predicted, the assumed flaw size can be larger than representative elements. Thus loads are averaged in a detailed analysis to more adequately represent the actual situation.

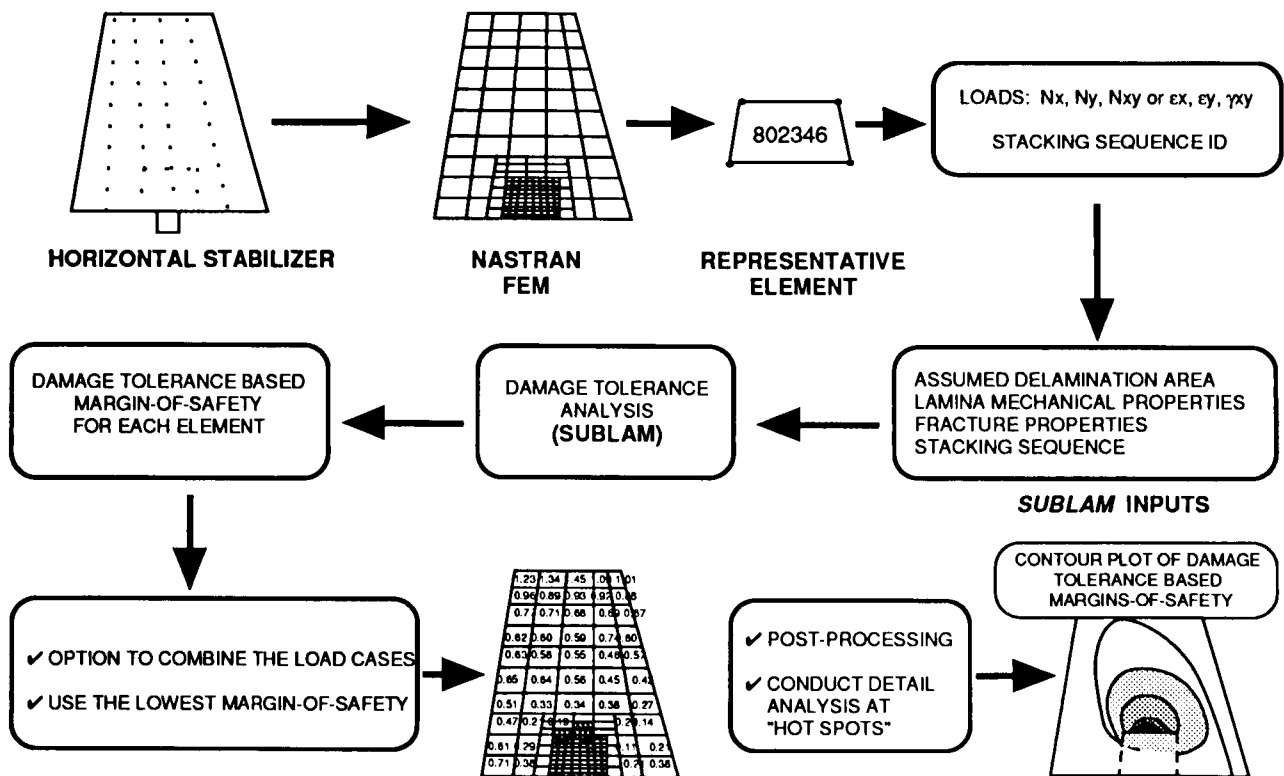


Figure 4 Damage Tolerance Methodology Flowchart

Once problem areas are identified, and the loads verified, several courses of action can be taken to increase the MS. The first option is to rearrange the laminate stacking sequence. Advantages are that no weight penalty is incurred and that no tooling changes are required. Another choice is to simply add plies in the direction of the primary compressive load. Unfortunately, the analyst becomes the bad guy by causing weight increases! Other potential problems are increases in load attracted to the area caused by increased stiffness.

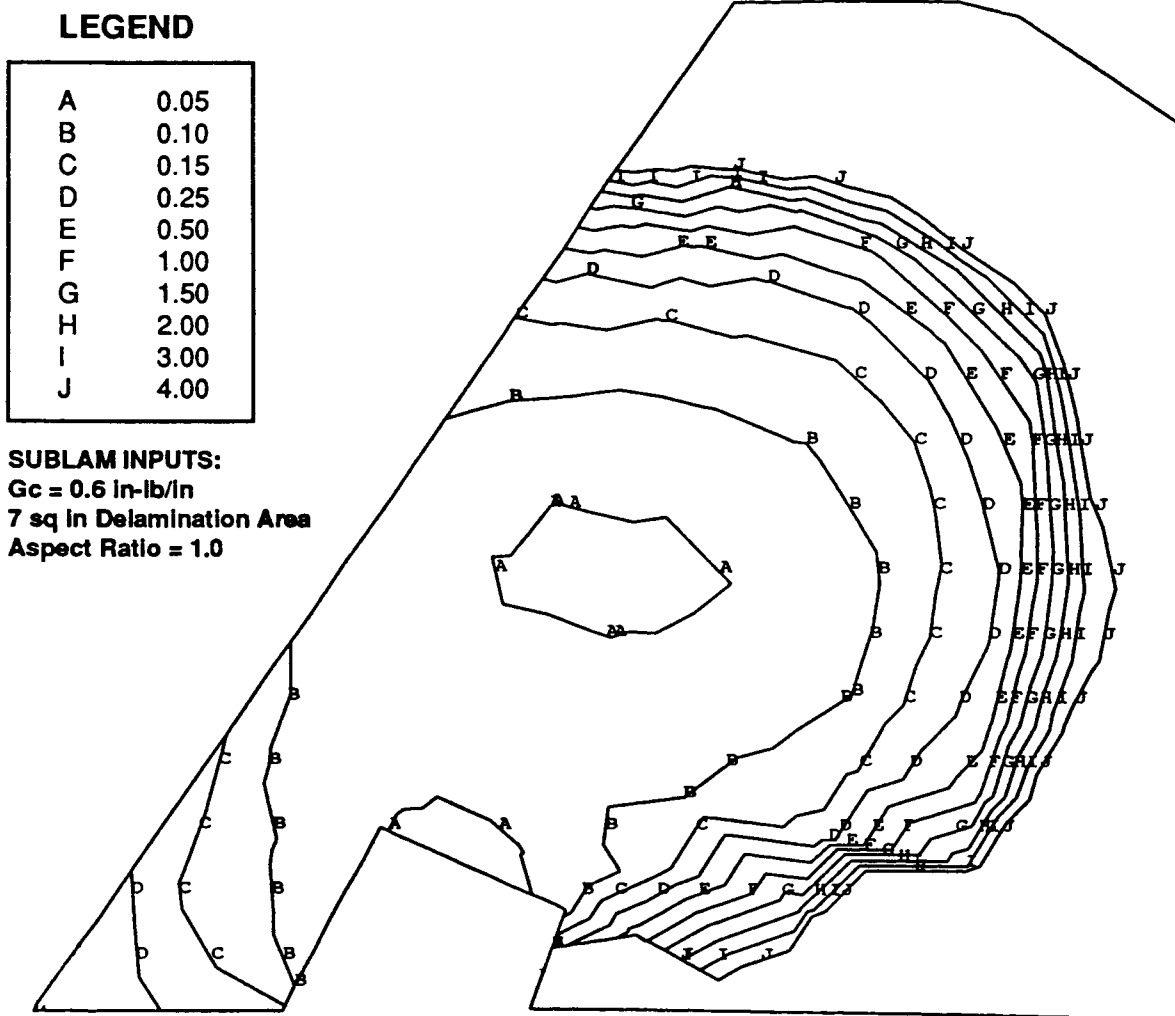


Figure 5 Contour Plot of Damage Tolerance Based Margins-of-Safety For A Composite Horizontal Empennage Skin

The delamination analysis methodology can also be used to perform updated damage tolerance analyses of composite structures. Loads and part geometries can change to meet updated mission profiles, design changes, or usage variations. The simplicity of the model allows an analyst to perform an expedient damage tolerance analysis. The model can be used to extend beyond the verification database when new materials and loads are introduced. In addition, the methodology can be utilized to reduce time involved in conducting analyses in support of engineering changes to production drawings. Design changes are costly in terms of labor hours and potential schedule impacts. Revised parts are reanalyzed to meet certification specification requirements. Again, application of efficient fracture analysis tools reduces time and labor costs.

Applications in the Quality Assurance Process

Development of Accept/Reject Criteria

In the design and development stage of an aircraft program, SUBLAM can be effectively utilized to establish quality control accept/reject criteria for manufactured parts. On a recent aircraft program, all of the composite parts were "zoned" for maximum allowable delamination sizes. Five zones were established based on predicted maximum allowable delamination sizes. Allowable delamination sizes were calculated using ultimate loads. Flaw depth was not specified in the criteria, only flaw area for the sake of conservatism, inspectability, and brevity. It was assumed that the flaw was located at the critical interface. Zoning of composite parts effectively reduces the time, cost, and complexity associated with nondestructive inspection.

Disposition of Discrepant Manufactured Parts

During the manufacture and assembly of composite airframe structures, delaminations can be induced by improper machining techniques, forcing together improperly shimmed parts, and foreign object damage (FOD). It is imperative to determine the significance of delaminations that are induced during the manufacturing process in a timely manner. SUBLAM has been effectively utilized in the quality control process at General Dynamics to aid engineering in the disposition of discrepant parts.

Typical procedures include a thorough nondestructive inspection to assess the depth(s) and area(s) of each delamination. The local stacking sequence, lamina mechanical properties, critical G, strain state at 120% of design limit load, and the flaw dimensions are input to SUBLAM. High aspect ratio delamination shapes are conservatively represented using a circular shape. A margin-of-safety is calculated at the specific ply interface of concern. A disposition is made based not only on results from the delamination analysis, but on the flaw location, repair options, dollar value of the part or assembly, and part classification (fracture critical parts require special attention). The part is either scrapped, returned to print via a remove and replace operation, or repaired.

Disposition of Service-Induced Damage

Similarly, the severity of service-induced delamination damage must be quickly and effectively assessed by field and depot level operations personnel. A number of disposition options are normally considered depending on part function, location, and classification. With proper guidance, fracture analysis tools similar to SUBLAM could be confidently used by field-level personnel to determine whether to operate as is, ferry flight the aircraft to a repair depot, remove the part from service, or specify standard repairs.

SUMMARY

An efficient damage tolerance analysis methodology has been developed that is useful during the design, development, and production stages of an aircraft program to establish damage tolerance based strain allowables, conduct damage tolerance analyses, aid in the disposition of discrepant parts both in manufacturing and in service environments, and establish acceptance and rejection criteria for composite structure. Incorporating damage tolerance in the design process will enhance product reliability, supportability, and maintainability without significant cost or weight implications.

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