ELASTIC STABILITY CONSIDERATIONS IN AIRCRAFT STRUCTURAL DESIGN

By Richard L. Schleicher

North American Aviation, Inc.

General instability in shell type structures is a relatively infrequent phenomenon. By general instability is meant that a complete section of a structure becomes elastically unstable, as compared with local crippling or buckling. The first step in design is to rough size the structure and determine its compatibility with air vehicle configuration and systems installations. After this step has been completed, it is possible to survey the layouts to determine whether general instability is likely to occur. The general parameters of length, diameter, frame spacing, element stiffness, etc., are then investigated to determine if a general failing stress exists which is less than any local column, beam column, or panel failing stress. In this way, a well-balanced design is determined and general stability failing modes eliminated.

The structural designer has a fairly wide selection of basic parameters during the early phases of design. These diminish, however, as the design progresses and consideration for systems installations becomes more important. In many cases the ideal arrangement of structural elements soon gives way to marked discontinuities. The classical example of a circular cylinder with symmetrically spaced elements is seldom realized. Uniform frame sections and spacing give way to deeper utility frames at random spacing. The effective stiffness of the deeper frames may be 20 - 50 times greater than the minimum required to prevent instability, considering the general proportions of the shell. In like manner, the uniform distribution of longitudinal stiffeners is replaced by a system of longerons supplying the axial compressive elements. Major frames serve as base points in support of beam-column longerons. The intermediate frames give elastic column support to the longerons and form boundary members for the shear panels in the shell covering. It has been found that longitudinal stringers become superfluous and are more difficult to accommodate than longerons, even though these too may be interrupted. Without longitudinal stringers, frame stiffness may actually be increased through simplified design.

A situation frequently arises in which frame depths must be arbitrarily increased to accommodate aircraft systems installations; namely, pipe lines, conduits, etc. This results in deep frames having unstable inner flanges. When the auxiliary systems fail to supply the necessary stability, other means must be employed. Straps of sheet metal are sometimes fastened to the inside flanges of frames, lacing them together and terminating at convenient major frames. These latter ties serve as node points to the frame flanges.

When major frames or bulkheads are spaced too far apart to stabilize the longerons laterally, other means must be provided. Each design must be inspected to ascertain whether the longeron is supporting intermediate frames or vice versa. Normally, the side skin will give continuous vertical stability to the longerons with the intermediate frames serving to stabilize the inside flanges. Cross plane flooring or shelving generally serves this purpose. In some instances, the lower longerons on both sides of the weapons bay in bombers have been stabilized by the doors attached to them. Continuous piano hinges are excellent for this purpose. During the transition from closed to open, the longeron has reduced lateral stability. However, since bombing runs are made during unaccelerated flight conditions, high loads are not encountered during the quick opening and closing cycles. In general, careful attention to frame design offsets instability failures in fuselage structures.

Wing structures are seldom affected by general instability if proper attention has been paid to rib and spar design. In thin wing box structures, bending deflections may be appreciable. Rib designs must be checked for the compressive loads brought about by the surface deflection. This consideration usually provides enough stiffness to support the wing surface against general instability. When closely spaced ribs are not feasible in thin wings, the skin thickness is increased and surface stiffness is achieved through the use of multi-spars. The problem then reverts to one of simple panel instability. There are a number of wing box structures in service today whose compression surface allowable stress is equal to the compressive yield strength of the material used. This has been achieved through the use of thick milled skins and multi-spar support.

The structural designer is not without constant regard for the general instability problem. He is frequently guided by his sense of proportion based on long experience. This sense has often resulted in stiffening certain areas and thus forestalling the progression of incipient failures.

TESTS OF CURVED PANELS

The structures engineer must frequently obtain verification of the effects of several variables in order to proceed with a particular design. Typical of the above is a series of tests to determine the effect of intermediate frame spacing on the strength of curved sheets in axial compression. The following concerns the test of a group of 4130 steel curved panels with varying frame spacing. The material was heat-treated to 125 - 140 ksi. The main frames were 2 inch "Z" sections spaced 18-7/16 inches on center and made of the same material. Light intermediate "J" section frames were made from 301 CRES steel. The 28 inch radius of curvature and main frame spacing were held constant - only the intermediate frame spacing was varied. The width of all panels was approximately 67 inches and all frames were riveted to the skin panels. The free edges were lightly clamped between two 3 x 3 x 3/8 steel angles. All specimens were tested in axial compression between two parallel steel loading plates 10 inches thick. The panels were made of two pieces of steel sheet welded longitudinally to form a single panel and heat-treated after welding. The warping caused by heat-treating did not result in flat spots greater than 1/16 inches in depth.

Each specimen failed by inward buckling of large sections of the panel. In each case, the post buckling load was also obtained and this, it was found, could be repeatedly attained after initial failure. The post buckling stress was equal to or greater than 68 per cent of the initial buckling stress for the main frame spacing of 28 inches. The smaller intermediate frame spacings resulted in a value of 79 per cent or greater. The buckling characteristic of the closely spaced frames was a gradual type of failure rather than sudden buckling. The light "J" section frames were quite effective in attaining high buckling stresses. Two of the specimens were reworked after initial failure by adding the intermediate frames and retested. The reworked specimen sustained from 19 to 60 per cent greater load.

One steel panel was subjected to heavy hammer blows to ascertain the additional effects of surface dents. The initial buckling load was reduced approximately 25 per cent whereas the post buckling load was only reduced approximately 4 per cent over the more perfect specimen. There was no reduction in post buckling load over the initial load. The results of these tests showed:

(a) The initial strength of a curved panel depends on the initial surface conditions.

(b) The post buckling strength, except for the damaged panels, is less than the initial buckling strength.

(c) Light intermediate frames add greatly to the buckling strength of curved panels.

(d) The strength of the panels after initial buckling was consistent and reproducible.

The pertinent stress coefficients derived from the test data are plotted in Figure 1. Also shown is the characteristic curve taken from Figure 3 of NACA Report 874. It will be noted that the test points fall considerably below the NACA curve. This is no doubt due to the initial condition of the specimens which closely approximated actual construction. The upper symbols indicate initial buckling and the lower symbols post buckling. The familiar stress coefficients are the same as found in Reference 1.

TEST OF A FUSELAGE SPECIMEN

It is sometimes necessary following detail element tests such as the curved panel tests just described, to proceed with a representative component test specimen. This was done in the case of the X-15 fuselage which is essentially a monocoque cylinder subjected to a multitude of loadings and environment.

The test component was a generalized section of the X-15 fuselage in the region of the integral propellent tanks. It was basically a 56 inch diameter cylindrical shell 80 inches long with a 14½ inch diameter inner cylinder, two toroidal bulkhead frames, and two side fairings. The details are shown in Figure 2.

The outer shell was .093 inch Inconel X sheet formed to a 56 inch diameter cylinder. Welding was used except for a mechanical joint at Station 60. Beads were formed in the side areas to provide increased stiffness in the circumferential direction and thermal relief in the longitudinal direction. One typical wing carry-through frame was welded to the shell at Station 16 with four attaching fittings for external loads. Four longitudinal angles were welded to the outside of the shell, acting both as longerons and fairing attachments. The main shell is used to contain LOX. The 14½ inch inner cylinder, which is used to store helium, was fabricated from .043 inch Inconel X material. Zee section circumferential stiffeners were spotwelded to the outside of the inner cylinder. One of the toroidal bulkheads consisted of two circular segments made from Inconel sheet welded to the large shell wall and the inner cylinder respectively. To these was riveted a .050 inch thick 7075-T6 clad aluminum alloy section to complete the torus. The other torus was formed in two segments welded to the outer and inner cylinders and along a circumferential seam joining the two segments. The side fairings used to house control elements and plumbing details in the X-15 completed the assembly. However, only one fairing duplicated the X-15 design which consisted of a flat outer sheet reinforced by a corrugated inner sheet - both made from Inconel X.

Tests were conducted in five parts. These included pressurization, external loads, and elevated temperature tests. A general view of the specimen is shown in Figure 3.

The first test was an internal pressure test to determine the collapsing strength of the aluminum alloy torus. Pressure was applied above the aluminum torus and failure occurred at 10.7 psi which was 71.3 per cent of the required pressure. Radial stiffeners spaced 15 degrees apart were required to achieve the 15 psi pressure required.

The object of the second test was to test the welded joints of the Inconel torus, the welded joints of the outer shell, and the strength of the inner shell under a collapsing pressure. Positive pressure was applied internally between Station 0 and the Inconel torus. Failure occurred by compression buckling of the inner shell at 80 psi. Failure was in a multi-node fashion as indicated by theory. As a result of this test the inner cylinder stiffener spacing was reduced in order to carry the required ultimate pressure of 111 psi. Both the Inconel torus and main outer shell withstood the pressure without failure. In addition, 100 cycles of limit pressure (78 psi) were also applied without damage. Likewise, a design negative pressure test of the Inconel torus to 20 psi resulted in no indication of failure.

Following the above tests, a negative pressure test of the outer shell was conducted. For this test, a sealed bulkhead was attached at Station 80. To prevent premature failure, the specimen was filled almost full with deionized water. The space at the top was evacuated to -6 psi with no failure resulting. Due to the head of water within the specimen, the net pressure at the bottom was -3.1 psi. This test was sufficient to demonstrate the collapsing strength of the outer shell.

The next order of tests included both room and elevated temperature load tests of the wing carry-through frame. The specimen was loaded through a set of loading beams attached to the wing fittings. For the room temperature tests, the frame was loaded to failure at 93 per cent of the design ultimate load. Since failure occurred only on one side, a repair was made and retested to a gradient across the frame of 555° F. The gradient was obtained by first cooling the inner flange of the frame with a fine spray of liquid nitrogen. Quartz glass radiant heaters were used on the outside. The temperature gradient was programmed to achieve a maximum value in 300 seconds. Limit load was first applied at room temperature and, while holding the load constant, the temperature gradient was achieved. Next the load was increased and failure took place at 90 per cent design ultimate load. The failure was at the same corresponding location on the side of the frame opposite to the previous failure. A reinforced frame was used in the final design. It is interesting to note that there was only a 3 per cent difference between the identical failures occurring at both room temperature and with the gradient noted above.

The specimen was then loaded in vertical bending (side fairings at the neutral axis) both at room and elevated temperatures. Moment was applied through a steel bulkhead and loading beams attached to Station 80. It was necessary to precool all four longerons prior to heating to achieve the proper temperature gradient. The room temperature load tests were carried to the required ultimate moment of 6,300,000 inch pounds without failure. The elevated temperature tests, to a gradient of 550° F., were run at increasing load levels (10 per cent increments) with a cool-down after each load level. In every case the cool-down was followed by load application and then the heat reapplied. Failure of the outer shell occurred at 160 per cent design ultimate load as the heating cycle was applied.

The remaining bending test was a side bending case in which the side fairing was placed in compression. This test, like the preceding one, was conducted both at room and elevated temperatures. The test setup was also the same as for the case of vertical bending. In addition to the side bending moment of 2,940,000 inch pounds, an axial compression of 9,900 pounds was applied to the side fairing. At room temperature, the main shell withstood 100 per cent side bending moment. The side fairing however failed at 43 per cent maximum load through the spotwelds connecting the outer skin to the corrugated inner skin. The spotwelds were replaced with monel rivets and the fairing failed at 85 per cent design ultimate load by crippling of the outer skin in an area which was beyond the support of the inner corrugated skin. The spotwelding used in the specimen was changed to a stitch weld of greater strength in the vehicle side fairing to achieve 100 per cent strength. During the elevated temperature part of the test, the outer shell withstood 150 per cent design ultimate bending moment without failure. However, during a subsequent loading the outer shell failed at 140 per cent design ultimate load.

The final major test was a transverse shear test in which an ultimate load of 46,700 pounds was applied at Station 80 and reacted at the floor mounting. This test was conducted at room temperature and was completed without failure.

This series of tests is typical of the procedure followed in the design of an airframe of unusual structural characteristics.

TESTS OF A LARGE CYLINDRICAL SPECIMEN

The purpose of this test was to evaluate the longerons, frames, and frame stabilizing straps of a large fuselage specimen shown in

Figure 4. This specimen was circular with a diameter of 100 inches and length of 64 inches. The shell wall was 6V - 4AL titanium riveted to 4AL - 3Mo - 1V titanium frames and ten H-11 steel longerons. The specimen was subjected to bending about mutually perpendicular axes, both singly and in combination as listed below:

$$\begin{array}{lll} \text{I} & - M_y = 45{,}000{,}000 \text{ inch pounds} & S_z = 210{,}000 \text{ pounds} \\ \text{II} & - M_z = 32{,}500{,}000 \text{ inch pounds} & S_y = 60{,}000 \text{ pounds} \\ \text{III} & - M_y = 42{,}000{,}000 \text{ inch pounds} & \left.\begin{array}{l} S_z = 91{,}000 \text{ pounds} \\ S_y = 50{,}000 \text{ pounds} \end{array}\right\} \\ & M_z = 23{,}000{,}000 \text{ inch pounds} & \end{array}$$

The basic internal stresses were obtained by elementary bending theory plus the effects of diagonal tension as these might affect the longerons. A multi-support beam-column analysis was developed for the longeron. Frame loads were derived considering the skin diagonal tension and the frame solution was based on the theory of minimum strain energy. The calculated strength of the frame inner flanges indicated low compressive strength and these were to be investigated as part of this test.

In calculating the beam column strength of the longerons, the interaction effects of the other longerons were not considered. The frame stiffnesses at each longeron were calculated and used as flexible supports. Considering the frame and longeron geometries, the deflection pattern of the longeron could be determined. Using an iteration process, the critical load of any longeron could then be determined. The predicted buckling stress of the side longeron was 195,000 psi. At 90 per cent of the side bending test (.90 x 32,500,000 inch pounds), visual observations gave signs of impending failure. The strain gages indicated stresses of 220,000 and 120,000 psi measured on opposite sides of a longeron element. The calculated stress level, however, was 180,000 psi. It was concluded that the calculated and predicted failing stresses were in good agreement. The strain gage readings also indicated a decrease in stress at the 90 per cent load increment. The test was stopped at this load level.

At the same time that the side longeron was at the point of failure, the upper and lower shoulder longerons were also indicating failure because of increased deflections. The critical predicted buckling stresses were 120,000 and 182,000 psi, respectively, for these longerons. The calculated stresses at this point in the test were 104,000 psi and 171,000 psi. Strain gages monitored during the test indicated 100,000 and 160,000 psi. The test in side bending was not carried beyond this point since design requirements were met and other tests were still planned. The test did indicate that the calculated and predicted critical longeron loads were close, considering that failure was imminent and not actual.

An energy solution of typical frames, considering the skin diagonal tension stresses and longeron effects, indicated that the optimum strap spacing for the inner flanges was 11.5 inches. After sustaining the maximum vertical loading condition of 45,000,000 inch pounds moment and 210,000 pounds shear, one half of the straps were removed and the frames continued to carry this test loading. Strain gage readings indicated impending failure of the flanges. The original strap spacing was resumed and failure occurred at 110 per cent of the above loading condition in the outer flanges of the frames. This failure was considered a general instability failure of the shell. Once the lateral strap spacing was determined, the calculated and predicted frame flange loadings of 1659 pounds and 1980 pounds, respectively, could be determined. Considering the many variables involved, this engineering approach to a typical shell instability problem proved adequate.

RECOMMENDATIONS FOR RESEARCH

(1) Expand present theory and techniques to include noncircular and unsymmetrical sections, especially with areas of re-entrant curvature.

(2) Expand present theory and techniques to include sandwich shells where the shear rigidity of surface elements is significant.

(3) Develop structural concepts embodying a high degree of post-buckling strength to minimize the hazard of immediate total collapse.

(4) Develop and verify methods of analysis to include interaction effects of multiple loading.

REFERENCES

(1) NACA Technical Report No. 874, "A Simplified Method of Elastic Stability Analysis for Thin Cylindrical Shells", by S. B. Batdorf.

(2) NASA Technical Note 3735, "Bending Tests of Ring-Stiffened Circular Cylinders", by James P. Peterson.

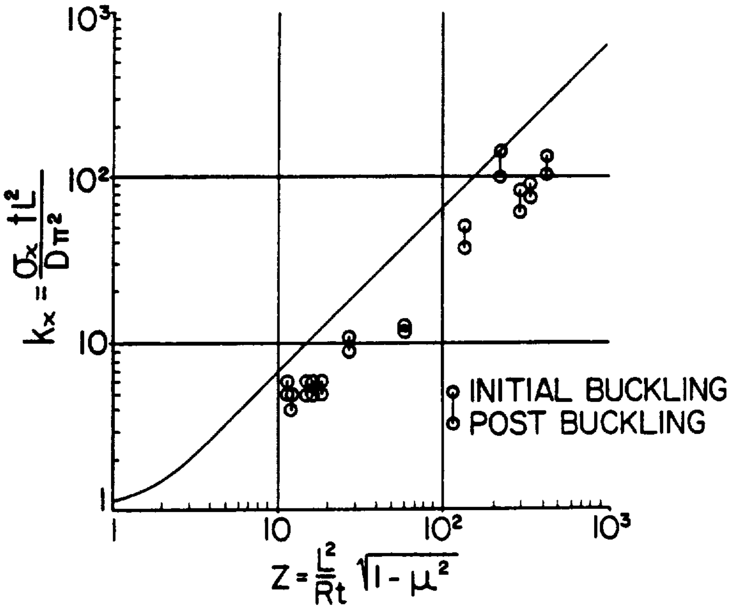

Figure 1 - AXIAL STRESS COEFFICIENTS FOR CURVED PANELS

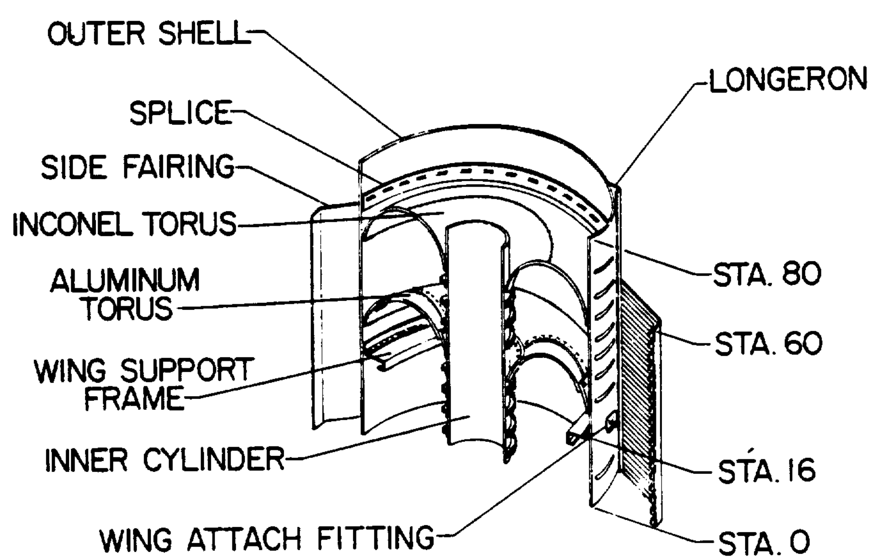

Figure 2 - X-15 FUSELAGE SPECIMEN

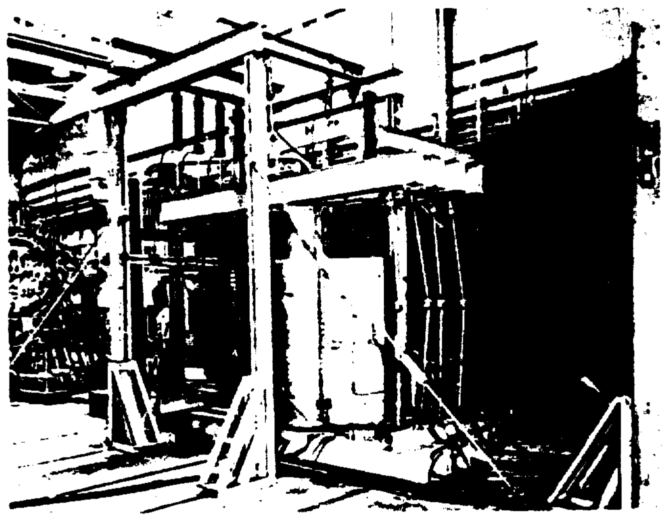

Figure 3 - X-15 FUSELAGE SPECIMEN

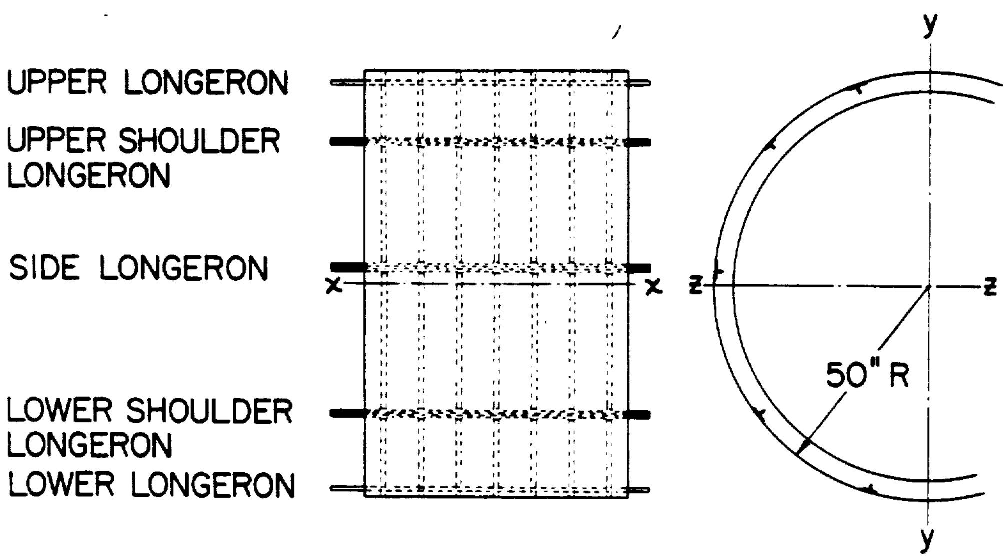

Figure 4 - FUSELAGE SHELL SPECIMEN