Design for Damage Tolerance

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The emergence of damage tolerance as a primary design consideration for new generations of aircraft is shown to be the result of concern for increased safety and reliability by airframe manufacturers and regulatory bodies. Design criteria applicable to the case of structurally damaged components are defined. Application of "failed single principal member" concept to typical airframe construction is found to have little effect on aeroelastic properties. Internal load paths for damaged structure are described, with typical examples of design practices aimed at minimizing structural vulnerability to damage.

Introduction

THE primary structure of a modern air vehicle represents the careful application of the most advanced knowledge and techniques in the disciplines of metallurgy and engineering mechanics. Basic design objectives applicable to structure of this type may be ranked as follows: 1) minimum weight for critical loadings, 2) stiffness sufficient to preclude flutter, 3) crack-free fatigue life for anticipated service history, and 4) capability to absorb reasonable in-service damage without compromising one-time mission completion.

These objectives interact on each other in complex ways and the air vehicle designer must configure his structural components so as to attain fulfillment of each without undue compromise of the others. The major emphasis on design technology during the formative period of air transport development was on the attainment of minimum weight while retaining adequate aeroelastic stiffness. The accumulation of service experience during the recent accelerated growth of mass air transport has, however, brought into clear focus the vital need for design which, while inherently crack resistant, incorporates a high degree of damage tolerance.

There has been a tendency on the part of regulatory bodies to regard these two qualities as alternate rather than complementary design objectives, but careful assessment of design environments shows that a properly balanced design can incorporate both crack resistance and damage tolerance without significant performance penalty. In the following sections, the essential philosophy and practice of damage tolerant design as applied to large jet aircraft are defined in the following terms: 1) aeroelastic criteria for damaged structure, 2) basic flight and ground criteria for damaged structure, and 3) design philosophy for damaged primary structure.

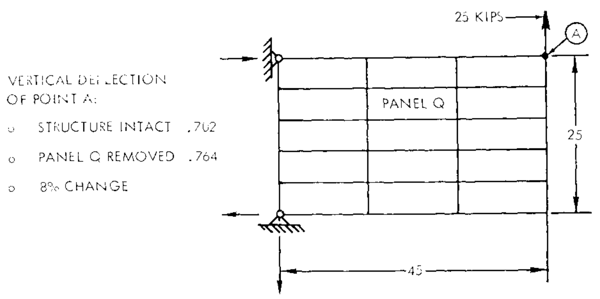

Fig. 1 Example stiffness degradation.

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Aeroelastic Criteria

In most instances, local damage to a multielement structure results in negligible stiffness change. Perturbations to the basic stress fields tend to damp out rapidly, resulting in little increase in the level of stored strain energy. It is thus possible to demonstrate full compliance with basic aeroelastic criteria for most cases of assumed damage, if the structure involved is highly redundant. An example which illustrates this behavior is presented in Fig. 1. The structure consists of a stiffened shear transfer panel comprising thin web with flange material riveted on to form a rectangular grid system. For an applied load of 25,000 lb, vertical displacement of the beam was calculated for the case of fully intact structure and for the case of one complete shear panel removed. The ratio of these deflections shows that the reduction in stiffness caused by failure of a single member is negligible.

Basic Flight Loads Criteria

Failure of any portion of the airframe primary structure will obviously result in some degree of static strength degradation for the component affected. Typically, the strength level of undamaged structure corresponds to the loading which occurs during critical design conditions multiplied by a 1.50 factor of safety. Since it must be assumed that structural damage may occur in flight and that the flight crew is unaware of the occurrence of such damage, ultimate strength for damaged structure is based on flight parameters which are nearly identical to those for intact structure, but at a reduced factor of safety, to reflect the limited period of operation involved. A significant difference between the static design ultimate case for undamaged structure and the damage tolerance condition lies in the possibility of local peaking of internal loads due to sudden rupture of the selected vulnerable element. This aspect of design is customarily handled through application of an additional multiplying factor (usually 1.15) on the design load for damaged structure in the absence of empirical data. When such data exists, actual factors as low as 1.0 are applied. Figure 2 shows the basic flight criteria applicable to strength analysis of damaged primary structure.

LOAD SOURCE	SIGNIFICANT PARAMETERS
SYMMETRICAL MANEUVER	$n_z = 2.0$
GUST ENCOUNTER	$U = 49$ fps @ V_B $U = 33$ fps @ V_C $U = 15$ fps @ V_D
YAWED FLIGHT	80% OF DESIGN LIMIT LOAD

Fig. 2 Damage tolerance flight criteria.

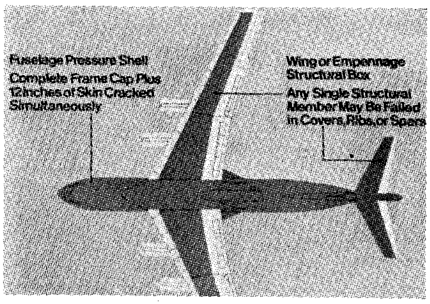

Fig. 3 Regions of primary structure subject to damage tolerance criteria.

Design Philosophy for Damaged Primary Structure

Reference 1 defines the damage which must be assumed if civil airframes are to be shown to comply with Federal Aviation Agency (FAA) regulations. Paragraph 25.571 of Ref. 1 states, in part: "It must be shown by analysis, tests, or both, that catastrophic failure or excessive structural deformation, that could adversely affect the flight characteristics of the airplane, are not probable after fatigue failure or obvious partial failure of a single principal structural element." The manner in which this requirement is applied in design is dependent on the detail configuration of the individual structural component. Figure 3 shows the regions of aircraft primary structure to which this requirement is applicable.

Probably the most vital class of airframe primary structure insofar as damage tolerance is concerned, is the pressurized fuselage shell. In this particular component tensile stresses set up by internal pressure exist over the entire surface area and reach full design magnitude on each flight. In typical fuselage shell construction, large continuous sheets are wrapped around a rectangular gridwork of straight longitudinal stringers and curved transverse frames. Figure 4 shows an interior view of a representative structure of this type.

The approach taken in developing damage tolerant pressure vessel structure is that of "obvious partial failure," since individual skin panel widths of up to 60 in. are used in the structure, and a continuous crack of this length would not only be detectable at an early stage, but highly improbable in view of the existence of the substructure grid. A logical degree of damage for this category of structure is illustrated in Fig. 5. The internal frame member is assumed to be fully cracked, and the skin panel to which it attaches has developed

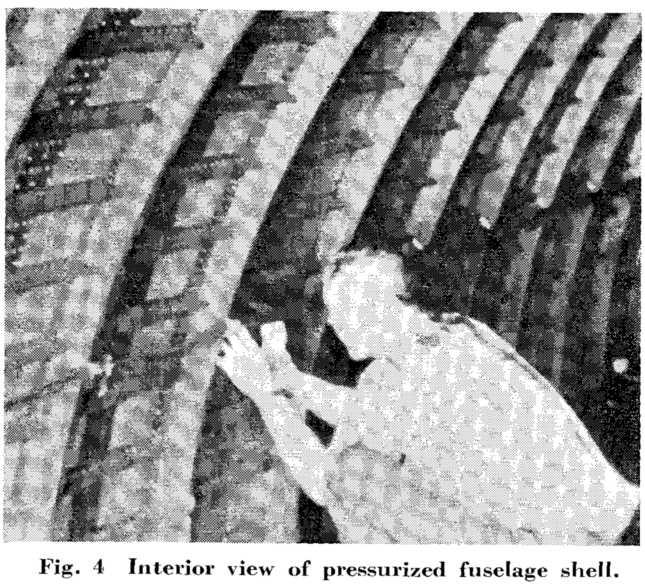

Fig. 4 Interior view of pressurized fuselage shell.

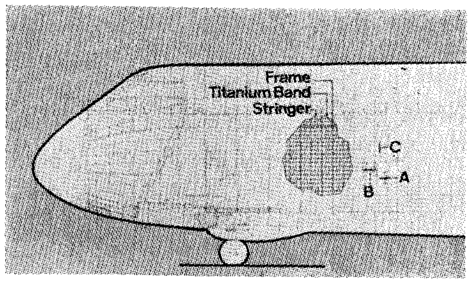

Fig. 5 Assumed damage in fuselage shell.

a longitudinal crack extending both ways from the frame line of attachment and terminating in the vicinity of the next transverse member.

If the shell stiffening arrangement involves relatively wide-spaced circumferential rings, the skin crack defined above may become excessively long and redistribution of pressure-induced skin load will result in the addition of excess axial load carrying material to the flanges of the rings. This penalty can be avoided through the use of intermediate crack-stopping straps located midway between rings. An installation incorporating such straps is shown in Fig. 6.

The basic geometric and metallurgial parameters for this installation are summarized in Fig. 7. The selection of aluminum alloy 7079-T6 for the skin panel is based on the superior resistance to crack propagation exhibited by this alloy over 7075-T6. The slightly inferior static strength properties of 7079 material are offset by the reduced density (0.099 lb/in.^{3} for 7079) and by the limitation on pressure-induced tensile hoop stress which is imposed by crack growth-rate considerations.

The crack-stopping straps must combine high axial load capability with minimum cross-section thickness in order to lie under the longitudinal stringers. Alloy titanium (6-4) is utilized for these straps in the structure of Fig. 7. The required thickness of 0.020 in. does not necessitate joggling of the stringers. Additional skin perforations by attachments is avoided at strap installations by bonding the straps to skin panels. Bonding is performed at temperature prior to installation of the skin.

The longitudinal stringers and circumferential rings incorporated in the structure of Fig. 7 are configured on the basis of static strength considerations only and are fabricated from 7075-T6 aluminum alloy extrusion. Particular care in design is necessary in the establishment of structural continuity in the skin plane at ring-stringer intersection points.

For the case where an advancing longitudinal skin crack terminates at such an intersection, a large axial load caused by skin pressure-induced loading redistribution must be transferred across the stringer by the circumferential ring skin attaching flange. This requirement has been satisfied

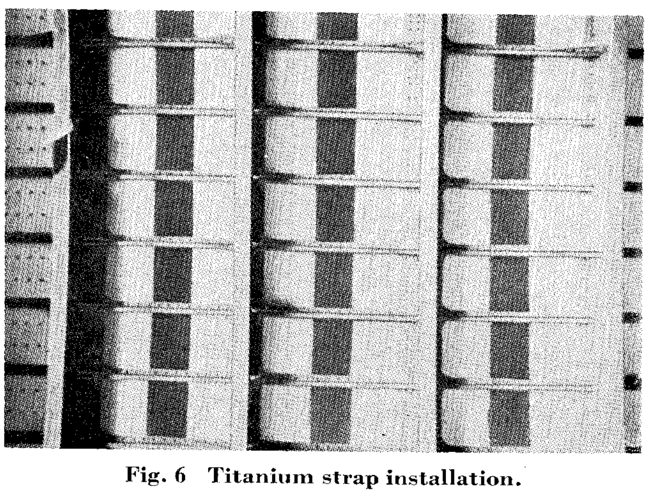

Fig. 6 Titanium strap installation.

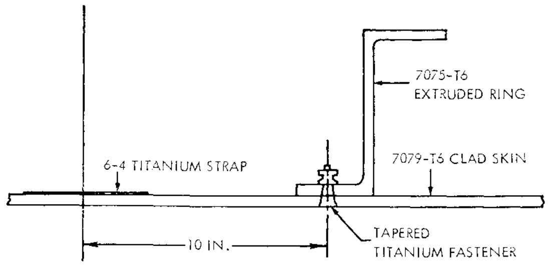

Fig. 7 Design parameters for fuselage shell.

in the configuration of Fig. 7 by splicing the ring flange with extended integral tabs on the stringers. Figure 8 shows the local structure at such an intersection.

Damage Tolerant Wing Box Cover

A second category of structure for which the preservation of inflight structural integrity is of vital concern is the wing box. Principal wing bending loads are resisted by axially stressed wing box cover elements. The design approach applicable to wing box surface panel construction can be that of the failed "single principal element." A configuration which is particularly adaptable to this concept of damage is that shown in Fig. 9. The surface is divided into spanwise panels or planks by parallel splice lines of attachment. Each plank comprises an extruded integrally stiffened plate of 7075-T6 aluminum alloy which has been taper-machined to the local static design load requirements throughout the span. Any chordwise crack development is assumed to extend across the full plank width. The axial load carried by such a plank is assumed to be redistributed to the adjacent planks through the existing attachment rows inboard and outboard of the crack.

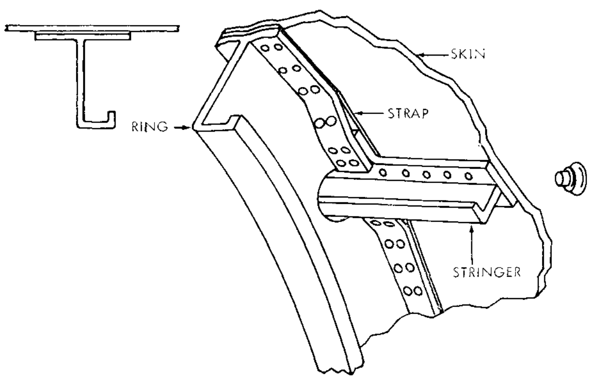

Fig. 8 Splice of fuselage ring flange.

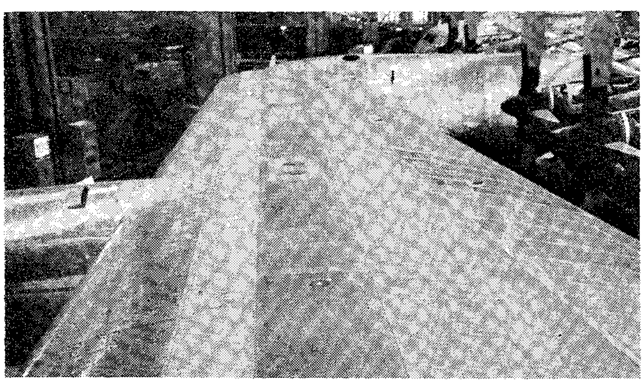

Fig. 9 Wing surface panel arrangement.

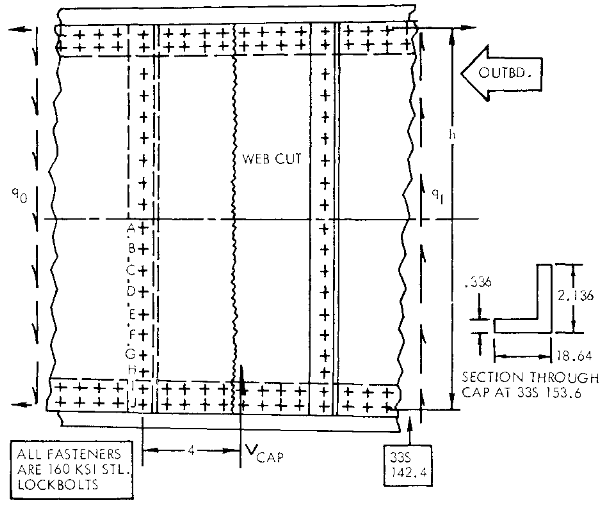

Fig. 10 Empennage shear beam loading.

The wing box shear beams are assigned the degree of damage illustrated in Fig. 10. The damage shown consists of a vertical crack extending across the full depth. The design configuration of the beam involves the use of relatively closely-spaced vertical stiffeners which are securely attached to the upper and lower flange elements of the beam. It has been established experimentally that the existence of a full depth web crack in the center of a typical bay of such a beam does not significantly affect shear stress levels in the immediately adjacent web bays. The load path for shear appears to be a form of portal-frame action involving secondary transfer through the upper and lower beam flanges.

BEAM CONDITION	MEASURED SHEAR STRESSES – % OF CALCULATED	
	GAGE 1	GAGE 2
INTACT	93	93
CUT	122	103

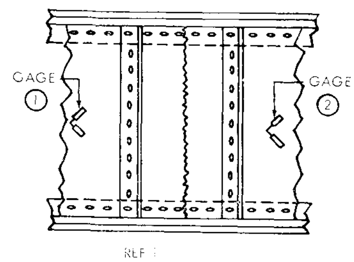

Fig. 11 Empennage shear beam test data.

Figure 11 shows the test structure from which these findings are derived, together with the stresses measured in shear webs located adjacent to the bay containing damage. The principal effect of the damage is an increase in shear flow over that predicted by theory in these adjacent bays. It is expected that further analysis will show that the shear redistribution to adjacent ribs both inboard and outboard of the cut web causes these increases. The magnitude of increase is not sufficient to cause exceedance of design ultimate values, however.

Conclusion

By careful attention to detail design, the highly loaded primary structure of modern aircraft can sustain reasonable amounts of damage without losing the capability of continued safe flight. Governmental regulations now in existence provide adequate damage tolerance criteria which, when applied knowledgeably, do not result in significant structural weight penalties.

References

[1] "Airworthiness Standards: Transport Category Airplanes," Federal Aviation Regulations, Pt. 25.

[2] "C-5A Empennage Test Box," Rept. ER8607, Jan. 1967, Lockheed-Georgia Co., Marietta, Ga.