Damage Tolerance and Logistic Transport Design

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Logistic transport aircraft are subjected to many possible sources of damage, including corrosion, fatigue, overloads, contact with foreign objects, and battle damage. Efficient and safe operation in spite of such damage is enhanced by incorporating damage tolerance as a design feature. The damage tolerance concept provides structural redundancies such that a reasonable amount of damage can be sustained without endangering safe operation. Reliability, maintainability, and operational safety of the aircraft can be improved through damage tolerant design with little or no weight penalty. Orderly and effective maintenance schedules, and extension of service life potential far beyond the original design service goal, are additional benefits. Ingenuity and laboratory tests are the keys to cost (and weight) effective damage tolerant design. It is felt that damage tolerance should be required in all designs of manned aircraft because of the benefits and operational safety that it provides.

Nomenclature

a = stringer spacing, in.
A_e = effective cross-section area of reinforcement accounting for eccentricity of load path, in.^{2}
b = frame spacing, in.
F_g = gross area stress, ksi
F_{tu} = ultimate tensile strength of the material, ksi
n = number of identical elements
P_{cr} = pressure differential at which rapid crack propagation takes place, lb/in.^{2}
$P(t)$ = ratio of noncracked population to initial population (cumulative probability of cracking), where time t is measured in units of median time to crack
R = radius of the pressure cylinder, in.
σ = standard deviation
t = material thickness, in.
W_e = effective width of intact material at the tip of the crack, in.
X_0 = initial crack length, in.

The Case for Damage Tolerance Design

DAMAGE tolerance relates to the capability of an aircraft structure to sustain a limited amount of damage without endangering safe operation. To do this, the structure must retain the capability of supporting a reasonable percentage of design load after being damaged. Complete failure of a structural member is permissible within the scope of this definition, provided alternate structures or systems exist that allow continued safe operation of the aircraft.

Utilization rates of modern military logistic transports approach that achieved in commercial airline operations. This situation, and the need for inherently high reliability in military airlifters to enable them to operate effectively in adverse environments, led Lockheed-Goergia to incorporate damage tolerance concepts into structural designs.

Using Lockheed aircraft as examples, the need for damage tolerance is clearly seen. The average daily utilization rate of the C-141 StarLifter exceeded 8 hr throughout the month of February 1968. One C-141, based at Dover Air Force Base, Delaware, achieved an average daily utilization rate of $21\frac{1}{4}$ hr in March 1968. Such high utilization rates, coupled with the extremely severe environments in military operations, can accelerate damage due to corrosion, fatigue, and overloads.

With more than ten years of operational experience, the Lockheed C-130 Hercules has proven to be an excellent testbed for damage tolerance design. Although losses of C-130 aircraft have occurred from various causes, none have been attributed to fatigue, corrosion, or related effects. Some examples of damage sustained by C-130's and other Lockheed-Georgia aircraft will serve to illustrate the value of damage tolerance design:

1) An improperly locked door, suddenly opening in flight at 20,000-ft altitude, caused explosive decompression and extensive damage to the fuselage forebody. The C-130, however, recovered and landed safely.

2) Corrosion damage to a C-130 pressure shell and structural bulkhead, caused by a leaking latrine, is shown in Fig. 1. In another case, corrosion caused by water entrapment between a titanium heat shield and the wing plank structure attacked a lower wing plank producing large holes in several integral panels.

3) Repeated ground and pressure loads initiated the dangerous crack shown in Fig. 2. Laboratory analysis of the C-130 structure indicated that sufficient strength remained to provide an adequate margin of operational safety.

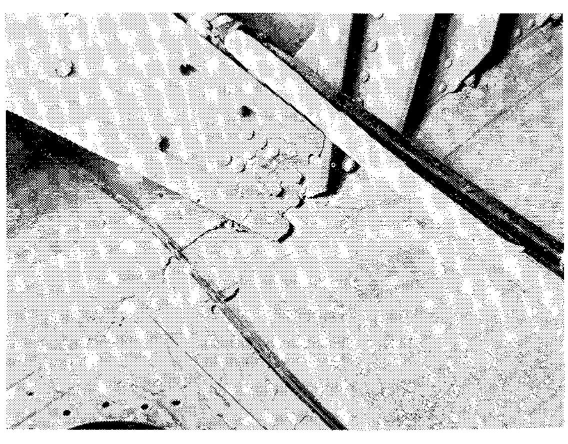

Fig. 1 C-130 corrosion damage.

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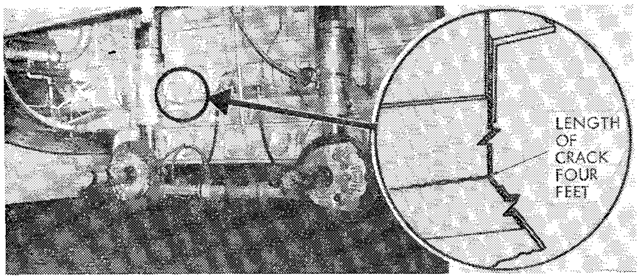


Fig. 2 C-130 chine crack damage.

4) An unexpected dive maneuver in a C-140 JetStar resulted in the damage to the horizontal tail structure pictured in Fig. 3. Here again, the aircraft recovered and landed safely.

5) Examples of C-130 battle damage contained by damage tolerance design are numerous. In one case, damage occurred to an integrally machined wing plank that broke several risers. No progression of the fracture was evident.

6) Damage sustained by pressure skins, frames, and fixed and movable control surfaces is illustrated in Fig. 4. As in the previous case, no progression of the damage is evident. Both of the C-130's survived these incidents. Other examples of nonfatigue related damage are pictured in Figs. 5–8. In all cases, damage was sustained without catastrophic failure of the aircraft structure.

7) Figure 5 simply shows that a C-130 is somewhat more damage tolerant than a goose.

8) In Fig. 6, the consequence of striking a horse that wandered onto the runway during landing of a C-141 is pictured. In this case, the damage propagated to some degree during the flight to a repair base.

9) Loss of a propeller in flight caused the damage to a C-130 pictured in Fig. 7.

10) The C-140 damage indicated in Fig. 8 was the result of an in-flight collision.

Although all the circumstances leading to these effects could hardly be accounted for, anticipation of possible in-service damage prompted damage tolerance design in C-130, C-140, and C-141 structures. The same degree of damage tolerance is provided in the giant C-5 structure.

In addition to the more or less spectacular causes of damage previously illustrated, laboratory fatigue tests and inspections verified that fatigue-oriented damage could be anticipated much sooner than normal operational records would indicate when aircraft were subjected to extremely high utilization and severe operating environments. Figures 9 and 10 picture fatigue cracks measuring 20 and 8 in. in length, respectively. These cracks were induced by laboratory fatigue tests on wing structures. Similar cracks have developed on in-service C-130 aircraft as a consequence of hard use. The laboratory tests, simulating the severe operating environment, led to inspection schedules to detect these cracks as early as possible. Damage tolerance design greatly lessened the possibility of such cracks causing a serious accident or requiring excessive down time for emergency repair. Cracks such as these were contained sufficiently by damage tolerance design to permit safe operation until the using command was able to ferry the aircraft to a repair base.

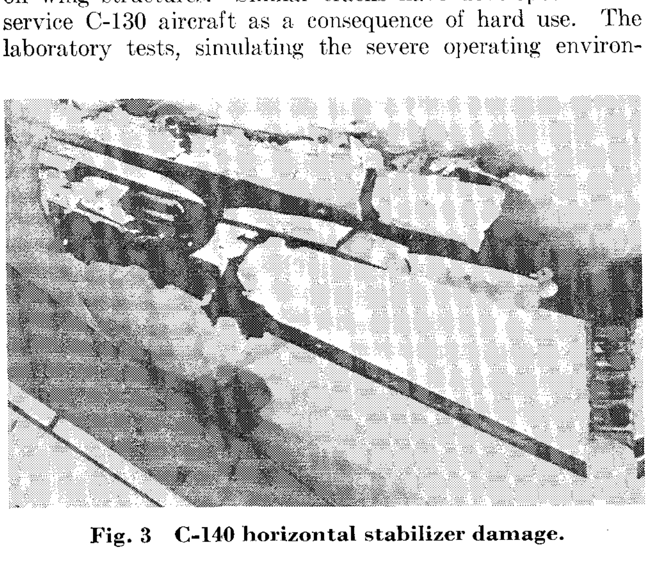


Fig. 3 C-140 horizontal stabilizer damage.

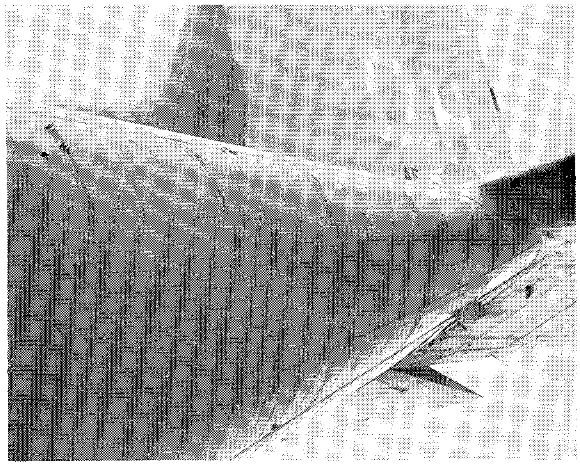


Fig. 4 C-130 battle damage.

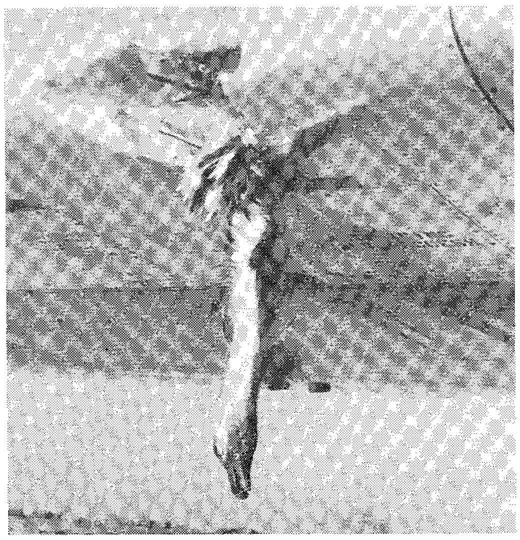


Fig. 5 C-130 wing damage.

Under conditions of severe usage, damage can develop and propagate with unexpected rapidity. A damage tolerant structure, however, greatly increases the chances of damage being detected before it reaches critical proportions affecting safe operation.

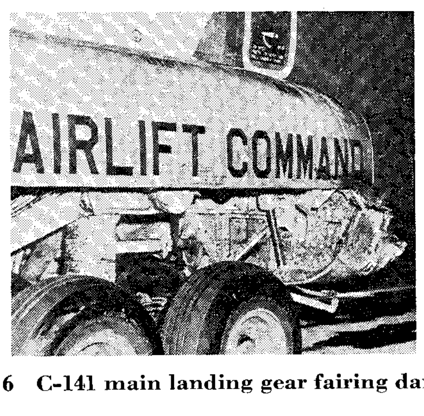


Fig. 6 C-141 main landing gear fairing damage.

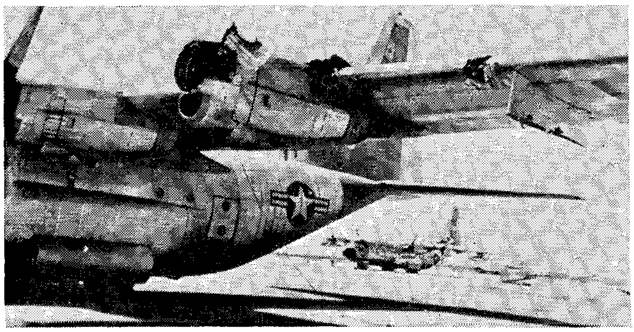


Fig. 7 Result of propeller loss on C-130.

To some extent, damage tolerance design is forgiving of human error. Sometimes, in fabrication, or during overhaul or repair operations, a part has been omitted or incorrectly installed. Subsequent acceptance tests involving pressurization or loading could have resulted in a serious accident had not the structure been designed to be damage tolerant. As embarrassing as these incidents are, they serve to substantiate the value of damage tolerance design.

Damage Tolerance Levels in Design

It is difficult, and in some instances impossssible, to ascertain all possible types and sources of in-service damage to aircraft. Consequently, it is often just as perplexing to specify how damage tolerant a structure should be. Damage tolerance levels must always conform to existing and accepted design standards, such as "Fail Safe" requirements, and even improve upon them.

Accordingly, at Lockheed-Georgia, damage tolerance minimum requirements meet or exceed the U. S. Federal Aviation Regulation, Pt. 25, relating to airworthiness standards for transport category airplanes.

At present, all C-130, C-140, and C-141 designs have been certificated as meeting these civil requirements in addition to military design requirements spelled out in various specifications. A design objective for the C-5 is to have it certificated also. In general, all of these requirements have been met with positive safety margins. This means that, in many instances, the structures involved have a greater damage tolerance than called for in the regulations. An important fallout of this philosophy is that in providing for damage tolerance capabilities meeting required load levels, even more protection is afforded in containing damage at lower load levels.

Table 1 indicates how damage tolerance criteria are applied to Lockheed-Georgia designs. This table delineates the

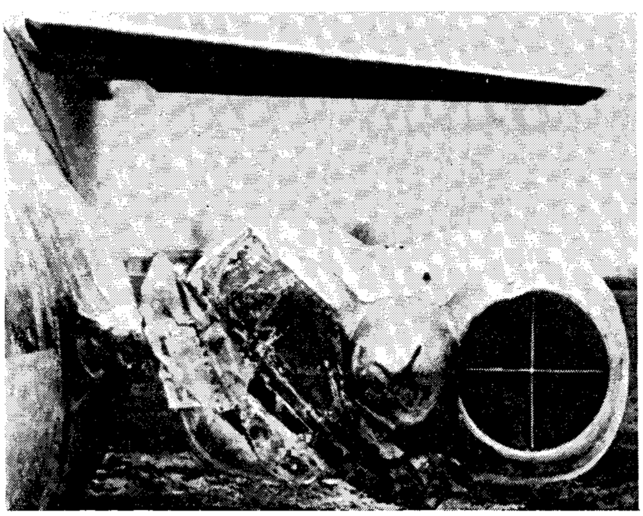


Fig. 8 Result of in-flight collision involving C-140.

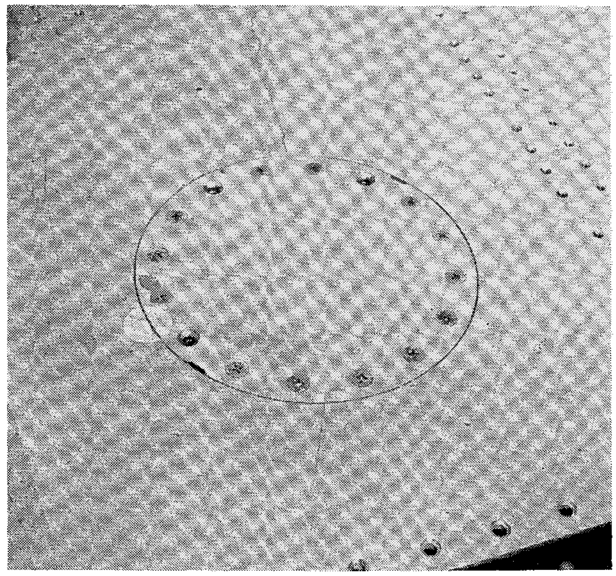


Fig. 9 C-141 fatigue crack.

extent of structural damage that can be tolerated in particular structures. Here, a type of damage is assumed and its extent is related to various aircraft. Then an ultimate load capability for the structure is established to serve as damage tolerance design criteria.

By placing emphasis on the type of damage that can be sustained, rather than on a speculative cause of damage, a more positive and rational approach is realized in evolving damage tolerance criteria. Data may also be related to damage or failures occurring in actual operation to validate assumed design damage.

Damage Tolerance Analytical and Test Methods

Analytical and test methods used to determine that desired damage tolerance levels are actually provided in structural designs are essentially the same as those used to confirm static strength provisions. Since analysis is empirical in nature and based on test results, few new tests are needed except where structural designs differ significantly from types already tested. Furthermore, since analysis relates to tests where damage is simulated while the structure is undergoing damage

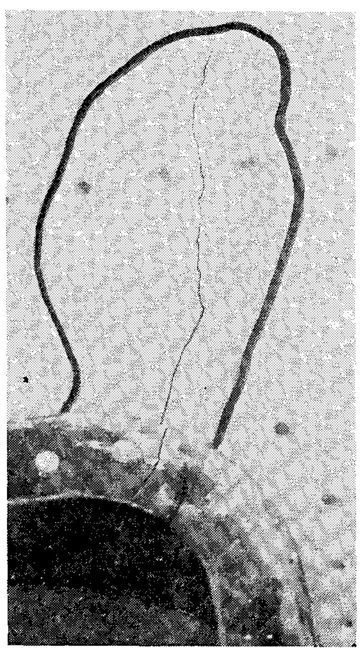


Fig. 10 C-130 fatigue crack.

Table 1 Damage tolerance design criteria

Type of damage assumed in design	Aircraft model	Extent of damage	Ultimate load capability or consequence of damaged structure
Longitudinal crack in pressure cabin	C-130	Circumferential ring failed, accompanied by skin crack across both adjacent skin panels	2.0 g maneuver or 49 fps gust encounter with full cabin operating pressure
	C-140, C-141	Circumferential ring failed, accompanied by 12-in. long skin crack	
	C-5A	Circumferential ring failed, accompanied by skin crack across both adjacent skin panels	
Circumferential crack in pressure cabin	C-130	Longeron failed accompanied by 36-in. long skin crack	2.0 g maneuver or 49 fps gust encounter with full cabin operating pressure differential
	C-140	Longeron failed accompanied by 12-in. long skin crack	
	C-141, C-5A	Any single longitudinal stringer failed, plus skin crack across both adjacent skin panels	
Chordwise crack in wing box upper or lower surface skin	C-130, C-140, C-141, C-5A	One spanwise skin plank fully cracked	2.0 g maneuver or 49 fps gust encounter
Vertical crack in wing box front or rear spar web	C-130, C-140, C-141, C-5A	Web crack extending from upper cap to lower cap	
Chordwise crack in horizontal and vertical stabilizer box upper or lower surface skin	C-130, C-140, C-141	Any single stringer or skin panel fully cracked	
	C-5A	One spanwise plank fully cracked	
Vertical crack in horizontal and vertical stabilizer box front or rear spar web	C-130, C-140, C-141, C-5A	Web crack extending from upper cap to lower cap	
Loss of one complete aileron	C-130, C-140, C-141, C-5A	Surface separates cleanly, not inflicting structural damage beyond immediate supports on parent structure	Roll response is impaired but mission accomplishment is feasible
Loss of one complete elevator or segment	C-130, C-140, C-141, C-5A	(Same as aileron)	Pitch response is impaired but mission accomplishment is feasible
Loss of rudder or rudder segment	C-130, C-140, C-141, C-5A	(Same as aileron)	Yaw control is impaired but mission accomplishment is feasible
Structural failure of any single component involved in transmission of manual control action from flight station to surface affected	C-130, C-140, C-141, C-5A	Fracture or jamming of any single bracket, pulley, cable, push rod, or support thereof	Use of duplicate system by unaffected pilot permits safe completion of mission
Structural failure or loss of hydraulic or electric power to any single power unit utilized in operation of control surfaces	C-130, C-140, C-141, C-5A	Component fracture jamming, hydraulic line rupture, or severance of electrical power for any control surface power unit	Duplicate power units, hydraulic systems, and electrical circuits permit safe continuation of mission
Loss of any single wing trailing edge flap segment	C-130, C-140, C-141, C-5A	Surface separates cleanly, not inflicting structural damage beyond immediate supports on parent structure	Unsymmetrical flight characteristics may be corrected by use of ailerons. Continuation of landing or takeoff operation at degraded field length minimums is feasible
Loss or malfunction of any single wing spoiler segment	C-141, C-5A		
Loss or malfunction of any single wing leading edge slat segment	C-5A		

tolerance design loads, no multiplying factors accounting for dynamic effects of failure under load are necessary. Here, as in static design tests, a margin of safety of zero is permissible.

Prior to initiating analysis, the extent, type of damage, and load levels to be achieved are specified as outlined in Table I. These basic criteria are designed to insure that damage may be readily detectable before structural strength is impaired be-

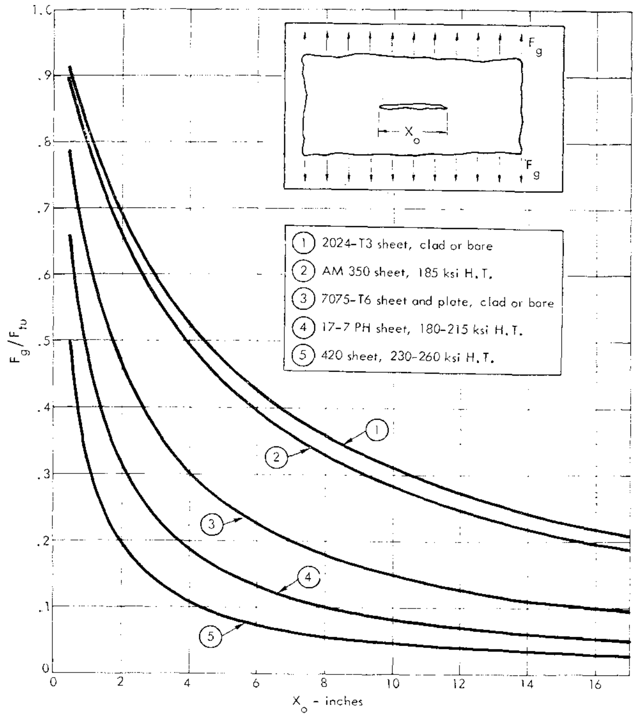


Fig. 11 Tear strength of flat sheet with a crack perpendicular to tension load (infinite sheet width).

yond the point of safe operation. Differences in design detail among various aircraft types account for the small variances in criteria applied to them. Once damage tolerance criteria are defined, the structure may be designed to conform accordingly.

A rather simple concept serves as the basis for determining ultimate strength of a damaged fuselage skin panel. This concept makes use of a fictitious effective width W_e, measured ahead of the tips of a crack in the skin.[1] Ultimate stress is assumed to be sustained over this effective width. Figure 11 illustrates variations of crack length that can be sustained as a

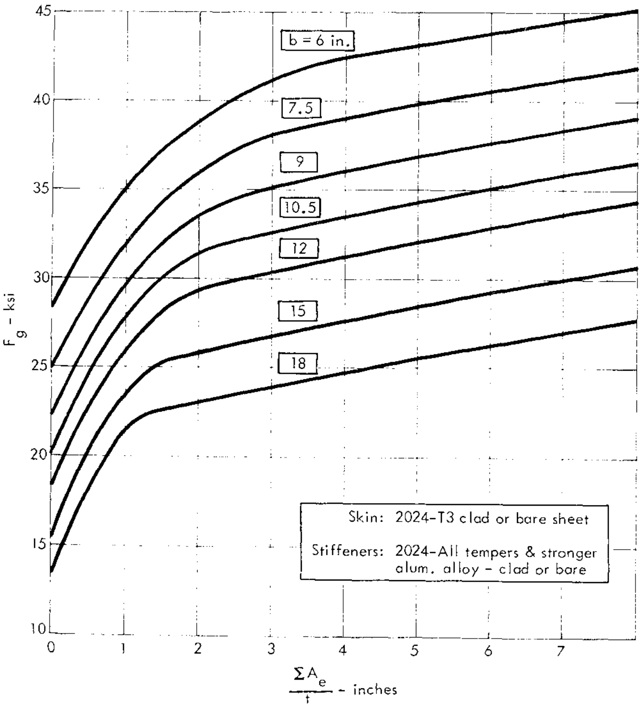
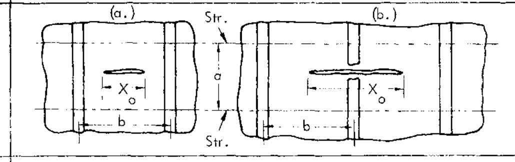
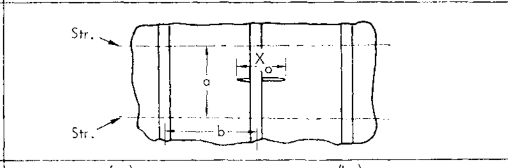
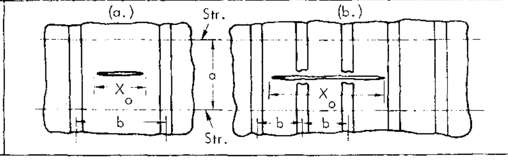


Fig. 12 Ultimate strength of flat, stiffened panels and slightly curved panels with circumferential crack.

Table 2 Curved, stiffened panels with longitudinal crack under internal pressure

Case	Assumed Damage	ΣA_e
1		0
2		A_e
3		$2A_e$

function of material types and gross stresses as determined from tests. Any residual strength analysis of damaged panels must account for significant variations in other parameters such as panel geometry, type and rate of loading, temperature, and sheet thickness. Provided the ranges of important parameters or combinations of parameters are not exceeded, good agreement is realized between predicted stresses and measured stresses.

The approach to determining damage tolerant residual strength considers natural crack stoppers such as stringers and skin splices located perpendicular to the longitudinal axis of the crack. Several typical cracks that may be considered are outlined in Table 2. Consideration must be given to the specific configuration in determining the effective area of the frames A_e for these calculations. This is treated in more detail in Ref. 1. Table 2 is used in conjunction with Figs. 11 and 12 to predict, in a straightforward manner, the residual strength of the damaged panel. The critical pressure at which explosive failure of the skin and frames occurs is then $P_{cr} = F_g(t/R)$. Predicted strength is then compared with required strength to determine the existing level of damage tolerance.

A multielement concept is employed to achieve damage tolerance on a wing surface. Here, the wing cover is fabricated by mechanically attaching several extruded planks together in a spanwise direction. Particular care is taken to insure that each spanwise splice is able to arrest propagation of damage and that attachments are strong enough to transfer load from the damaged panel to adjacent structure without

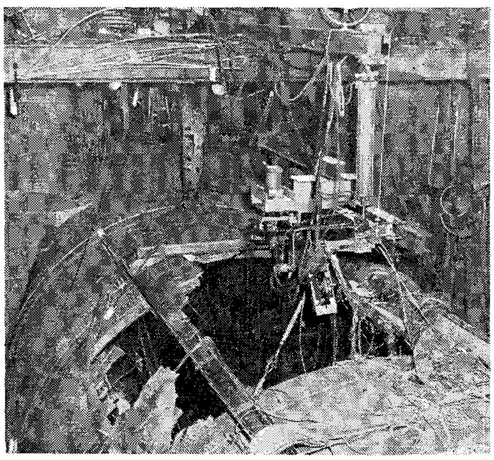


Fig. 13 Results of early C-130 pressure shell test.

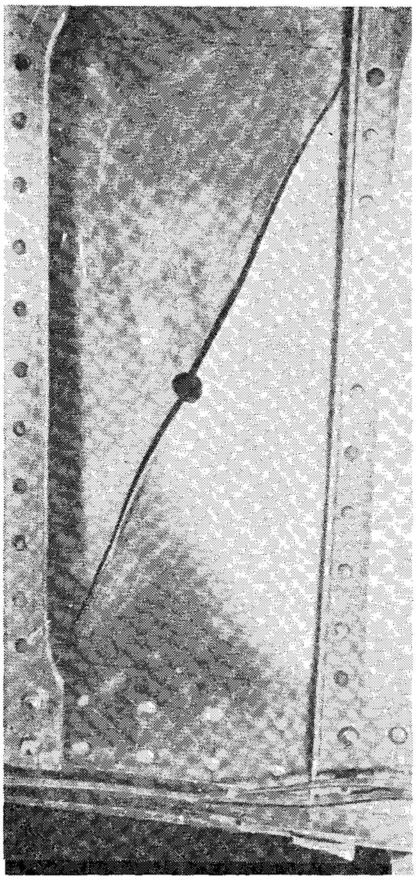


Fig. 14 Damaged spar web under test load.

causing further failures. Lockheed-Georgia requires that all such attachments be designed to fail in bearing. This provides a soft joint that can deform and transfer the required load without producing high local stress concentrations. Component tests are conducted early in the design phase on critical regions of wing covers to verify that desired damage tolerance levels are achieved.

Cracking of a spar web is another typical wing damage mode that may be treated analytically. Damage tolerance is achieved by ascertaining that unfailed wing structure is capable of redistributing the damaged spar web load without causing propagation of the damage. Shear is redistributed by portal action of the spar caps and the crowns of adjacent upper and lower wing panels. If two or more spars are initially incorporated into wing design damage tolerance is obtained by good detail design and without incurring any weight penalty.

Figures 13–17 illustrate tests conducted to verify analytical procedures and determine if desired damage tolerance levels have actually been achieved in design. Figure 13 shows a C-130 pressure shell that attests to the inadequacy of damage

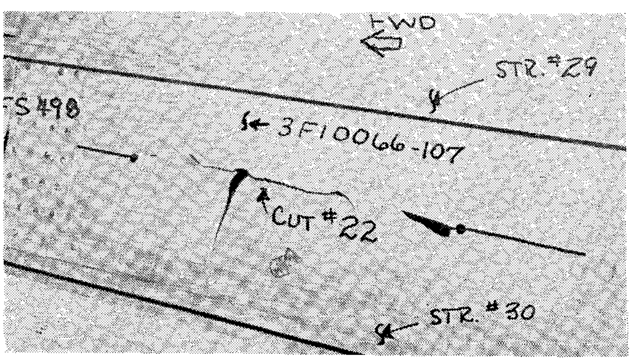


Fig. 15 Results of C-141 pressure shell test.

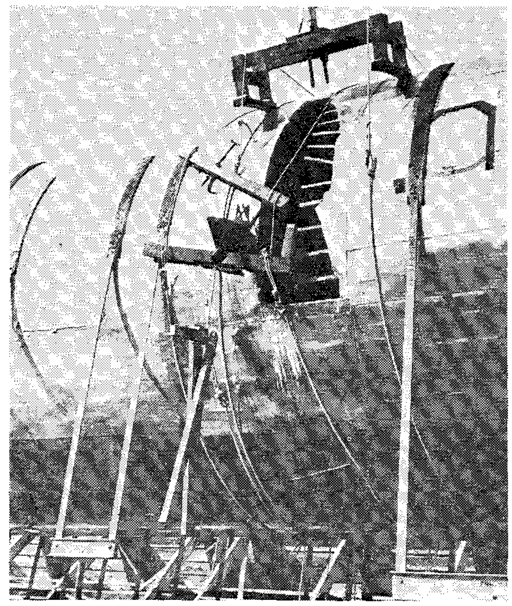


Fig. 16 Results of pressure shell damage tolerance test.

tolerance capabilities of an early design. A damaged spar web under limit load is depicted in Fig. 14.

Figure 15 pictures results of damage tolerance tests on a C-141 pressure shell. Note how the damage was contained by the frame and titanium damage tolerance straps.

Figures 16 and 17 show exterior and interior views of a pressure shell subjected to damage tolerance pressurization tests.

Test spears used to generate damage in structures under test are designed to simulate impact damage when they are shot into and through fuselage skins, stringers, and rings. A rotary saw is used to cut through fuselage skins and stringers to simulate fatigue cracks. In these tests, damage is imparted to structures while under load. These techniques evolved from the need to simulate the effects of propeller parts separating and piercing fuselage shells, or turbine blade failures where fragments pierced shells and wing structures, or growing fatigue cracks. In some instances, cracks have been

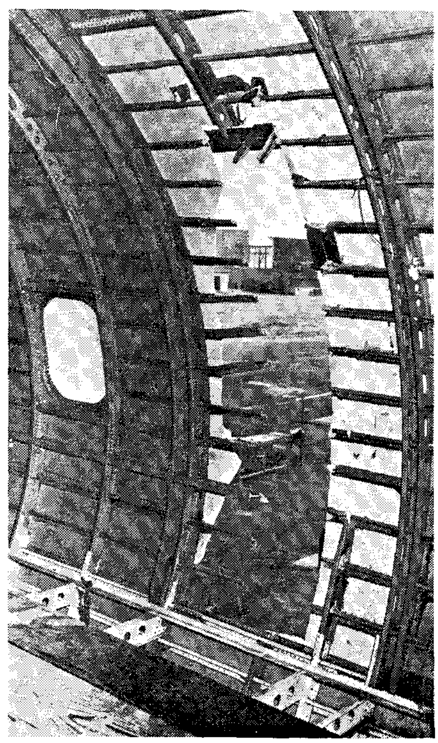


Fig. 17 Results of pressure shell damage tolerance test.

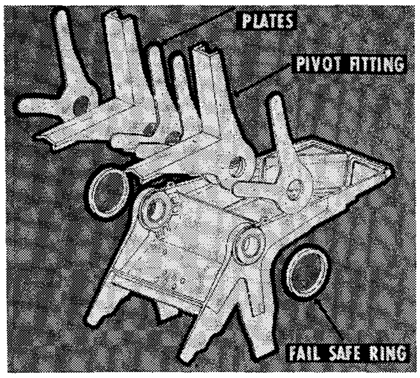


Fig.18 C-5 horizontal stabilizer pivot fitting.

generated in structures under relatively low-level repeated loads until they reached required damage tolerance lengths. The load was then increased to the maximum required level or until failure occurred.

Damage Tolerance Structure

Several practices are followed by engineers to derive damage tolerant structures. Lockheed-Georgia employs positive crack stoppers whenever practical, especially where skin panel dimensions or the dimensions of a structural segment under consideration materially exceed reasonable damage limits. Positive damage limiters or alternate paths are provided where the structure may be subjected to small arms fire. Crack stoppers or alternate load paths are mandatory where a material having a low resistance to crack propagation is used.

As mentioned previously, the use of joints with bearing critical fasteners is one method of positively limiting damage propagation. Where possible, shear critical or tension critical fasteners are avoided. If these are used, the joint involved is treated as being damaged and an alternate load path is provided in design.

In many cases, different materials are used in a structure to take advantage of differences in fatigue or static characteristics. Fail-safe titanium straps, for example, are extensively used in C-140, C-141, and C-5 structures as damage limiters in fuselage skins. These titanium straps, or damage limiters, and the skins they are bonded to are not likely to experience fatigue failures to the same degree at the same instant in time.

The C-130 outer wing upper surface structure consists of four integrally machined wing planks whose span-wise joints are joined by bearing critical fasteners arranged in single rows. Multiple hooks and eyes with a latched tell-tale provide damage tolerance for the C-130 aft loading ramp.

The C-141 horizontal stabilizer trim actuator features dual load paths as a damage tolerance feature. Dual load paths are also provided in the design of the horizontal stabilizer pivot fitting for the C-5 shown in Fig. 18. Damage tolerance titanium straps are used extensively in the C-5 fuselage structure. The damage tolerance straps are located under frame members, as well as midway between frames.

These examples provide some insight as to how damage tolerance concepts are applied to various structures. At Lockheed-Georgia, these concepts supplement, but never replace, fatigue resistance or repeated load requirements imposed on designs. No compromise of fatigue resistance requirements is ever entertained simply because a structure is considered to be damage tolerant. Furthermore, incorporation of damage tolerance design does not eliminate the need for structural inspections or the replacement of faulty or damaged components. Damage tolerance design is applied to enhance reliability, maintainability, and operational safety

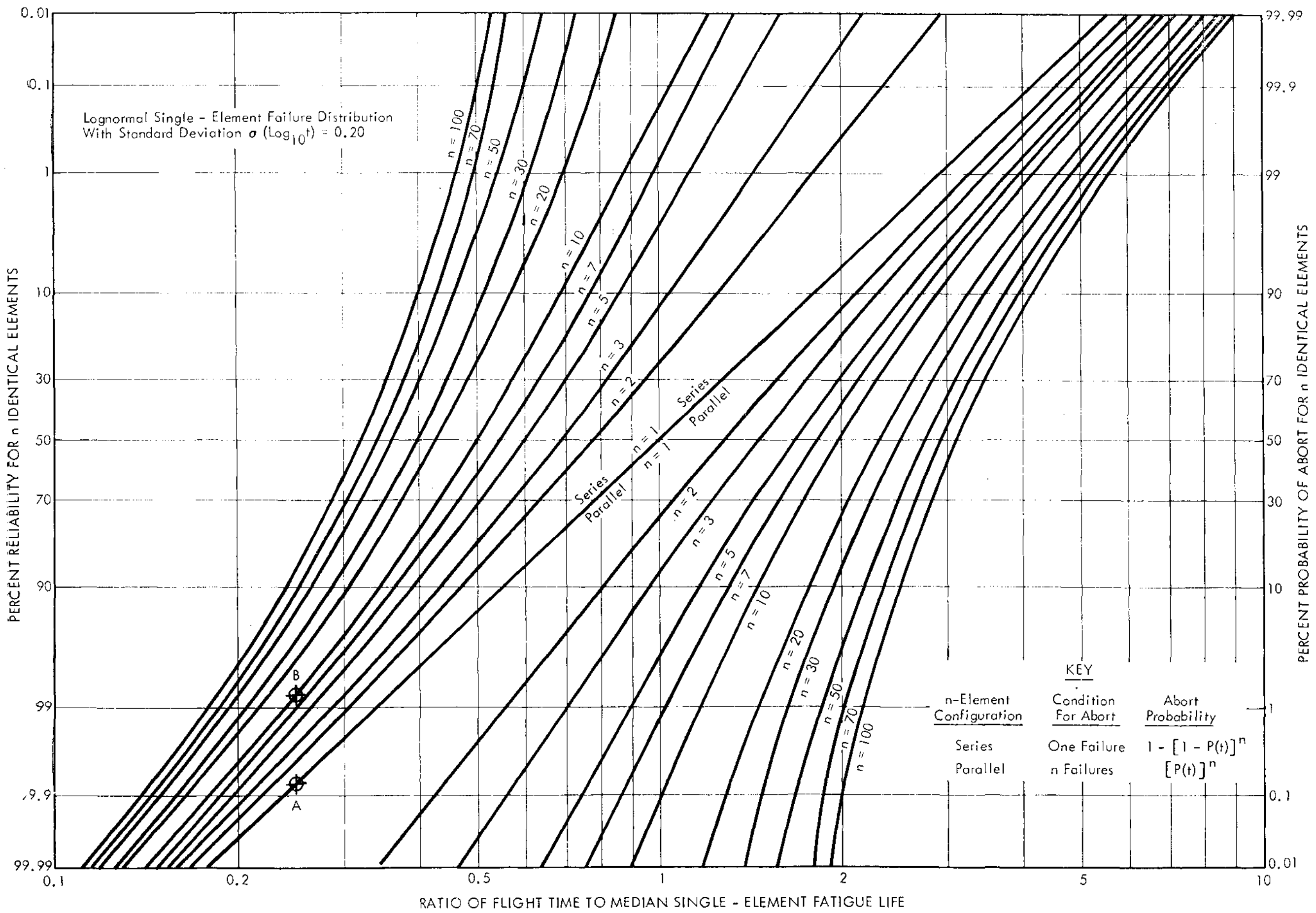


Fig. 19 Abort probability for series and parallel configurations.

and actually assists in establishing orderly and effective inspection and maintenance schedules.

Effects on Airframe Costs and Weight

In general, Lockheed-Georgia, after examining the question in depth, has found that cost and weight penalties incurred as a result of the application of damage tolerance design are so small as to be virtually unidentifiable. For example, it was proposed recently to substitute steel damage tolerance straps for the titanium straps used throughout the C-5. It was felt that the use of steel, in lieu of titanium, would result in cost savings. The proposal predicted that increases in weight and tooling costs by switching to steel would be offset by decreases in metal costs and labor. Further analysis of the proposal indicated that the resultant weight increases would be sizeable and more or less confirmed proposal predictions. Lockheed-Georgia's policy of carefully considering value-of-a-pound-of-weight-saved on the C-5 program, however, led to the final conclusion that the weight increase was not acceptable and that the titanium straps were the most cost effective solution. Taking this a step further, if the damage tolerance straps were not incorporated into the design initially, the structure would undoubtedly be heavier. To provide the positive damage limiting effect of the titanium straps, some other recourse, such as heavier and thicker fuselage skins, would be needed. Aside from this consideration, the titanium damage tolerance straps share the structural load to the extent that they reduce the amount of skin material needed to satisfy static and other strength requirements.[2]

In adopting damage tolerance design, ingenuity can be exercised to the effect that little or no weight penalties are incurred and additive cost increments are minimized. Even where initial damage tolerance design techniques call for increases in weight or costs, or both, alternate concepts are vigorously examined with a view to reducing these penalties to an insignificant level. Fortunately, at Lockheed-Georgia, this objective has been achieved in all designs within reasonable limits.

Most aircraft structures are inherently redundant in that a variety of load paths exist in their design. The additional time and effort to insure that these redundancies satisfy damage tolerance requirements is a small price to pay for increased reliability and operational safety.

A multiplicity of design requirements often favor damage tolerance design. Flutter requirements, for example, may dictate a multispar wing design. Skin gages are often sufficient to minimize buckling under load, greatly alleviating the possibility of skin tears. Structural configurations, such as multisupport ribs, multicontrol systems, and the like, are generally designed to accommodate requirements other than damage tolerance, but are inherently damage tolerant.

All things considered, Lockheed-Georgia experience strongly avows that positive gains resulting from incorporation of damage tolerance design in structures more than offset the generally small costs and weight penalties incurred.

Effects on Reliability and Maintainability

It is estimated that in incorporating damage tolerance design into the C-141 some 500 structural parts were added out of a total of approximately 320,000 parts. This represents a numerical increase of about 0.16%. Adding structure to a design increases over-all complexity and can be expected to have some effect on over-all reliability. When these additions enhance damage tolerance capabilities, however, the effect on reliability can be highly favorable.

As size and number of structural parts increase for larger aircraft, as for the C-5, more engineering opportunities exist to provide redundant load paths. Consequently, the number of parts added to meet damage tolerance requirements should be proportionately reduced.

Since all Lockheed-Georgia aircraft are designed for damage tolerance, the company does not have sufficient service experience with its own aircraft to provide a meaningful comparison with nondamage tolerance designs. Therefore, to assess effects on reliability and maintainability, resort is made to probability analysis.

By pooling various constant amplitude, random fatigue, and program test results, A. M. Freudenthal demonstrated that the probability density of fatigue cracking can be represented by a lognormal distribution where the standard deviation of the logarithm of the number of cycles to cracking exhibits only a slight dependence on fatigue life.[3,4]

Figure 19 depicts curves representing the cumulative probability of abort for series and parallel configuration for the case $\sigma = 0.20$. This is a representative value taken from Refs. 3 and 4. The abscissa of Fig. 19 denotes the ratio of flight time to median single-element fatigue life.

Lockheed-Georgia attempts to design structures so that any single-element failure can occur without an abort. Suppose, however, that this were not the case. Referring to Fig. 19, and assuming a fleet of 2000 aircraft with a service time of 25% of service life demonstrated by fatigue tests (point A, Fig. 19), a probability exists that 2000×0.0015, or 3 aircraft would have aborted if only one critical element were present. If the design contained 10 such critical elements (point B, Fig. 19), the probability of abort enlarges to 2000×0.015 or 30 aircraft. To avoid aborts, it is clearly desirable to provide damage tolerance capabilities.

If these structures were not designed to be damage tolerant where a fatigue crack would lead to complete structural collapse, instead of the aborts there would likely be losses of aircraft. Losses of this magnitude would be prohibitive.

Figure 19 illustrates two major complementary phenomena accounting for the effectiveness of damage tolerance design in reducing abort probability to almost negligible proportions. These are related to the convergence and divergence of series and parallel curves, respectively, at low failure probabilities.

Convergence of series configuration curves implies that many systems may be placed in series without severely reducing over-all aircraft reliability if the reliability of each system is maintained at a high level.

On the other hand, divergence of parallel configuration curves indicates that the period of high reliability can be significantly extended for each system by introducing only a small number of parallel (redundant) elements as long as the failure probabilities of added elements remain small. This may be assured by frequent inspection or replacement of elements.

The left oridinate of Figure 19 indicates that the reliability of an aircraft with a single vulnerable element is 0.9985. With 10 vulnerable elements, this reduces to 0.985. Therefore, a given aircraft operating for 25% of demonstrated fatigue life would have a 99.85% chance of not experiencing an abort if it had only one critical element and only a 98.5% chance if it had 10 critical elements. Here again, damage tolerant design is necessary to achieve high reliability.

Where there are no catastrophically critical elements, the aircraft could be operated with assurance that periodic inspections will disclose potential problem areas prior to development of a dangerous situation. This has been the case with Lockheed-Georgia designed aircraft. Time and again, the C-130, for example, has experienced fatigue damage in service without catastrophic consequences and repairs have been made at the user's convenience without disruption of normal fleet operations.

Reliability and maintainability are closely related. Furthermore, an increase in the reliability confidence level becomes highly important if it is economically practical to operate the aircraft beyond its demonstrated service life, or if significant changes in operating patterns from the normal are experienced. If an aircraft is to be operated beyond its anticipated service life, reliability of the structure increasingly depends on damage tolerance design to compensate for increased probability of structural degradation caused by fatigue or corrosion.

Summary

Damage tolerance design is a valuable asset at Lockheed-Georgia. The hard usage and harsh environments that the company's aircraft are subjected to in service has dictated adoption of damage tolerance concepts in all designs. In-service experience has proven the value of this approach in supplementing conventional design techniques.

Accordingly, the following steps relating to damage tolerance design are assiduously practiced at Lockheed-Georgia:

1) Every basic air and ground load supporting structure is designed to be tolerant of a reasonable amount of damage, regardless of its cause.

2) Achievement of desired damage tolerance levels is demonstrated analytically, and proven, where necessary, by structural tests.

3) Desired design fatigue life is analytically demonstrated with appropriate scatter factors. A factor of 4.0 is used where the analysis is supported by subcomponent tests. Where full-scale components, such as fuselages, wings, and the like, are subjected to fatigue tests, a factor of 2.0 is used in analysis.

4) The completed structure is subjected to complete airframe repeated load tests.

5) Finally, comprehensive maintenance inspection methods, periods, and procedures are established and recommended to the user.

By combining damage tolerance design with conventional fatigue resistance and repeated load techniques, aircraft designed and produced by Lockheed-Georgia have greatly enhanced reliability, maintainability, and operational safety characteristics.

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NASA Programs for Development of High-Temperature Alloys for Advanced Engines

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An intensive research effort is underway at the NASA Lewis Research Center to provide improved materials for the hot components of advanced aircraft gas turbine engines. Research is being conducted both in-house and under NASA sponsorship to develop advanced materials for such applications as stator vanes, turbine buckets and disks, combustion chamber liners, and the latter compressor stages. Major areas of work deal with the development of nickel and cobalt base alloys, chromium base alloys, dispersion strengthened materials, composite materials, and protective coatings. Progress in NASA programs dealing with all these areas is described.

Introduction

To increase performance, advanced aircraft gas turbine engines must use higher turbine inlet gas temperatures. Air cooling permits materials to be used at elevated gas temperatures. However, cooling must be paid for by increased engine complexity and by some sacrifice in performance, when compared with the same temperatures, if achieved without cooling. It therefore remains an important objective to provide materials that will survive at higher temperatures thus permitting higher gas temperatures—either without cooling or at least with reduced cooling requirements. There is a need for improved high-temperature materials for engine components such as stator vanes, turbine buckets and disks, transition ducts, combustion chamber liners, and the latter compressor stages. The NASA Lewis Research Center is actively participating in research to provide advanced materials for such applications both by conducting in-house work and by funding research in other organizations.

Depending upon the engine component, materials must operate at temperatures ranging between approximately 1200 and 2200°F (649 and 1204°C). For such applications as the SST, the operating time requirement is on the order of thousands of hours. Superimposed upon the temperature and time requirements are other factors such as stress, strain, thermal and mechanical fatigue, and the erosive, corrosive effects of high-velocity combustion gases.

Cast and wrought nickel base alloys and to a lesser extent cobalt base alloys have been and continue to be the workhorse materials for the hot components of gas turbine engines. Current nickel base alloys contain a large number of alloying constituents that contribute to one or more of three basic

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