4.2 WING AND EMPENNAGE CONSTRUCTION AND STRENGTH

The Model 35 wing and empennage are conventional metal stressed-skin constructions. This section addresses the questions of how such constructions carry flight loads, how they fail, and how their strengths are determined.

A stressed-skin construction[19-21] consists of a structurally secondary nose box, a primary load-carrying structure, and secondary trailing edge structure which includes the control surfaces (Figure 4-11). The entire aerodynamic surface is subjected to flight loads caused by the difference in aerodynamic pressure acting on the upper and lower surfaces. The secondary structures transmit their shares of these loads chordwise to the primary structure, as shown in Figure 4-12 for a typical nose box section. The primary structure transmits these loads, together with its own share of the pressure load, spanwise to the fuselage.

The primary structure consists of front and rear spars, upper and lower skins, and either internal ribs or skin beads (Figure 4-13). The spars and skins are the major load-carrying components of the primary structure. The main function of the ribs or skin beads is to keep the shape of the structure stable.

The loads acting on a typical cross section of the primary structure are shown in Figure 4-13. The spanwise distribution of these loads causes two kinds of structural deflection: bending and twisting. Figure 4-14 illustrates these deflections for a wing subjected to normal flight loads. In bending, the wing deflects upward from the plane of its planform, while the twisting is a rotation about the spanwise axis. The empennage deflections are similar, but neither wing nor empennage deflections are large enough to be visible under normal conditions.

The loads and deflections have the following effects in terms of internal structural loads. Bending creates tension in the lower skin and the lower halves of the spars, compression in the upper skin and the upper halves of the spars, and shear in the spar webs. Twisting creates shear in the skins and spar webs. Where an edge of a skin is riveted to the internal structure, the skin shear is transmitted via bearing loads between the rivets and fastener holes.

The maximum internal loads accumulate on the wing at an inboard location. The maximum structural stresses (loads per unit cross sectional area) may occur outboard of the maximum load station, however, if the skin thicknesses or spar sections decrease at some station outboard of the fuselage.

Flat-stock components such as spars and skins fail in tension by tearing. Tearing failures usually start from stress concentration points such as regions near fastener holes. As long as the stress concentration factor is known, however, the material strength determined from a tensile test specimen can be used to estimate the component strength by calculating the local applied stress in terms of the flight loads (Figure 4-15).

Components subjected to compression will fail by buckling rather than tearing if the section thickness is small compared to the unsupported span. The component is supported on any line along which attachments restrain it against lateral (out-of-plane) deflection. For example, the upper skin panel shown in Figure 4-16 is supported spanwise by rivetting to the spars and chordwise by rivetting to the ribs (see also Figure 4-13). The unsupported skin area inside these attachment lines will buckle as shown if the applied compression stress exceeds the skin buckling strength.

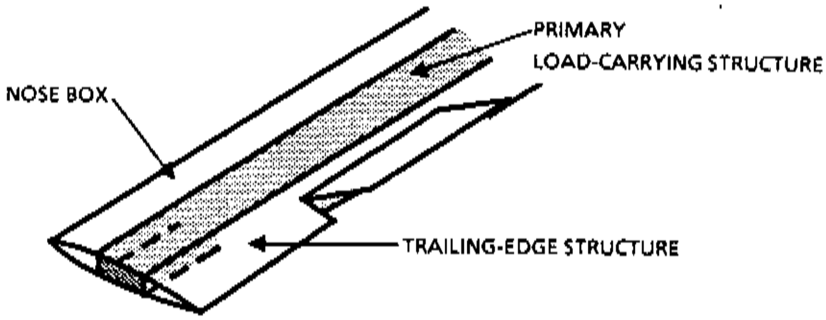


FIGURE 4-11. MAJOR STRUCTURAL COMPONENTS OF AN AERODYNAMIC SURFACE

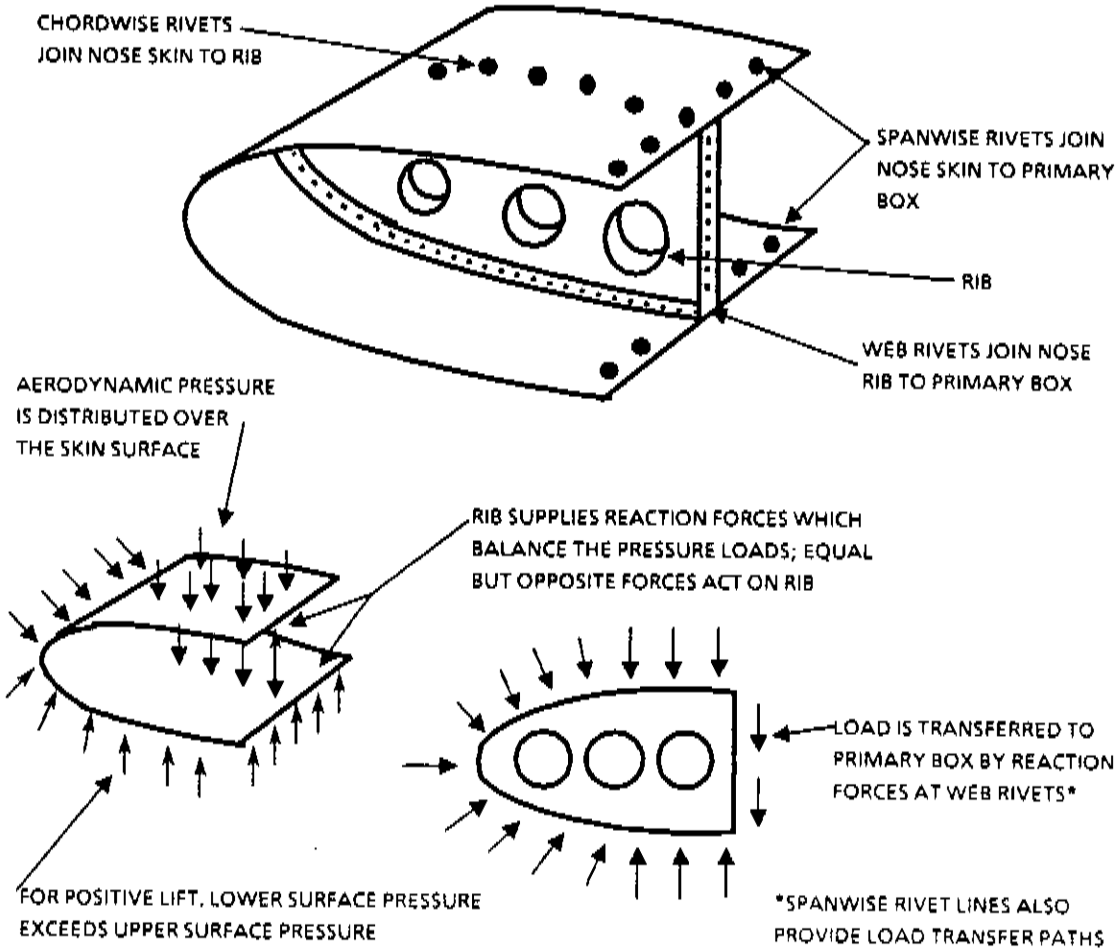


FIGURE 4-12. SECONDARY LOAD TRANSMISSION THROUGH NOSE BOX

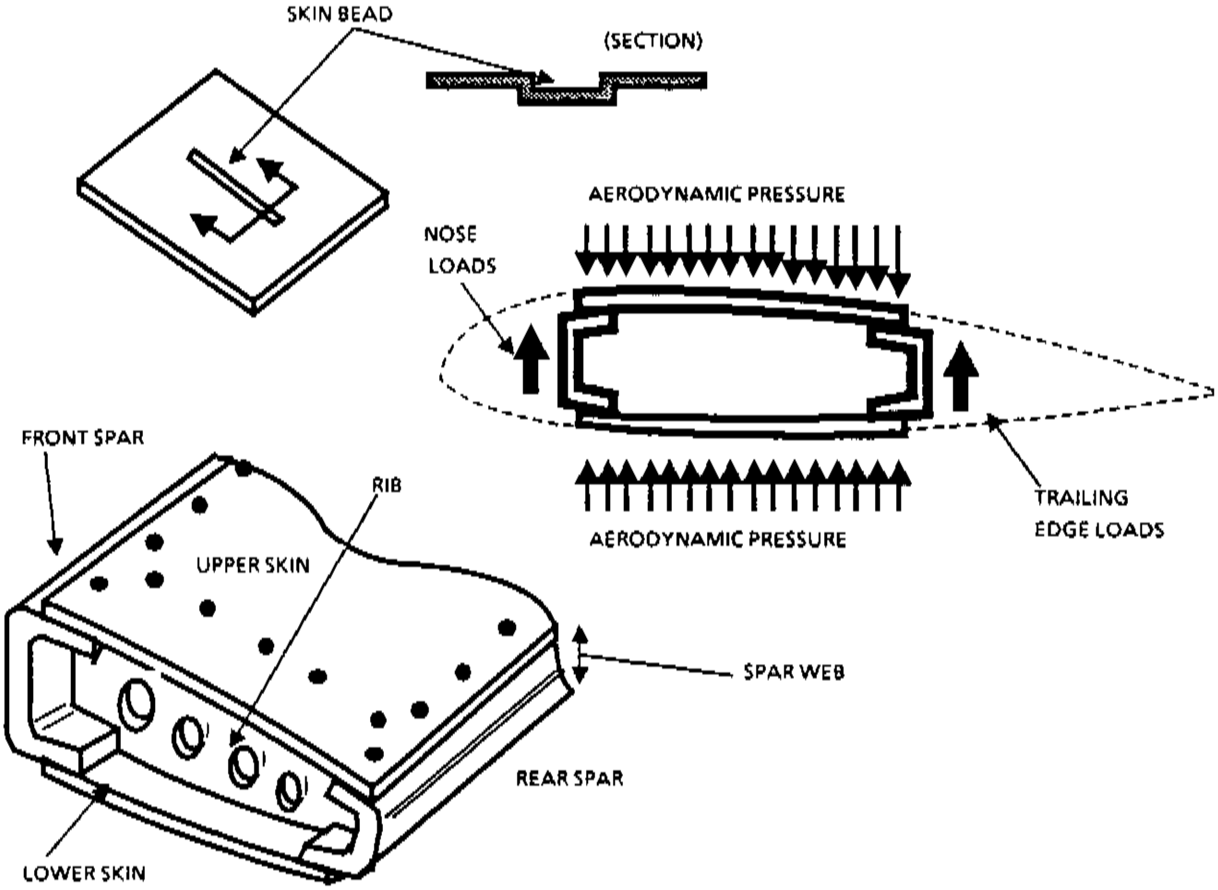


FIGURE 4-13. CONCEPTUAL SKETCHES: PRIMARY LOAD CARRYING STRUCTURE

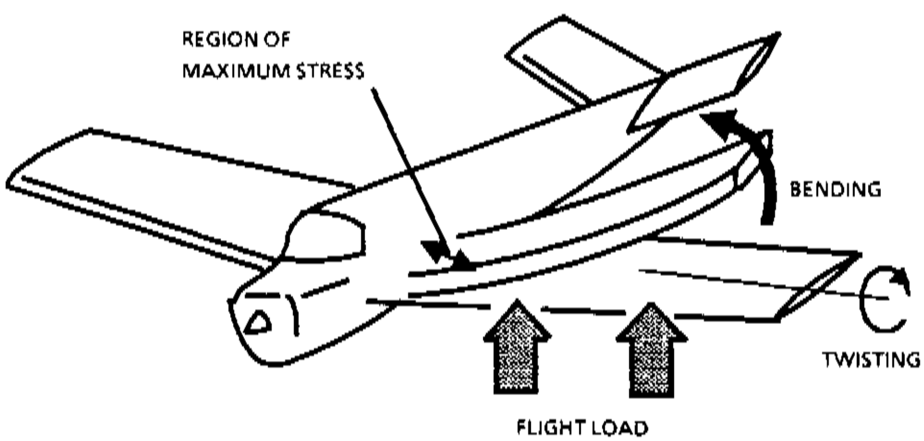


FIGURE 4-14. EXAGGERATED ILLUSTRATION OF STRUCTURAL DEFLECTIONS

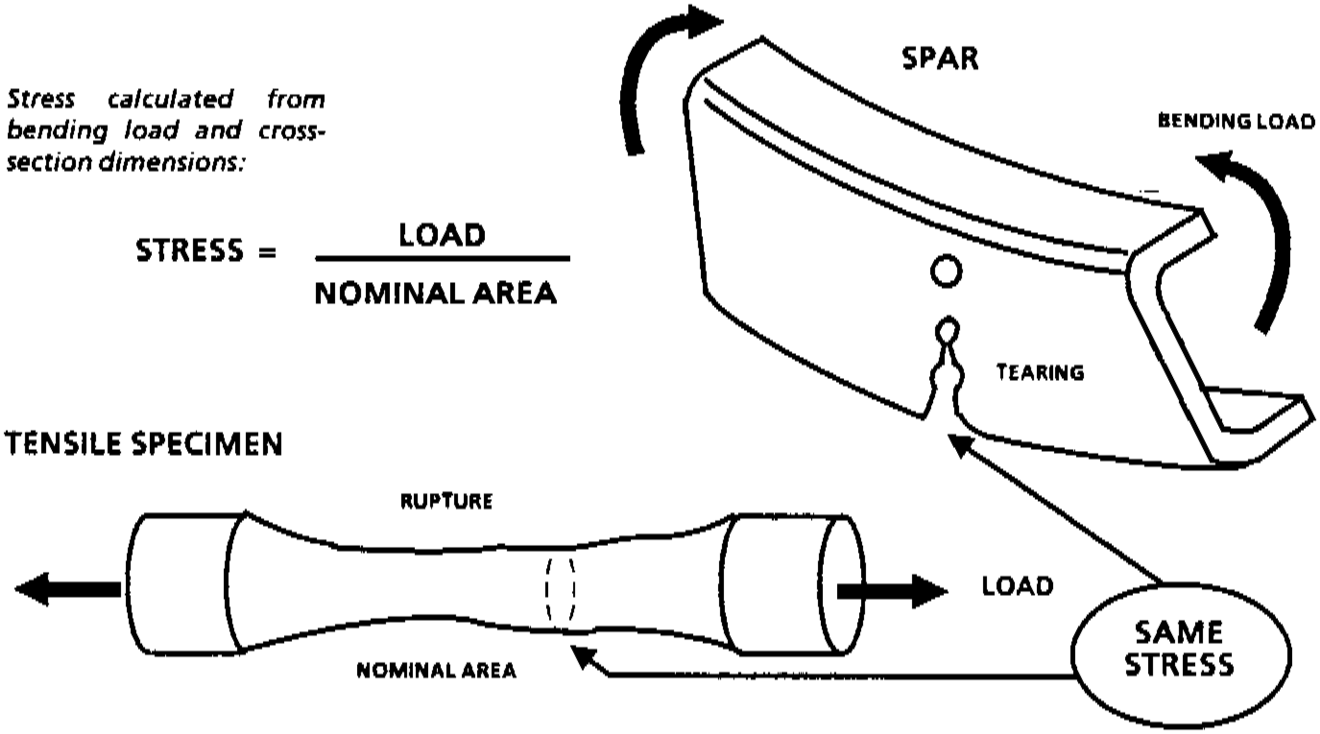


FIGURE 4-15. RELATION OF MATERIAL STRENGTH TO STRUCTURAL STRENGTH

The buckling strength of a skin panel is controlled by the panel's shortest unsupported span, which is usually the spacing between the ribs. Buckling strength decreases as rib spacing increases, up to the point at which the unsupported area is a square. For greater rib spacings, the chordwise separation between the spars determines the buckling strength.

Skin beads (see Figure 4-13) also provide resistance to lateral deflection and can thus substitute for ribs. A skin bead is less effective than a rib for lateral restraint, however, and beads must be spaced more closely than ribs to obtain the same buckling strength.

Twisting of the primary structure (see Figure 4-14) subjects the skins and spar webs to shear. This shear is equivalent to a combination of tension and compression along diagonal directions at ±45° from the spanwise axis (Figure 4-17). If the shear stress is large enough, the panel will buckle diagonally, as shown in the lower part of the figure.

If the upper and lower skins have the same buckling strength, then the upper skin of a wing panel will buckle first because this skin is subjected to combined shear and compression from the twisting and bending flight loads, respectively. Empennage structures behave in a similar manner, except that the lifting load on a stabilizer may be either upward or downward to balance the airplane, depending on the c.g. location and the type of maneuver being executed. Hence, either the upper or lower skin of a stabilizer may buckle first.

Since spars are generally thicker than skins and since their webs and flanges are partly self-restraining, the compression buckling of a spar usually takes the form of a local deformation known as crippling (Figure 4-18a). Spars can also cripple under flight loads because of the compression stress which acts on one flange. After

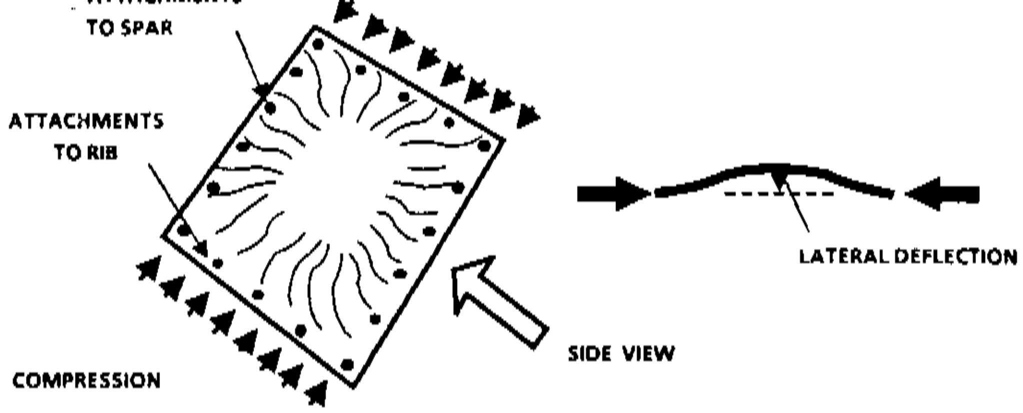


FIGURE 4-16. COMPRESSION BUCKLING OF A SKIN PANEL

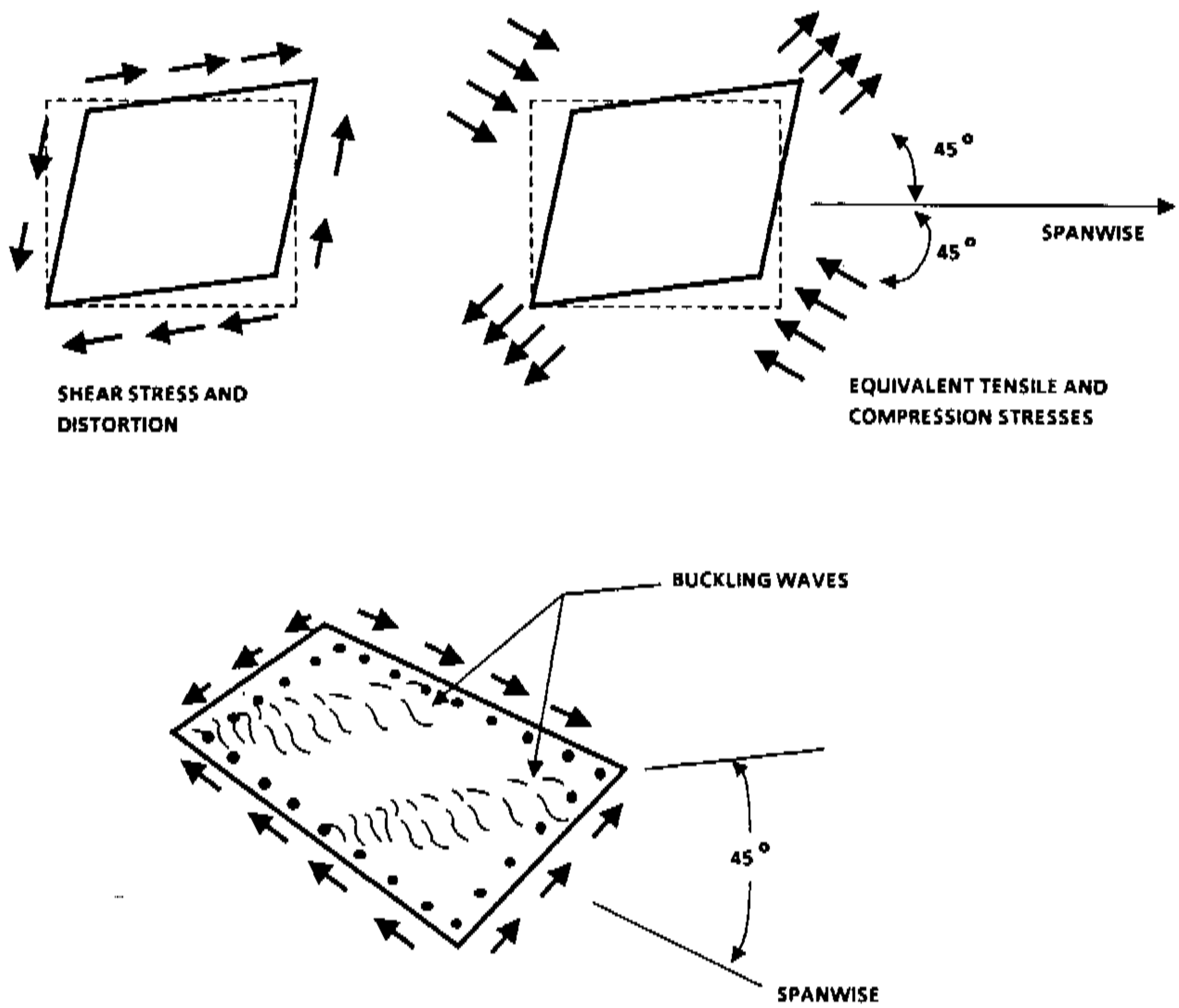
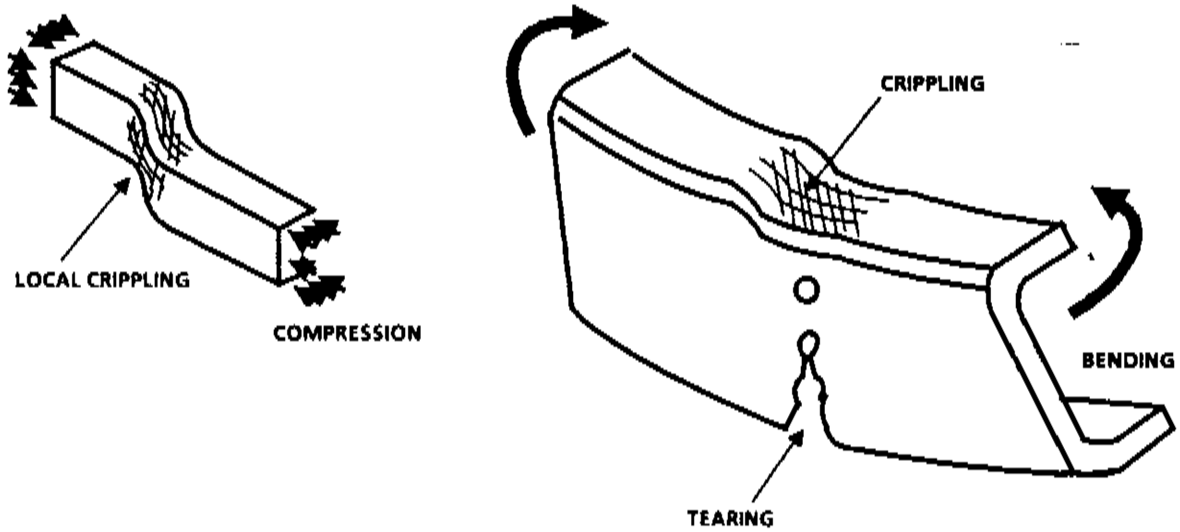


FIGURE 4-17. SHEAR BUCKLING OF A SKIN PANEL

crippling begins, the spar quickly becomes overstressed at the crippled section and starts to tear at or near the tension flange (Figure 4-18b). In such cases it may be difficult to determine by visual inspection alone whether the failure started by crippling at the compression flange or tearing at the tension flange.



(a) BEHAVIOR IN PURE COMPRESSION (b) BEHAVIOR UNDER FLIGHT LOADS

FIGURE 4-18. SPAR CRIPPLING FAILURES

Airframe strength requirements are based on design limit loads (DLL) and design ultimate loads (DUL). The various sources of flight loading from which DLL are calculated were discussed in Section 4.1.

DLL generally represents the most severe loading that a pilot is expected to be able to impose on the airframe via control inputs and (as a separate case) the most severe gust loading expected, within the permissible envelope of airspeed and airplane load factor. The certification requirements state that the structure must not sustain any permanent deformation at loads less than DLL. Above DLL, however, the structure may sustain moderate plastic yielding or buckling. If a buckling deformation is limited to skin deflections such as shown in Figures 4-16 and 4-17, very little if any permanent deformation will result. Either yielding or buckling may cause a few rivets to pop. Permanent deformations (including popped rivets) will not disable the structure, i.e., it should be able to continue sustaining normal flight loads until the airplane can be landed and repaired.

DUL will allow for uncertainties inherent in flying and flight structures such as the following:

- <u>Dynamic amplification</u> - Wings and tails encountering gusts or subject to sudden maneuvers respond dynamically to the applied loads. The structural deflections and stresses will then overshoot the values calculated from the static loads. The amount of overshoot depends upon the airplane speed and structural and aerodynamic damping.

- The flight envelope - Pilots may exceed published speeds or load factors, or the airplane may encounter a stronger gust than specified in the design requirements.

- Production quality - A fleet of nominally identical airplanes inevitably has a range of airframe strengths because production quality varies. A component such as a wing is typically certified on the basis of one strength test at most. There is no guarantee that the production articles will be at least as strong as the test article.

A 1.5 factor of safety is an attempt to account for these uncertainties, i.e., for each loading condition, DUL is prescribed to be 1.5 times DLL. The airframe strength requirements state that the structure must sustain DUL for three seconds without failing.

When a new airplane is being designed, the manufacturer estimates the flight loads for all conditions requiring consideration and performs stress calculations to size the structural components. The most critical known load condition is selected for testing when the new design is to be certified. A test is considered to be better than analysis as evidence of strength, and testing is therefore essential for a new design.

The strength test is performed in a ground-based rig which applies loads to the structure at discrete locations. A large number of load points is used to approximate the flight load distribution, but the test rig cannot completely reproduce the effects of aerodynamic loading.

For example, the upward bending effects of aerodynamic wing loads are commonly simulated by pulling upward on the test wing through "tension patches" which are glued to the upper skin. When tension patches cover a large area of unsupported skin, the buckling strength of the test wing may differ from the buckling strength of a wing in flight. Figure 4-19 illustrates some typical cloth tension patches used by Beech Aircraft Corporation which are designed to minimize this effect. Some other manufacturers use metal tension patches.

Another shortcoming of the ground test is that the applied loads cannot simulate certain aerodynamic or static aeroelastic effects. The test loads are mechanically or hydraulically controlled to increase at a prescribed rate and to maintain a constant direction. Conversely, the flight loads may change in either direction or magnitude when the structure deflects. Some examples are:

- Bending - The aerodynamic pressure distribution which causes the flight bending load continues to act at right angles to the wing or empennage surface, but the corresponding test loads have a fixed direction with respect to the ground. In the test rig, the result is reduced bending stress for a given load. The effect is small as long as the angular deflection of the structure is small.

- Twisting - Twisting of the wing or empennage increases its angle of attack. The increased angle of attack causes a larger lifting load, which in turn can further increase the angle of attack. This is a static aeroelastic condition: static because neither the load nor the structural response are vibratory; aeroelastic because the air load and the elastic deflection are coupled. The equilibrium twist is small and the flight load is close to the test load when the structure is stiff. However, the flight condition may be quite different from the test when the structure is very flexible in twist or when it approaches its ultimate strength. A flight failure of this kind is called static aeroelastic divergence (see Section 5).

- Load redistribution - A wing or empennage structure typically accumulates permanent damage in a sequence of events when the structure exceeds its ultimate strength (1.5 x DLL). Partial (component) failures during the sequence remove some of the internal load transmission paths from the structure. The internal loads are then redistributed and increasingly concentrated on the remnant components. The rapidity of the concentration determines the ultimate failure of the structure. Load concentration may be different in a test rig than in flight because component failures may redistribute the test loads away from the most critical component.

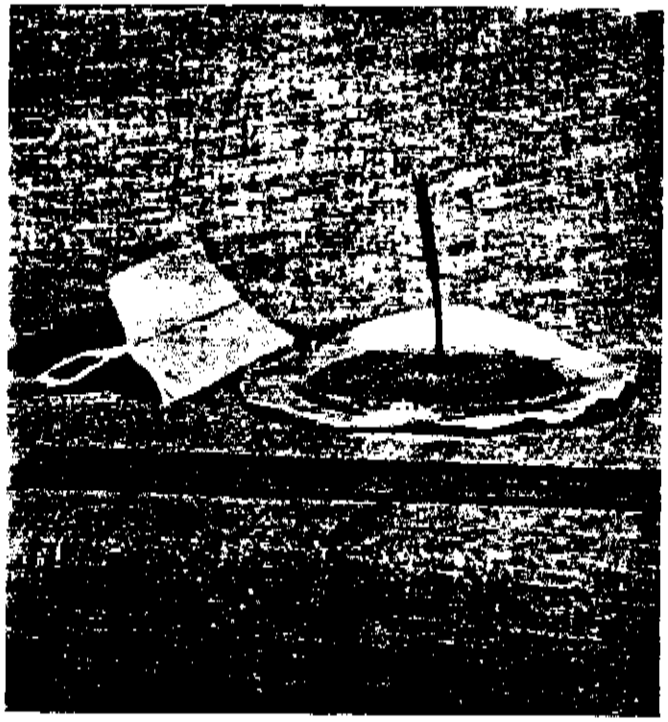


FIGURE 4-19. TENSION PATCHES USED FOR STRENGTH TESTING

Load and stress calculations are acceptable for certification of a design that involves a moderate evolutionary step from an earlier design. The evolution typically consists of growth in gross weight and/or cruising speed, increases in skin thickness and/or spar cross section, and retention of the earlier attachment details. In such cases, stress analysis of the old design can be validated by means of correlation with the earlier strength test, and the validated analysis can then be used to project the strength of the evolved design. A series of such evolutions eventually reduces confidence in the projection to the point where the manufacturer decides to recertify with a test. The Model 35 series is one of many examples of this evolutionary process.

Engineering design of airframes requires test verification to assure that the precise strength requirements for DLL and DUL are satisfied. Since repeated testing is costly and since the manufacturer wants to produce a safe product, conservative biases are generally injected into the design process, and airframe strengths tend to exceed the airworthiness requirements. The excess strength is called margin, e.g., a wing whose strength is 120 percent of DUL is said to have 20 percent margin.

4.3 WING STATIC STRENGTH

This section summarizes the results of the task force assessment of static strength in the Model 35 series wings. The assessment was based mainly on review of Beech Aircraft Corporation documents, with reference to the applicable provisions of CAR 03. The task force made independent calculations, however, on the key issue of spanwise lift distribution.

The discussion begins with a review of the original design load estimates (Section 4.3.1). Section 4.3.2 discusses the effect of the fuselage on aerodynamic load distribution on the wings. The load estimates for later models in the series are reviewed in Section 4.3.3. The results of wing strength tests are summarized in Section 4.3.4.

4.3.1 Analysis of Load Estimates for Original Model 35 Wing

Paragraphs 03.2143 through 03.31 in CAR 03 describe the design requirements for the Normal Category of general aviation aircraft. Beech established the design loads for the original 35 based on a 2500-lb. gross weight. The most critical load for the wings is for the symmetrical condition at a positive limit load factor of 3.8 g, which gives:

$$F_Z = 9335 \text{ lb.}; \quad D_X = 1553 \text{ lb.} \qquad (4\text{-}3)$$

at right angles and parallel, respectively, to the waterline of the plane[22]. These loads are the design limit loads.

The above loads consist of contributions from the wing and fuselage. It is only that portion of the loads on the wing that dictates the wing strength requirement. In the calculations of wing characteristics and air load distributions for structural purposes[23], the basic wing planform is shown in Figure 4-20 in which the shaded region is the area occupied by the fuselage. The total area of the wing planform is 177.6 sq. ft., about 16 percent of which is buried in the fuselage. The basic spanwise lift distributions for flaps neutral and flaps deflected are determined by the methods of References 24 and 25. Reference 23 gives the detailed distributions which are used for determining the loads for the wing and fuselage and Reference 26 gives the results of the corresponding strength tests. Beech also evaluated the total lift[22] for various flight conditions based on their own wind tunnel test data on the measured lift coefficient[27]. The total aerodynamic loads at the design limit estimated by the two approaches agreed well with each other[26]. However, there are no measured data to substantiate the assumed spanwise lift distribution.

To determine the spanwise lift distribution, Beech implicitly assumed that the presence of the fuselage did not influence the aerodynamics on the wing planform and that the fuselage contributed the same amount of lift as the portion of the wing planform it replaced. Since the fuselage masks about 16 percent of the total wing

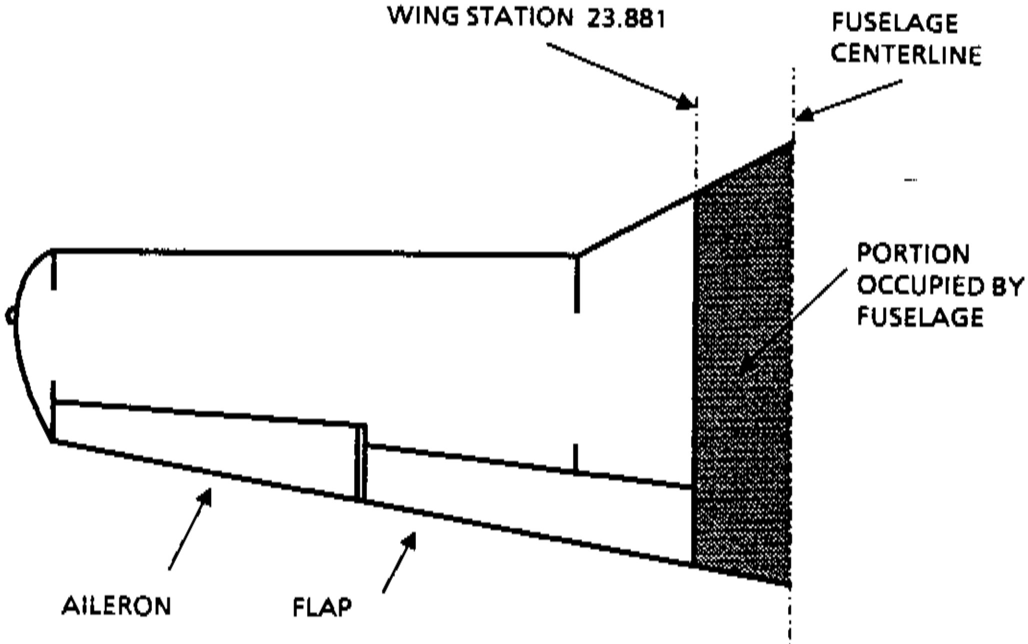


FIGURE 4-20. MODEL 35 WING PLANFORM

planform area, the total load[28] at the wing-fuselage attachments amounts to about 84 percent of the total lift, i.e., for one side:

$$F_Z' = (9335 / 2) \times .84 = 4663 \times .84 = 3916 \text{ lb.} \quad (4\text{-}4)$$

$$D_X' = (1553 / 2) \times .84 = 774 \times .84 = 650 \text{ lb.} \quad (4\text{-}5)$$

The wing weighs 124 lb. (not including the landing gear, 44.6 lb., and fuel and fuel tank, 129.8 lb.). After the inertia load of the wing is subtracted, the net design limit load in the vertical direction is:

$$(F_Z)_{net} = 3916 - 124 \times 3.8 = 3444 \text{ lb.} \quad (4\text{-}6)$$

The ultimate force resultant[33] is:

$$F_R = 3444 \times 1.5 \times 1.013 = 5238 \text{ lb.} \quad (4\text{-}7)$$

where the factor of 1.013 accounts for D_X from Equation 4-3. The inertia loads of the main landing gear, fuel, and fuel tank act in the direction opposite to the normal air load. The values of these inertia loads are 254 lb. (44.6 x 3.8 x 1.5) and 740 lb. (129.8 x 3.8 x 1.5), respectively, at the ultimate condition. When they are subtracted from the air load, the actual ultimate load at the wing root for the Model 35 is:

$$F_U = 4244 \text{ lb.} \quad (4\text{-}8)$$

4.3.2 Fuselage Lift

The presence of the fuselage can affect the flow around the wing near the wing root and change its spanwise lift distribution (see Section 4.1). The amount of lift that is generated by the fuselage depends on the fuselage shape and the effect of wing-body interference, which itself is a function of angle of attack and the details at the wing-body junction. Determining the fuselage contribution to the total lift is not a simple matter. The analytical methods used to calculate fuselage lift were not completely developed until the late 1960's to early 1970's, almost 30 years after the Model 35 was designed.

To assume, as Beech did, that the fuselage lift is the same as the lift generated by the hidden portion of the wing planform is the simplest approximation to determine the air loads on the wing. According to this approximation, only 84 percent of the total lift is carried by the wings. However, there is neither wind tunnel nor flight test data to substantiate whether the 84 percent factor is too high or too low for the wing-body configuration of the Model 35. Beech has recently reevaluated the fuselage lift contribution using a computer program based on modern aerodynamics methods. Depending on the angle of attack, the fuselage contributes from 12 to 14.5 percent of the total lift[29]. In review of other literature, the task force noted that for a long thin fuselage of circular or elliptical cross section (not quite the same as that of the Model 35 fuselage) and body diameter to wing span ratio of about 0.12 (about the same as that of the Model 35), the fuselage lift is less than 10 percent of the total lift[14]. In the light of the difference in fuselage cross section, the recent Beech estimate of 12 to 14.5 percent appears reasonable.

As reported in Reference 64, Beech tested the wing alone and the wing-fuselage of an earlier airplane with a wing-fuselage similar to the Model 35. The results (Figure 4-21) indicate that, for a given angle of attack, the wing alone has a higher lift coefficient than the wing-fuselage combination. This implies that the presence of the fuselage induces spanwise lift redistribution, as discussed in Section 4.1. Wind tunnel or flight tests are needed to determine whether the redistribution is inboard or outboard.

To demonstrate the possible effect on wing load, the task force made the conservative assumption that the difference in measured lift coefficients reflects outboard redistribution. This method is comparable to that used by Beech in SAR 49-700 (Reference 30, page 17) in which wing-fuselage combination effects were considered for Model H35. For a gross weight of 2550 lb.*, the total lift on the wing and fuselage is:

$$F_Z = 2550 \times 3.8 - 34 = 9656 \text{ lb.} \qquad (4\text{-}9)$$

where the 34 lb. figure subtracted in Equation 4-9 is the balancing tail load.** At V_D=250 mph (dynamic pressure, q=159.9 lb./sq.ft.) and S=177.6 sq.ft., the lift coefficient required for the wing and fuselage is:

$$C_{L_{W+F}} = 9656 \,/(159.9 \times 177.6) = 0.34 \qquad (4\text{-}10)$$

*The original Model 35 was actually certified to 2550 lb. maximum gross weight, not the 2500 lb. weight used in the Beech load estimates discussed earlier.
**For the combination of maximum gross weight (aft c.g.) and angle of attack considered in this example, the balancing tail load is upward and reduces the lift required from the wing.

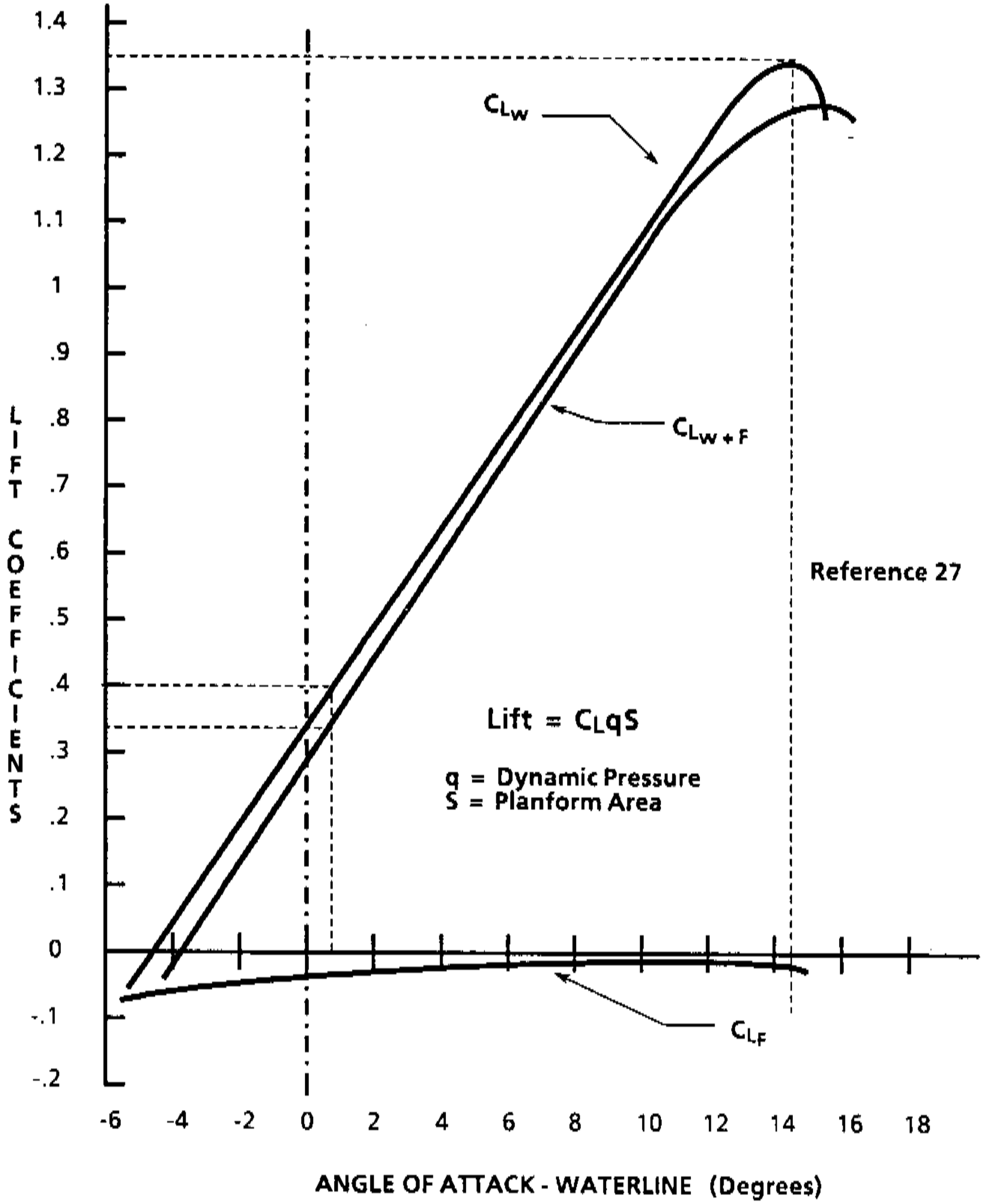


FIGURE 4–21. LIFT CURVES FOR WING ALONE AND WING–FUSELAGE COMBINATION

From Figure 4-21, the angle of attack corresponding to this lift coefficient is found to be 0.8 degrees using the combined lift coefficient, $C_{L_{W+F}}$, for the fuselage and exposed wing. In order to estimate the portion of lift provided by the exposed wing area, the lift coefficient based on wing lift only, C_{L_W}, is used with the reference area reduced by the portion of the wing area that is contained by the fuselage. The lift coefficient for the exposed wing area is $C_L = 0.4$ at the same angle of attack. The postulated outboard redistribution effect is approximated by assuming that the exposed 84 percent of the wing planform operates at $C_L = 0.4$, i.e., for one side:

$$F_Z' = (0.4 \times 159.9 \times 177.6 / 2) \times 0.84 = 4771 \text{ lb.} \qquad (4\text{-}11)$$

The fuselage carries less than 2 percent of the total lift. After the inertia load of the wing is subtracted, the ultimate force resultant is found to be (neglecting drag):

$$F_R = (4771 - 124 \times 3.8) \times 1.5 = 4300 \times 1.5 = 6450 \text{ lb.} \qquad (4\text{-}12)$$

This value is slightly more than 123 percent of the force estimated by Beech as given in Equation 4-7. This approach to estimating wing load leads to the conclusion that the fuselage carries only two percent of the total lift for the subject flight condition as opposed to 16 percent assumed by Beech.

Since the inertia loads of the main landing gear, fuel, and fuel tank oppose the lift, the ultimate load at the wing root is:

$$F_U = 5456 \text{ lb.} \qquad (4\text{-}13)$$

in contrast to the Beech estimate of 4,244 lb. (Equation 4-8).

The estimate above is believed to be overly conservative. Using the same procedure to calculate the wing load at 160 mph, the cruising speed, we obtain:

$$F_R = 5547 \text{ lb.} \qquad (4\text{-}14)$$

($C_{L_{W+F}} = 0.83$, $C_L = 0.854$) which is only about six percent higher than the force estimated by Beech. In this case the fuselage carries about 14 percent of the total lift

4.3.3 Load Estimates for Wings of Later Models

After the original Model 35, all the Bonanza models were certified in the Utility Category with a positive limit load factor of 4.4 g. The method for estimating the wing loads of the A35[31] and C35[32] was similar. From Models A through M the gross weight increased in stages from 2650 lb. for the A35 to 2950 lb. for the M35, and the cruising speed increased from 160 mph to 180 mph. In References 33 through 39, Beech did not give detailed load analyses of the later models. The adequacy of these later models was simply justified on the grounds that the extra margins were demonstrated by the test on an A35 wing which failed at 130 percent of design ultimate load[40] and on an F35 wing which failed at 135 percent of the A35 design ultimate load[41]. In the meantime the structural improvements made to the newer models to increase their strength should be more than adequate to take care of the moderate increases in the gross weight and operating speed. However, prediction of the design ultimate load of the Model A35 wing is subject to the same uncertainty as the prediction of the design ultimate load on the original Model 35 (Section

4.3.2). More conservative assumptions, as discussed in Section 4.3.2 will result in a similar reduction in the extra structural margins.

The issue of lift contribution by the fuselage was addressed to some extent by Beech. In Reference 30, Beech first calculated the air loads for the Model H35 using the same method described in Reference 43 and then reduced the air load over the fuselage to 25 percent of the nominal planform contribution and redistributed the remainder outboard to the wings. As a result, the net ultimate load and bending moment at the wing root change from 7298 to 8532 lb. and from 546,000 to 636,500 in.-lb., respectively, which represents an increase of about 17 percent. This assumption is clearly very conservative and there is no apparent reason for the drastic change in the distribution. It appears that because of the available structural margin, someone at Beech decided to use more conservative assumptions.

Beginning with the Model N35[42], Beech seems to have taken the reduction in the lift coefficient by the fuselage into account by using a different lift coefficient for the wing/fuselage combination as compared to that of the wing alone. However, there is still uncertainty concerning the actual spanwise lift distribution on the wing. The redistribution of the air load outboard to the wing was not performed for the models other than the Model H35. Table 4-1 summarizes the design ultimate net loads and bending moments calculated by Beech.

TABLE 4-1. DESIGN ULTIMATE WING LOADS

Model	Net Shear (lbs)	Net Moment (in-lbs)	Reference
35	5238	389,200	SAR 49-904, p.4
A35	6440	482,200	SAR 49-904, p.4
H35	8532*	636,500*	SAR 49-700, p.17
N35	7785	604,650	SAR 49-20, p.128
S35	8827	680,600	SAR 49-22, p.121

*the fuselage air load was reduced to 25% of that calculated by the method of SAR 35-5 and the remainder redistributed outboard to the wings

4.3.4 Wing Strength Tests

Bonanza wings were subjected to full-scale static strength tests simulating critical load conditions to satisfy the requirements for type certification. The following general description of the test procedure is abstracted from the task force review of Beech engineering reports.

The wing was mounted on a jig, or if the wing and fuselage were tested together, then the fuselage would be fixed to a jig with the wing attached to the fuselage. Hydraulic jacks were used to load most items such as seats, front and rear parts of the fuselage, and main landing gear, while sand bags and lead bars were used to simulate fuel loads. A jig loaded with weights was used to simulate the engine. The aerodynamic loading on the wing was applied by a hydraulic jack and load ring dynamometer through a whiffletree system to simulate the air load distribution. The traces of the whiffletree were fastened to the wing surface by means of canvas tension patches (Figure 4-19) cemented to the wing upper skin.

Test records included structural deflections, stresses, and observations of buckling and structural deformation. Deflection measurements were made at a number of locations along the fuselage and spanwise along the wing using deflection scales and a transit for reading the scales. Stress measurements were made using bonded electrical resistance strain gages.

Loading was performed in a scheduled sequence of load increments, with measurements made at each step. For a wing and fuselage combination, the main landing gear, fuel tanks, and front seats were loaded first, the wing lift was applied next, and the front and rear parts of the fuselage were loaded last[26]. The load increments for one test were 40, 66.7, 80, 90, 95, and 100 percent of design ultimate load. These increments sometimes varied for different tests, but measurements were generally made at limit load (66.7 percent of ultimate) and at the ultimate load. The loading was relieved after reaching selected load increments such as the limit load point. At these load-relief points, deflection readings were taken to measure permanent set, and any changes in the appearance of the structure were noted. The loads were then reapplied, and the incremental loading continued.

A test in which the wing is brought to design ultimate load and then relieved without structural failure is all that is required for certification, but the test result gives no information about the wing's margin. Most of the Model 35 wing strength tests were in this category. Four wings were loaded to failure, however, and these tests do provide information about margin.

There were two such tests of the original Model 35. In the first test, the wing failed at 6498 lb. of applied load and 478,700 in.-lb. of bending moment, 123 percent of ultimate[43]. The failure was confined between Wing Station (WS) 108 inboard to the landing light at WS 73. In the second test the wing failed at 5811 lb. load and 428,100 in.-lb. moment (110 percent of ultimate)[44]. The failure was in the main spar at WS 100 and in the leading edge inboard of WS 100. However, in the second case, the wing had been subjected to a negative gust in flight, resulting in buckling of the skins and cracking in an angle near the rear spar lower flange at WS 66. This pre-test damage apparently reduced the wing strength in the test.

Several full-scale tests of the Model A35 were conducted prior to its final design configuration. The final test of the A35 wing was on a jig and the wing was tested to destruction[45]. The ultimate strength was 8370 lb. load and 626,900 in.-lb. moment (130 percent of ultimate for the Model A35). The upper main spar flange buckled between WS 84 and WS 94. The buckles extended aft to the rear spar at WS 66, and forward and inboard across the diagonal nose rib.

The F35 wing was also tested on a jig to failure[41]. At 8694 lb. load and 651,000 in.-lb. moment (135 percent of ultimate for the A35), the nose upper skin forward of the front spar between WS 59 and WS 66, and the removable door panel at the fuel filler cap were buckled. Table 4-2 summarizes the results of these tests.

No tests have been conducted on the later model wings, although the design ultimate load for the later models increased due to increases in gross weight. Beech justified the adequacy of the later wings on the ground that the design ultimate loads of models prior to the S35 are all less than the strength of the F35 wing, while the later wings are stronger because of structural improvements. For the Model S35 and later, even though the design ultimates are slightly higher than the maximum loads tested (8827 versus 8694 lb. load and 680,600 versus 651,000 in.-lb. moment), the many structural improvements in the later models were judged to be more than enough to withstand the slightly higher design ultimate loading.

TABLE 4-2. RESULTS OF TESTS ESTABLISHING WING MARGINS

Model	Net Shear (lbs)	Net Moment (in-lbs)	% Design Ultimate	Reference
35	6498	478700	123	SAR 35-986
35	5811*	428100*	110	SAR 35-988
A35	8370	626900	130	SAR 49-936
F35	8694	651000	130**	SAR 49-953

* Prior buckling in skins and crack in an angle in the rear spar.
** 135% ultimate of A35 was tested which is equivalent to about 130% ultimate of F35.

Beech's justification is valid only if the estimated design ultimate loads are correct or on the conservative side. As demonstrated earlier, depending on the actual spanwise lift distribution, the wing may or may not have adequate strength to withstand the design ultimate load. For example, the ultimate load estimated by the task force in Section 4.3.2 is 6450 lb., which is almost the same as the tested strength of the original Model 35. If the conservative method of Reference 30 were used to calculate the design ultimate load for the original Model 35, one would obtain 6240 lb. load and 463,900 in.-lb. moment, which are almost the same as the strength established by the applicable test. In other words, the original Model 35 wing would show only a slight margin above the design ultimate load required for a 3.8g limit load factor. Similarly, if the design ultimates for the Model S35 and later were calculated employing the methods of Reference 30, much higher ultimates would result. The large variations of design load as a function of estimation method suggest that a better understanding of spanwise lift distribution is needed for a definitive assessment of wing margin.

4.3.5 Summary

Beech has demonstrated that the strength of the wings of the Models 35, A35, and F35 exceed the design ultimate loads as required by the airworthiness regulations. For the wings prior to the Model S35, the design ultimates are less than the strength of the Model F35, while the wings are stronger than or at least as strong as the Model F35 wing. Therefore, their strengths meet or exceed the requirements. For Models S35 and later, the design ultimates are only slightly higher than the strength of the Model F35, but the wings of these later models are much stronger because of their many structural improvements. Therefore, their strengths should meet or exceed the requirements. The foregoing conclusion is only true if Beech's