in the subsequent investigation of the drag-truss system, due attention should be given to all the force components which will be applied to the attachment points by the lift truss.

3.1121. *Lift struts.* Consider the strut-braced monoplane wing shown in figure 3–2. The spars in the figure are shown perpendicular to the basic wing chord (the reference line for normal and chord loads is the M. A. C. of the wing). If the spars are not perpendicular to the chord reference line, the resultant of the chord and normal loads should be resolved into components parallel and normal to the spar, as shown in figure 3–3a. Also, in the general case, the drag truss will not be perpendicular to the spar face. This angularity should be considered (fig. 3–3b), unless it is of small order, which would result in a negligible correction.

The vertical reactions on the front and rear spars from the lift struts may be determined by taking moments about point C (fig. 3–2) of all the external loads on the spars (sec. 3.114). Then $R_f=\frac{M_{cf}}{g}$; and $R_r=\frac{M_{cr}}{g}$, where M_{cf} and M_{cr} are the moments about the spar-root attachment, point C, of the front and rear spars, respectively.

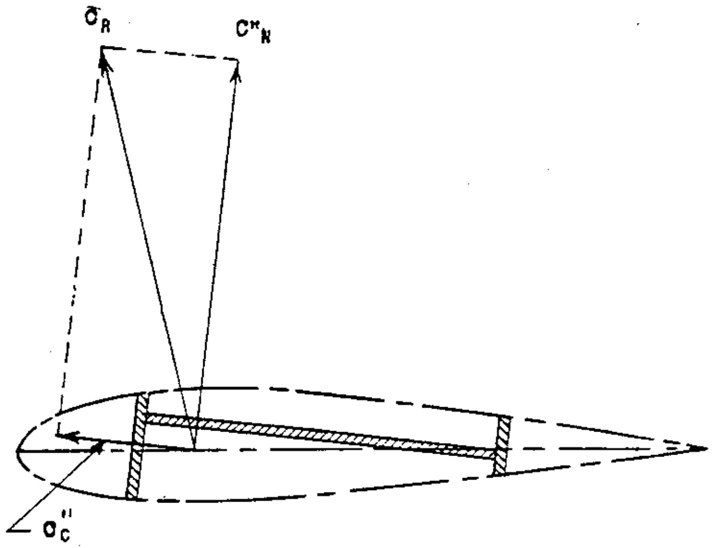
The strut and spar axial loads may be determined by graphical or analytical methods on the basis of the truss A B C, if the fitting is eccentric to the neutral axis of the spar. If the graphical method is used, the correction for angularity of the strut to the V–H plane should not be overlooked.

The strut loads also can be determined by the following formula, which includes the correction for angularity:

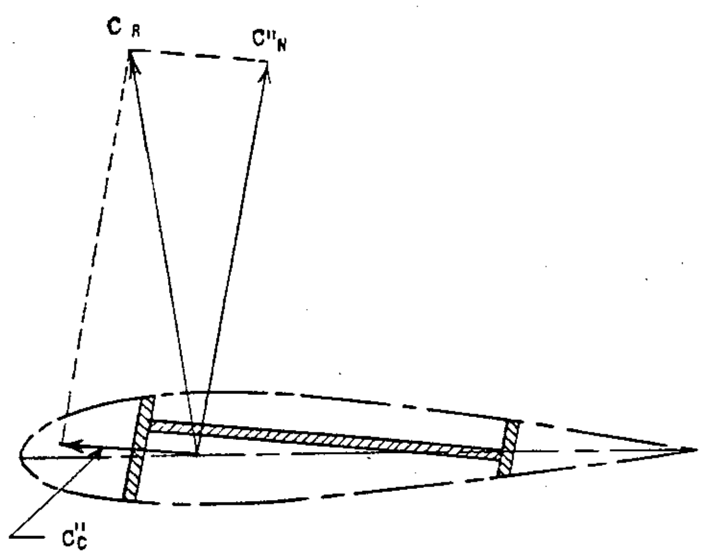
$$\textit{Strut load}=\frac{M}{h}\times\frac{\textit{true length}}{\textit{projected length in V-H plane}} \tag{3:5}$$

After the loads in the struts have been determined, the axial load in each spar is: (strut load)$\times\left(\frac{H}{L}\right)$ and the chord component acting on the wing from each strut is: (strut load)$\times\left(\frac{D}{L}\right)$.

When an eccentricity, e, in the root fitting exists, the chord loads and reactions will act in a plane which generally is not parallel to the line AC. The effect of the eccentricity is to modify the vertical reactions at the strut point and root.



(a) DRAG TRUSS PERPENDICULAR TO SPAR FACE



(b) DRAG TRUSS NOT PERPENDICULAR TO SPAR FACE

Figure 3–3. Resolution of forces into components acting on spars and drag truss.

The increment of reaction to be added or subtracted is: $\Delta R=\frac{R_h e}{g}$ (fig. 3–4d). Then, the total vertical reaction component at the strut point is $R+\Delta R$. It is, at once, apparent that the value of the drag-truss reaction, R_h, is a function of the strut load (fig. 3–4c); therefore, if extreme accuracy is desired, it becomes necessary to solve for the reactions on the lift and drag truss by means of simultaneous equations which include expressions for all the unknowns involved. The reactions

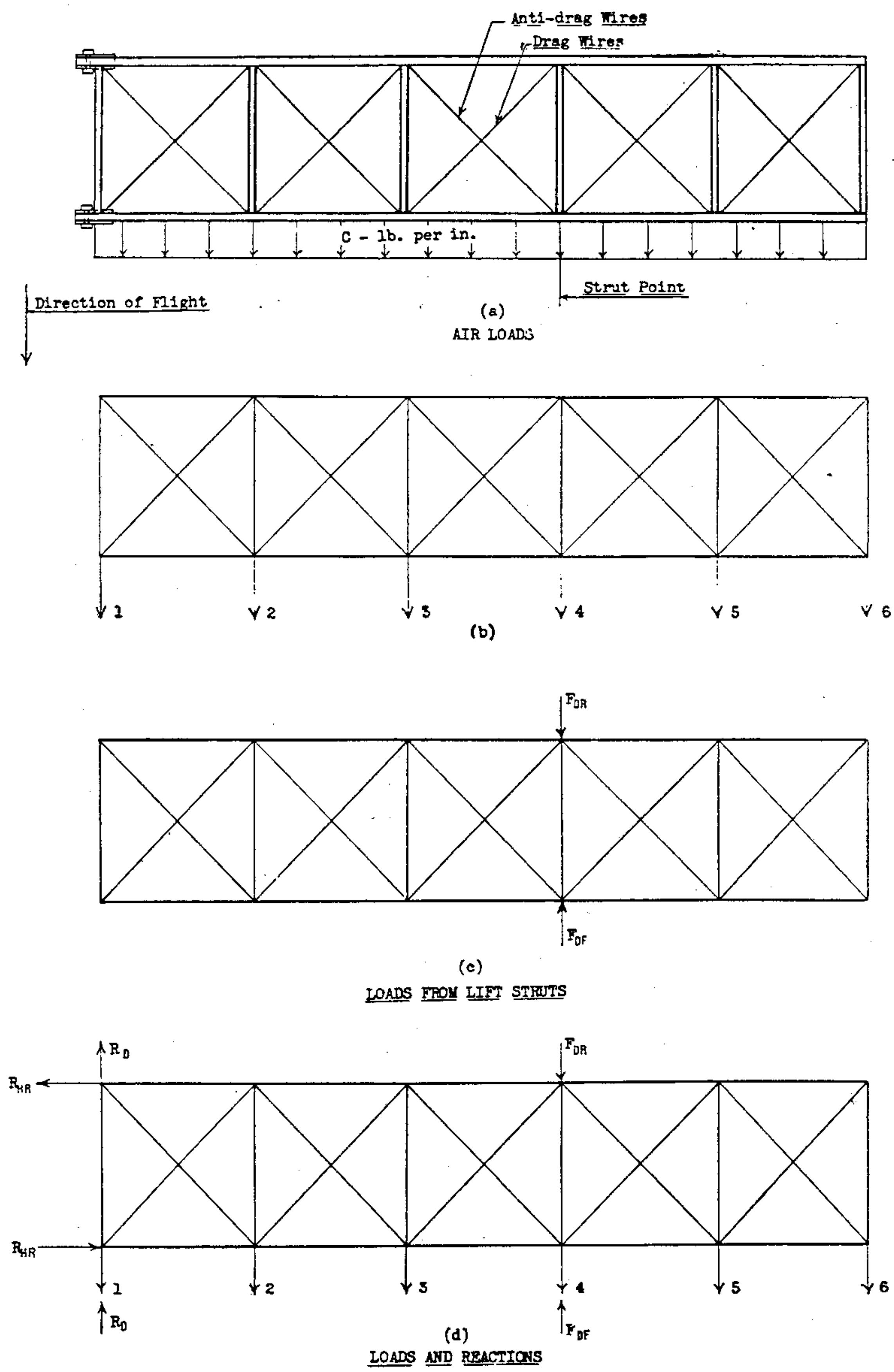


Figure 3–4. Drag truss.

may also be determined by trial and error with comparable results if sufficient trials are made. However, unless the value of $\triangle R$ is in excess of 2 percent of R, it is considered satisfactory to assume that the total reaction is $R+\triangle R$.

3.1122. *Jury struts.* In computing the compressive strength of lift struts which are braced by a jury strut attached to the wing, it is usually satisfactory to assume that a pin-ended joint exists in the lift strut at the point of attachment of the jury strut. The jury strut itself should be investigated for loads imposed by the deflection of the main wing structure. An approximate solution based on relative deflections is satisfactory, if the jury strut is conservatively designed to withstand vibration of the lift strut. When the jury strut is considered as a point of support in the wing-spar analysis, rational analysis of the entire structure should be made (ref. 3–17).

3.1123. *Nonparallel wires.* When two or more wires are attached to a common point on the wing, but are not parallel, the distribution of load between the wires may be determined by least work or equivalent methods. The following approximate equations may be used for determining the load distribution between wires, provided the loads so obtained are increased 25 percent.

$$P_1=\left[\frac{V_1A_1L_1L_2^3}{V_1^2A_1L_2^3+V_2^2A_2L_1^3}\right]B \qquad (3:6)$$

$$P_2=\left[\frac{V_2A_2L_1^3L_2}{V_1^2A_1L_2^3+V_2^2A_2L_1^3}\right]B \qquad (3:7)$$

where:

B=beam component of load to be carried at the joint.

P_1=load in wire 1.

P_2=load in wire 2.

V_1=vertical length component of wire 1.

V_2=vertical length component of wire 2.

A_1 and A_2 represent the areas of the respective wires.

L_1 and L_2 represent the lengths of the respective wires.

The chord components of the air loads and the unbalanced chord components of the loads in interplane struts and lift wires at their point of attachment to the wing should then be assumed to be carried entirely by the internal drag truss.

3.1124. *Biplane lift trusses.* In biplanes that have two complete lift-truss and drag-truss systems intereonnected by an N strut, a twisting moment applied to the wing cellule will be resisted in an indeterminate manner, as each pair of trusses can supply a resisting couple. An exact solution involving the method of least work, or a similar method, can be used to determine the load distribution (ref. 3–16). For simplicity, however, it may be assumed first that all the external normal loads and torsional forces about the aerodynamic center of the cellule are resisted by the lift trusses. This assumption is usually conservative for the lift trusses, but does not adequately cover the possible loading conditions for the drag trusses. A second condition should therefore be investigated by assuming that a relatively large portion (approximately 75 percent) of the torsional forces about the aerodynamic center of the cellule are resisted by the drag trusses. In the case of a single-lift-truss biplane, the drag trusses must, of course, resist the entire moment of the air forces with respect to the plane of the lift truss.

3.1125. *Rigging loads.* Wire-braced structures should be designed for the rigging loads specified by the procuring or certificating agency. Sometimes it may be necessary to combine the rigging loads with internal loads from flight or landing conditions.

The effects of initial rigging loads on the final internal loads are difficult to predict, but, in certain cases, may be serious enough to warrant some investigation. In this connection, methods based on least work or deflection theory offer the only exact solution. Approximate methods, however, are satisfactory if based on rational assumptions. As an example, if a certain counter-wire will not become slack before the ultimate load is reached, the analysis can be conducted by assuming that the wire is replaced by a force acting in addition to the external air forces. The residual load from the counterwire can be assumed to be a certain percentage of the rated load and will, of course, be less than the initial rigging load.

3.113. *Drag-truss analysis.*

3.1130. *Single drag-truss systems.* Single drag-truss systems are employed in strut- or wire-braced wings where the ratio of the span of the overhang to the mean chord is not excessive. The requirements of the specific agency involved should be reviewed in regard to the upper limit on this value above which double-drag bracing is required.

An example of a conventional drag truss is shown in figure 3–4 for a strut-braced monoplane wing. The chord loading, C, in pounds per inch run (fig. 3–4 (a)) may be distributed to the panel points of the truss (*b*) as concentrated loads 1, 2, 3, 4, etc. In addition to the chord loads due to air load, the lift struts also apply loads in the chord plane. In section 3.1121, the method of determining the chord components was given. These components are shown in figure 3–4 (*c*), assuming that the wing is so loaded that the lift struts are subjected to tensile loads. If items of concentrated weight, such as fuel tanks and landing gear, were not accounted for when the running chord load was computed in table 3–2, the resultant inertia loads from these items of weight should be applied to the drag truss. In figure 3–4 (*d*) are shown all the loads and reactions acting on the drag truss.

The loads in the drag-truss members may now be determined by graphical or analytical methods. Exact division of the drag reaction, R_D, on the truss is generally indeterminate, insofar as the front and rear root-spar attachments are concerned. In general, overlapping assumptions should be made, or the drag reaction conservatively assumed to be resisted entirely by one root fitting. Occasionally, the drag reaction may be divided equally between the front and rear root-spar fittings if they have approximately the same rigidity in the drag direction.

3.1131. *Double drag-truss systems.* A double drag truss is employed in cantilever wings or braced wings where it is necessary to provide additional torsional rigidity outboard of the strut point. The investigation of double-drag trusses follows the same line of procedure outlined in section 3.1130. The design of the double truss is usually dictated by torsional rigidity requirements rather than by the actual design loads applied to the structure.

In showing compliance with requirements in which the upper drag wire in one bay and the lower drag wire in the adjacent bay are assumed in action (the remaining wires in these two bays assumed to be out of action), the loads on the strut take the form shown in figure 3–5. R_{wu} and R_{wl} represent the wire force components along the drag strut. In general, it will be necessary to balance these components in the drag direction by a reaction, $R_{wl}-R_{wu}$; then, taking moments about a convenient point, the vertical couple force R_c may be determined. Having the forces and reactions on the drag strut, the internal forces readily may be determined.

3.1132. *Fixity of drag struts.* Drag struts should be assumed to have an end-fixity coefficient of 1.0, except in cases of unusually rigid restraint, in which a coefficient of 1.5 may be used.

3.1133. *Plywood drag-truss systems.* In a two-spar, plywood-covered wing, the plywood covering, together with the drag struts, is usually depended upon to carry the chord shear. Section 3.12 gives methods of analysis of this type of structure.

3.114. *Spar shears and moments.* The fundamental principles of statics should be employed in the determination of wing-spar shears and bending moments. Before proceeding with the detailed determination of these items, it is essential, in order to avoid errors, that all the external loads and reactions be determined for the spar.

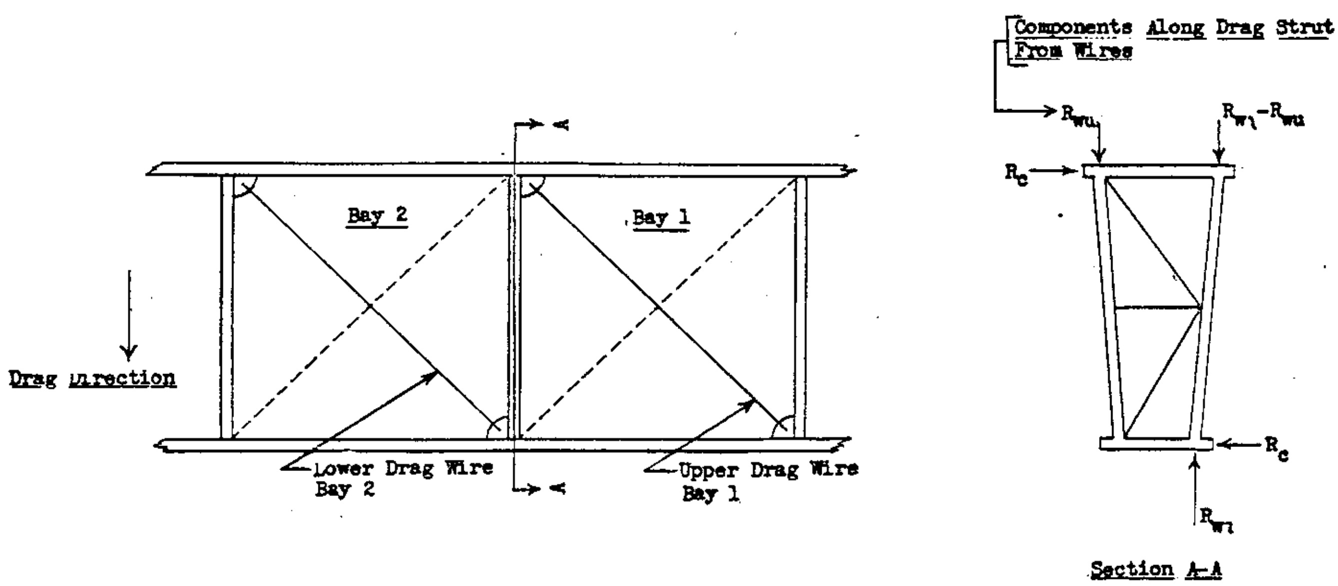
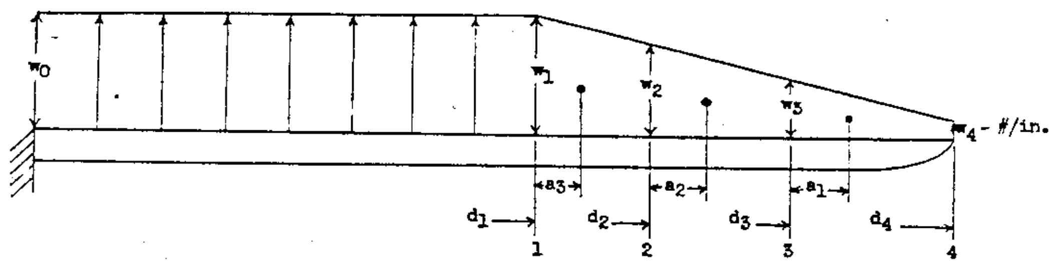


Figure 3–5. Double drag truss—two drag wires in action.



1	2	3	4	5	6	7	8	9	10	11
Section	Distance from Root	Distance between Sections d	Load per in. w	Average load per in. w_a	Load between Sections F	Arm to centroid (1)* a	Moment $M' = Fa$	Shear $S = \Sigma F$	Moment $M'' = Sd$	Moment at Section
4	d_4		w_4					0		0
		d_4-d_3		$\frac{w_4+w_3}{2}$	Item 3 x Item 5	(1)* a_1	(2)* F_{4-3} x a_1		0	
3	d_3		w_3					F_{4-3}		Items 8 + 10 + 11
		d_3-d_2		$\frac{w_3+w_2}{2}$	Item 3 x Item 5	a_2	F_{3-2} x a_2		(3)* S_3 x (d_3-d_2)	
2	d_2		w_2					$F_{4-3}+$ F_{3-2}		Σ
		d_2-d_1		$\frac{w_2+w_1}{2}$	Item 3 x Item 5	a_3	F_{2-1} x a_3		S_2 x (d_2-d_1)	
1	d_1		w_1					F_{4-3} + F_{3-2} + F_{2-1}		Σ
0	0		w_0							

*NOTES

(1) The center of gravity of a trapezoidal loading may be determined by the formula $\frac{x}{e} = \frac{2+R}{3(1+R)}$ where $R = \frac{h_2}{h_1}$; then $a_1 = \frac{x}{e}(d_4-d_3)$

(2) F_{4-3}, F_{3-2} is load between stations 4 and 3; 3 and 2, etc. (Item 6)

(3) S_3, S_2 etc. is shear at stations 3, 2, etc. (Item 9)

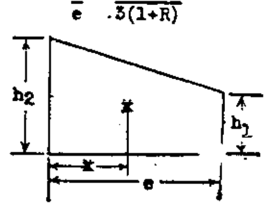


Figure 3–6. Determination of shears and bending moments.

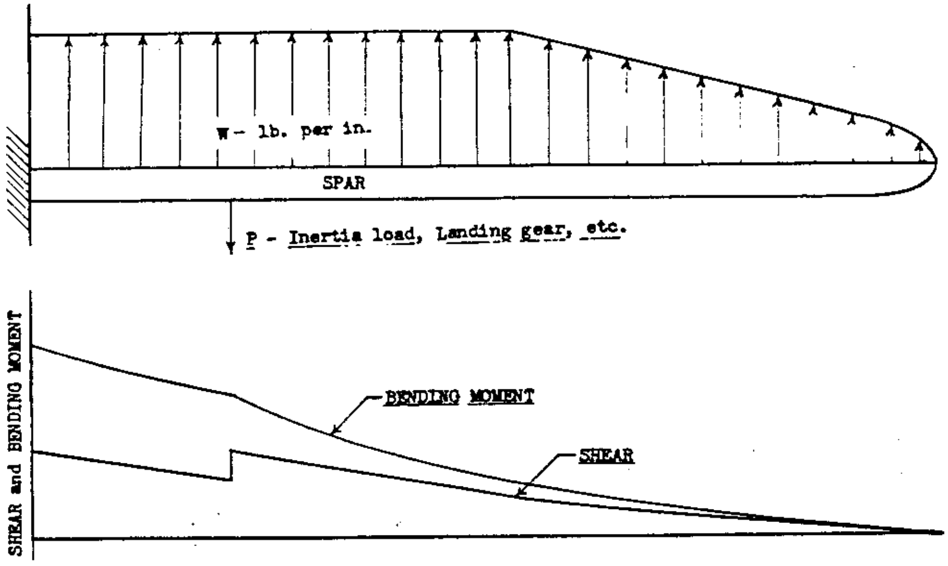
The primary bending moments at various stations on a cantilever spar may be determined conveniently by the equation:

$$M_x = M_1 \pm S_1 x \pm \sum Fa \qquad (3:8)$$

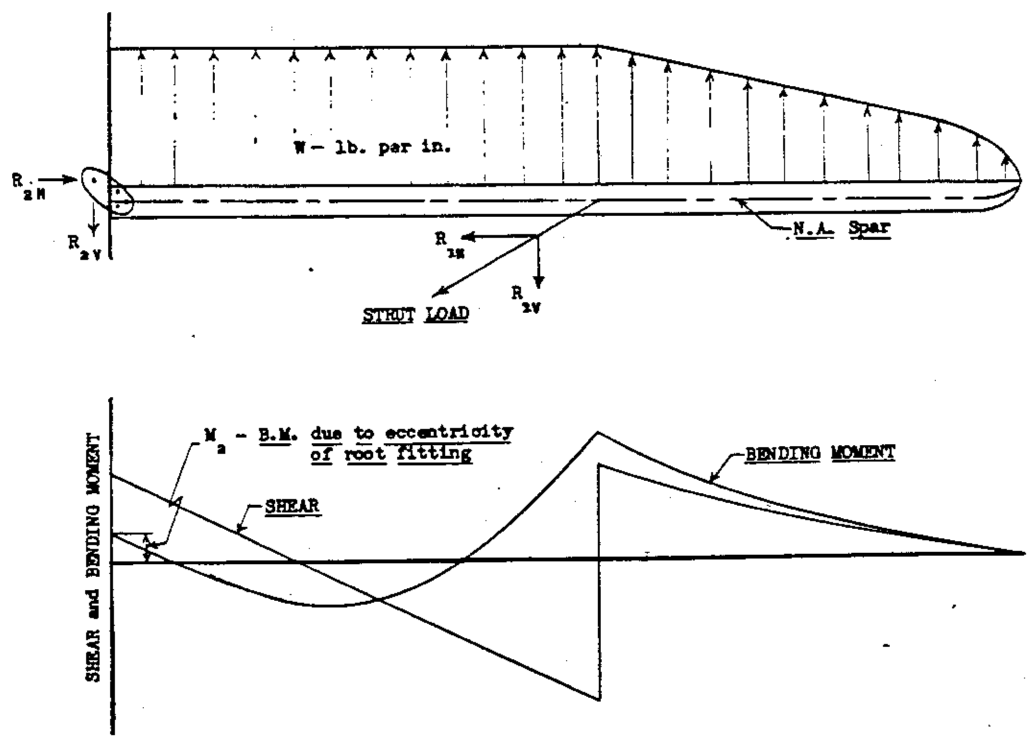
where M_1 and S_1 are the moment and shear at station *1*; x, the distance between station *1* and x; and $\sum Fa$, the sum of the moments about station x of all the loads acting between the stations. It will be found desirable to prepare a table similar to the one shown in figure 3–6 to facilitate the computations. If the distances between the various stations are relatively small, the center of gravities, a of the trapezoidal loadings may be assumed to lie midway between the stations with negligible error and slightly conservative results. If concentrated loads exist at points on the span, the table may be modified easily to account for these loads.

The case of an externally braced spar may be handled in a manner similar to that for the cantilever spar, insofar as the determination of the shears and moments outboard of the strut and the moment at the root due to external loads are concerned. The root moment required in section 3.121 to determine the lift-strut reactions may be obtained conveniently by the foregoing procedure.

The general form of the moment and shear curves is shown in figure 3–7, (*a*) and (*b*), for braced and cantilever spars. It always is desirable to plot the bending moment and shear curves as a general check of the computations and



(a) BENDING MOMENT and SHEAR DIAGRAM - CANTILEVER SPAR



(b) BENDING MOMENT AND SHEAR DIAGRAM - BRACED SPAR

Figure 3–7. (a) Bending moment and shear diagram—cantilever spar. (b) Bending moment and shear diagram—braced spar.

to facilitate the investigation of stations along the span not covered in figure 3–6.

3.1140. *Beam-column effects (secondary bending).* In connection with the bending moment and shear curves for a braced spar inboard of the strut point, where the spar is loaded as a beam and a column simultaneously, the effects of secondary bending should be taken into account by use of the "precise" equations or the "polar diagram" method. The solution of the beam-column problem is covered extensively in several textbooks relative to airplane structures, and, therefore, will not be covered here (refs. 3–1, 3–15). It is necessary, however, to base such computations on ultimate loads rather than on limit loads, in order to maintain the required factor of safety. Continuous spars having three or more supports should be investigated by means of the three-moment equation or other methods leading to equivalent results.

3.1141. *Effects of varying axial load and moment of inertia.* The drag-truss bays of a braced wing usually are shorter than the lift-truss bay, as indicated in figure 3–4. The axial loads in the spars due to the chord loading, therefore, vary along the span. Although the "precise" equations for a beam-column assume a constant value of axial load in the beam, it is generally satisfactory to determine a weighted value of axial load for use in determining the "precise" bending moment. Referring to figure 3–8:

$$P_c=\frac{P_1L_1+P_2L_2+P_3L_3}{L_1+L_2+L_3} \qquad (3:9)$$

where P_c is the weighted axial load due to chord loading, and P_1, P_2, and P_3 are the spar axial loads in the drag bays 1, 2, and 3. The total axial load in the spar is:

$$P_t=P_s+P_c \qquad (3:10)$$

where P_s is the spar axial-load component from the lift strut or wire.

Generally, the moment of inertia, I, also varies along the span and a weighted value of I may be determined for use in the "precise" equations, as follows:

$$I_w=\frac{I_1L_1+I_2L_2+I_3L_3}{L_1+L_2+L_3} \qquad (3:11)$$

where I_1, I_2, and I_3 are the moments of inertia in bays 1, 2, and 3. If the "polar diagram" method is used, the actual variation can be taken into account.

3.115. *Internal and allowable stresses for spars.*

3.1150. *General.* The allowable stresses for spars may be found in section 2.3. In beams subjected to combined bending and compression, the margin of safety computed by a simple comparison of the internal and allowable stresses may be meaningless, particularly when the beam-column is approaching the critical buckling point. True margins of safety may, therefore, be determined only by successive approximations. For example, if a spar is rechecked after increasing all external loads and moments by 10 percent, and still found satisfactory, the true margin of safety is at least 10 percent.

3.1151. *Wood spars.* In general, a spar will be subject to bending, axial (tension or compression), and shear stresses. The total stress due to bending and axial load may be computed by the usual expression:

$$f_t=\frac{Mc}{I}+\frac{P}{A} \qquad (3:12)$$

where M includes secondary bending. In computing the section properties of a wood spar, the following points are worthy of attention. Consider the spar section shown in figure 3–9.

(*a*) Where the two vertical faces of the spar are of different depths, the average depth of the section may be used, as shown by h.

(*b*) If the webs are plywood, only those plies parallel to the spar axis and one-quarter of those plies at 45° may be used in the computation of A and I of the sections. These are approximate rules to allow for the difference in modulus of elasticity of the plywood and the solid wood. If the plywood webs are neglected entirely, the computation of the section properties is simplified and the results are more conservative.

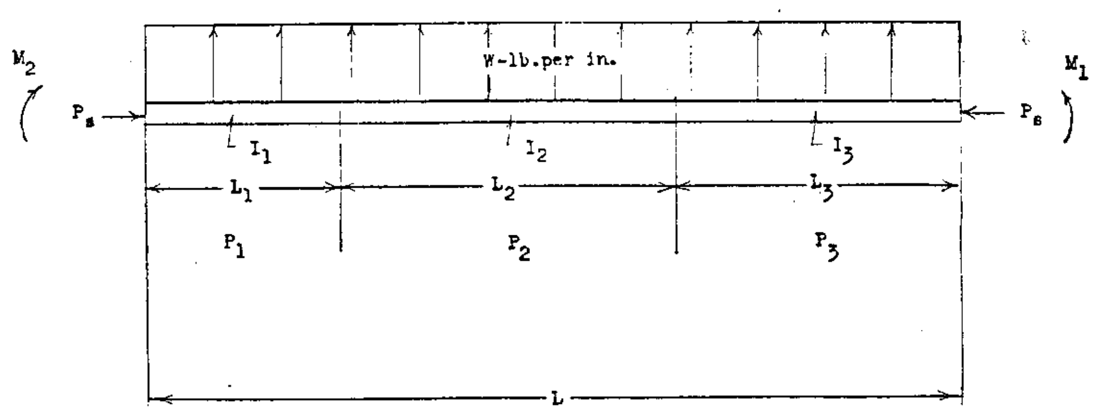


Figure 3–8. Distribution of forces on wood spar section.

(c) When investigating a section, such as A–A in figure 3–9, the full section should be considered effective only if the glue area is sufficient to develop the full strength of the side plates. In general, the distance a should not be less than 10 times or 15 times the thickness of a side plate for softwoods and hardwoods, respectively. The reinforcing blocks should be beveled, as shown, to prevent stress concentration which may lead to consequent failure in the glued joint at the edge of the reinforcement.

(d) Filler blocks may likewise be used in computing the section properties, provided the length of the blocks and their glue area to webs and flanges is sufficient to develop the required bending stresses.

(e) In the detailed investigation of a spar section, the reduction in strength due to bolt holes should be considered when computing the section properties. In computing the area, moment of inertia, etc., of wood spars pierced by bolts, the diameter of the bolt hole should be assumed greater than the actual diameter by the amount specified by the procuring or certificating agency. In computing the properties of section A–A (fig. 3–9), it should be assumed that all the bolt holes pass through the section, because failure might actually occur along the line u–v.

The longitudinal shear stress in the web of a spar may be obtained from the expression:

$$f_s = \frac{SQ}{b'I} \qquad (3:13)$$

In the determination of Q, for spars with plywood webs, the recommendations in (b) should be followed. However, the value of b' in the expression should be the total web thickness. For tapered spars, the shear stress may be reduced to allow for the effects of taper in accordance with section 3.1352.

3.116. *Special problems in the analysis of two-spar wings.*

3.1160. *Lateral buckling of spars.* For conventional two-spar wings, the strength of the spars against lateral buckling may be determined by considering the sum of the axial loads in both spars to be resisted by the spars acting together. The total allowable column strength of both spars is the sum of the column strengths of each spar acting as a column the length of a drag bay. Fabric wing covering may be assumed to increase the fixity coefficient to 1.5. When further stiffened by plywood or metal leading-edge covering extending over both surfaces forward of the front spar, the fixity coefficient may be assumed to be 3.0.

3.1161. *Ribs.* Analytical investigation of a rib generally is not acceptable as proof of the structure. In some cases, however, a rib may be substantiated by analysis when another rib of similar design has been analyzed, and subsequently strength-tested. In general, it may be desirable to analyze a rib in order to determine the approximate sizes of the members.

3.1162. *Fabric attachment.* Although the fabric-

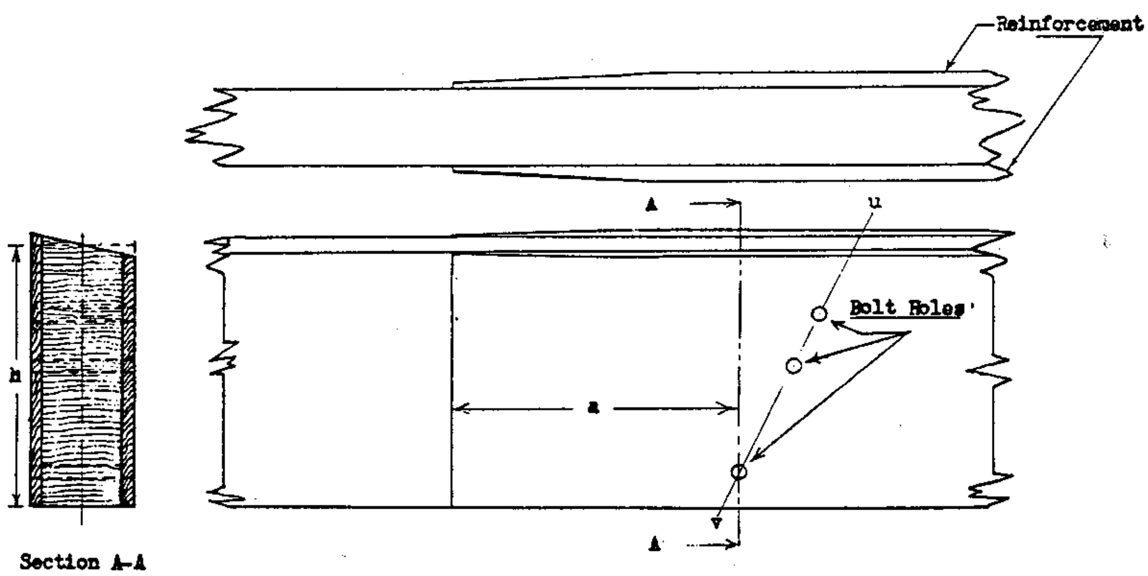


Figure 3–9. Wood spar section.

attaching method usually is not stress analyzed, it is, of course, important that the rib-lacing strength and spacing be such that the load will be adequately transmitted to the ribs. The specifications of the procurement or certificating agency in regard to lacing-cord strength and spacing should be followed. Unconventional fabric-attachment methods should be substantiated by static tests or equivalent means to the satisfaction of the agency involved.

3.12. TWO-SPAR PLYWOOD COVERED WINGS.

3.120. *Single covering.* Two-spar wings covered with plywood on only one surface (upper or lower) should be considered as independent spar wings, in accordance with section 3.11, and the plywood covering designed to carry the chordwise sheer loads with the ribs functioning as stiffeners and load distribution members. The center of shear resistance of the plywood covering may be eccentric to the applied drag load (fig. 3–15 *b*). The resulting torque will then be resisted by a couple consisting of up-and-down force on the two spars.

3.121. *Box type.* Two-spar wings with both upper and lower surfaces covered with plywood, forming a closed box, should be treated as shell wings in accordance with section 3.13.

3.13. REINFORCED SHELL WINGS.

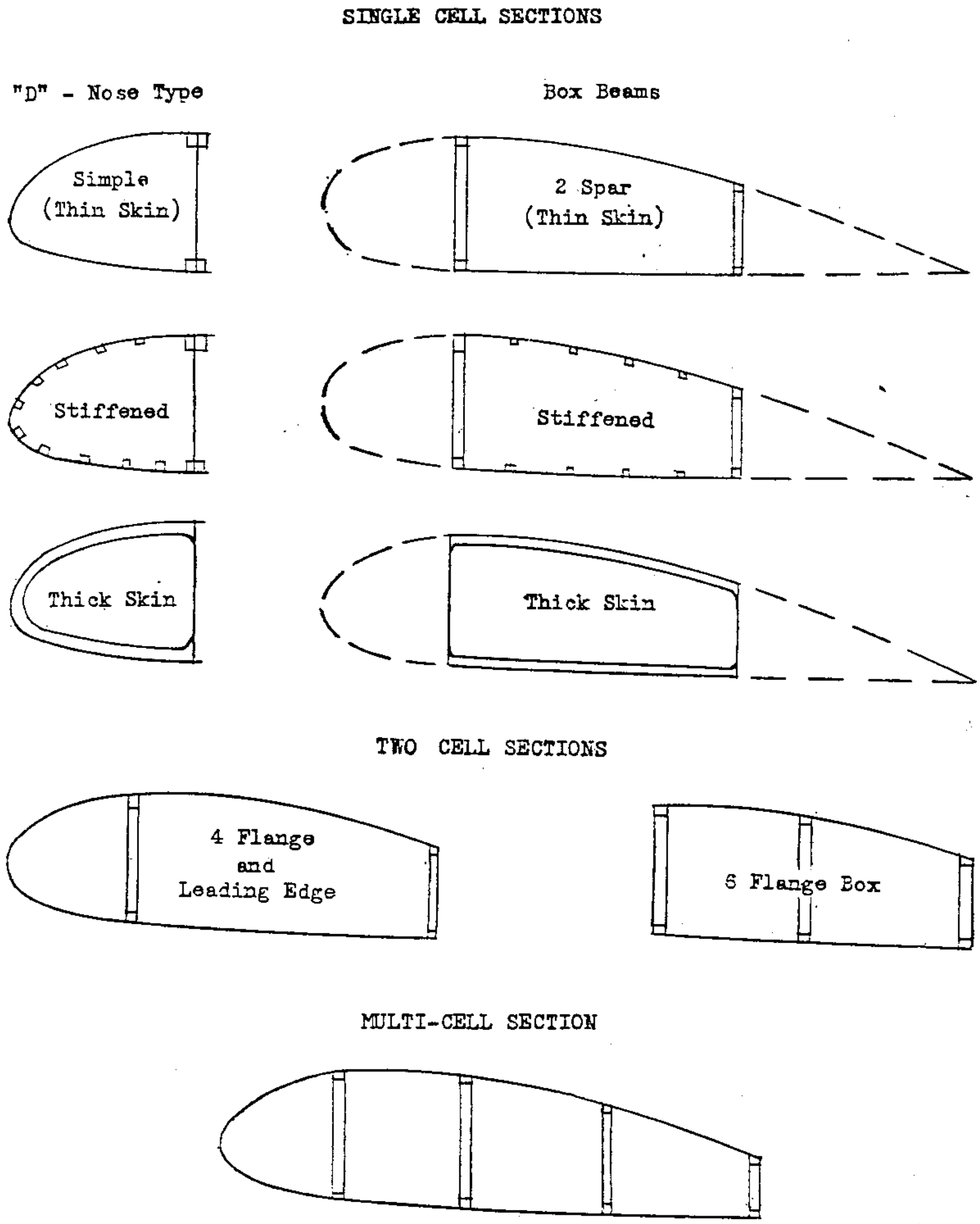
3.130. *General.* The types of wing structure considered under this heading are those in which the outside covering or skin, together with any supporting stiffeners, resists a substantial portion of the wing torsion and some of the bending. Various types of shell wings may be classified according to: the number of vertical shear webs, or number of "cells" into which these webs divide the wing section; whether the spanwise material is concentrated mainly at the shear webs or distributed around the periphery of the section as longitudinal stiffeners; whether the skin is "thin" so that it buckles appreciably at ultimate load, or "thick" so that it does not buckle appreciably. Typical shell wing sections are shown in figure 3–10.

In shell wings the distributed airloads normal to the surface are carried to the ribs by the skin and its stiffeners. The ribs maintain the shape of the section and transmit the airloads from the skin to the vertical shear webs or to other portions of the skin such as the leading edge, which are capable of carrying vertical shear. Main or "bulkhead" ribs perform similar functions for concentrated loads, such as those due to nacelle landing gear, and fuselage reactions. The vertical shear from the ribs is carried to the wing reaction points by the shear webs and portions of the skin. The shear in these elements creates axial bending stresses in the beam flange material. When comparatively stiff spanwise stiffeners are used, they also act as effective flange material, receiving their axial loads from the webs through shear in the skin. The contribution of the skin to the bending strength of the wing depends on its degree of buckling and relative modulus of elasticity.

From this general picture, it is evident that broad simplifying assumptions are necessary to make a stress analysis of a shell wing practicable, and that the computed stresses in the various elements are likely to be less exact than in the case of staticially determinate independent spar wings. In metal shell structures, elements which become too highly stressed generally yield without difficulty and the load is redistributed to less highly stressed elements. In wood structures, however, some types of elements are unable to accommodate themselves to secondary stresses which would be of no importance in metal structures, for example, buckles of sharp curvature relative to the thickness are apt to split plywood. The stress analysis methods presented in this section should therefore be considered only as reasonable approximations until the designer has had experience in applying a particular method to a particular type of structure and has correlated the analysis procedures with the results of static tests.

3.131. *Computation of loading curves.*

3.1310. *Loading axis.* In determining the shear and bending stresses in shell wings, it has been found convenient to transfer the distributed air and inertia loads to a suitable spanwise loading axis by computing net beam, chord, and torque loadings at points or stations along such axis. The position of the loading axis may be chosen arbitrarily if the corresponding moment and torque components acting at a particular section of the wing are then properly applied to the various elements of the section in a manner consistent with their structural behavior. Since a reinforced shell wing is usually a complex nonisotropic structure in which some of the elements resist axial loads in a particular direction only, the true stress conditions resulting from the interaction of elements having various directions at a given section are often difficult to analyze. It is therefore recommended that the loading axis be located inside the wing, approximately parallel to the



Note: Other types of Two Cell Wing Sections may have stiffeners or thick skin similar to the single cells shown above.

Figure 3–10. Typical shell wing sections.

principal bending and shear elements. Such a location should tend to reduce errors in the process of transferring external loads and torques to the loading axis and redistributing them to the structural elements. Section 3.135 shows that the use of a loading axis in the main shear web is often convenient for the shear distribution analysis, without further transfer of loads and torques.

If the loading axis is located as suggested, it is necessary for it to change direction where the principal structural elements change direction; for example, where an outer wing panel having dihedral or sweepback joins a straight center section. The loadings due to the air and inertia loads are computed for each segment of the axis in the usual manner, but at the point of direction change, the total moments and torque from the outboard segment should be resolved into the proper components relative to the inboard segment.

The formulas given in section 3.1311 for computing the running loads and torque at various stations on the loading axis use airfoil moment coefficients (or center of pressure locations) based on airfoil sections parallel to the airflow. For a loading axis which is not perpendicular to such sections, these equations will therefore give small errors in the bending moment and torque values. These errors may be neglected unless the angle of inclination of the loading axis is large.

3.1311. *Loading formulas.* The net running load at points along the loading axis and the net running torsion about these points may be found from the following equations:

$$y_b=(C_N q+n_z e)\frac{C'}{144} \tag{3:14}$$

$$y_c=[C_C q+n_{x_2} e]\frac{C'}{144} \tag{3:15}$$

$$m_t=[\{C_N(x-a)+C_{M_a}\}q+n_2 e(x-j)]\frac{(C')^2}{144} \tag{3:16}$$

where:

y_b=running beam load in pounds *per inch of span.*

y_c=running chord loads in pounds *per inch of span.*

m_t=running torsion load in inch-pounds *per inch of span.*

a, j, and x are expressed as fractions of the chord at the station in question and locate points on figure 3–11 as follows:

a locates the point in the airfoil on which the moment coefficient, C_{M_a}, is based.

j locates the resultant wing dead weight at the station.

x is the distance from the leading edge to the loading axis, at the station.

q=dynamic pressure for the condition being investigated.

C_N and C_{M_a} are the airfoil normal and moment coefficients at the section in question.

C_C=airfoil chord coefficient at each station. The proper sign should be retained throughout the computations.

C'=the wing chord, *in inches.*

e=the average unit weight of the wing, in pounds per square foot, over the chord at the station in question. It should be computed or estimated for each area included between the wing stations investigated, unless the unit wing weight is substantially constant, in which case a constant value may be assumed. By properly correlating the values of e and j, the effects of local weights, such as fuel tanks and nacelles, can be accounted for directly.

n_2=the *net* limit load factor representing the inertia effect of the whole airplane acting at the center of gravity. The inertia load always acts in a direction opposite to the net air load. *For positively accelerated conditions n_2 will always be negative*, and vice versa. Its value and sign are obtained in the airplane balancing process.

$n_x{}^2$=net limit chord-load factor approximately representing the inertia effect of the whole airplane in the chord direction. The value and sign are

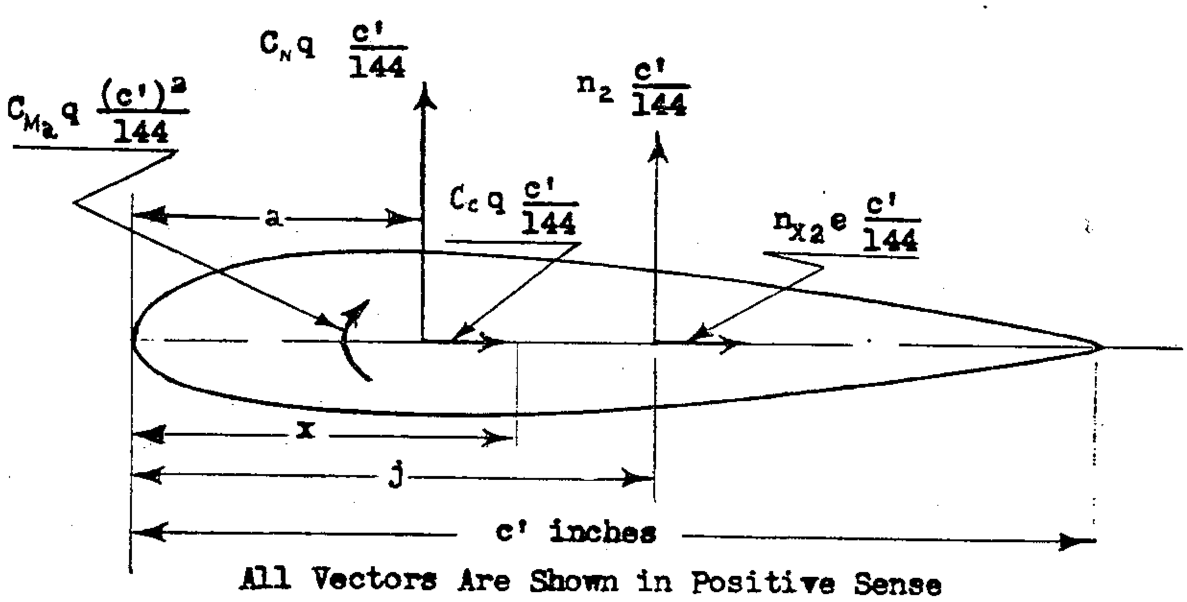


Figure 3–11. Section showing location of load axis.

Table 3–3. Computation of net loadings (constants)

		Stations Along Span				
1	Distance from root, inches					
2	C'/144 = (chord in inches) /144					
3	x, fraction of chord					
4	a, fraction of chord (a.c.)					
5	j, fraction of chord*					
6	e = unit wing wt., lbs/sq.ft.*					
7	x – a = ③–④					
8	x – j = ③–⑤					
9	$\frac{(C')^2}{144}$					

* These values will depend on the amount of disposable load carried in the wing.

obtained in the airplane balancing process. *Note that, when C_C is negative, n_x^2 will be positive.*

Positive directions for all quantities are shown in figure 3–11. The computations required for this form of analysis can be carried out conveniently through the use of tables similar to tables 3–3 and 3–4.

The values of y_b, y_c, and m_t should be plotted against the span and, in case irregularities are found, they should be checked before proceeding with the calculations.

It is sometimes desirable to compute the airloadings and inertia loadings separately. The inertia loading, shear, moment and torsion curves then need be computed for only one condition (say, $n_2=1.0$), the values for any other condition being obtained by multiplying by the proper load factor. The foregoing formulas may be modified for this purpose by omitting terms containing n_2 for the airloading, and omitting terms containing q for the inertia loading.

3.132. *Computation of shear, bending moment and torsion.* The summation of the areas under the loading curves determined by the method described in section 3.131, from the tip to any wing station will give the values of the total load (shear) and of the total torque (torsion) acting at the station.

It is advisable to plot curves of the shear and torsion values against the span to determine if any irregularities have occurred in the computations. If concentrated weight and load items were not accounted for in the loading computations, they should be taken care of by additional computations, and their effects shown on the shear and torsion curves.

The bending moments at any station of the wing can be found either by computing the moments, about the station, of the areas under the loading curves outboard of the station, taking into consideration moments due to concentrated loads, if such are present; or by summing up the areas under the shear curves from the tip to the station. A convenient tabular method of computing these values is also shown in figure 3–6; and typical curves are shown in figure 3–7.

The following quantities are now assumed to have been determined and plotted for any station on the loading axis:

S_{b_L}, the total beam load (shear) through the loading axis in pounds.

S_{c_L}, the total chord load (shear) through the loading axis in pounds.

M_{t_L}, the torsion about the loading axis in inch-pounds.

M_{b_L}, the beam moment in inch-pounds.

M_{c_L}, the chord moment in inch-pounds.

Formulas of section 3.1311 give moments and torques whose magnitudes and directions are not necessarily consistent with the direction of the loading axis, but the errors may usually be neglected (sec. 3.1310).

3.133. *Computation of bending stresses.* The methods outlined herein are based on the application of the conventional bending theory to the wing section as a whole, rather than to individual

Table 3–4. Computation of net loadings (variables)

CONDITION ___

q	$C_{N_{I(etc)}}$	C'_C	C'_M	n_2	n_{x_2}

			Distance b from root					
		(Refer also to Table 3–2)						
Normal Load	10	C_N (variation with span)						
	11	$C_N q$ = (10) x q						
	12	$n_2 e$ = (6) x n_2						
	13	(11) + (12)						
	14	y_b = (13) x (2) lbs./in.						
Chord Load	15	C'_C (variation with span)						
	16	$C'_C q$ = (15) x q						
	17	$n_{x_2} e$ = (6) x n_{x_2}						
	18	(16) + (17)						
	19	y_c = (18) x (2) lbs./in.						
Unit Torque	20	C_{M_a} (variation with span						
	21	(7) x (10)						
	22	(20) + (21)						
	23	(22) x q						
	24	(12) x (8)						
	25	(23) + (24)						
	26	m_t = (25) x (9)						

spars deflecting independently. It is assumed that the axial deformation due to bending, for any element of the wing section, is proportional to the distance of the element from the neutral axis of the section. This means that in multispar shell wings the deflection of all spars is assumed to be substantially the same. These assumptions are valid only where the wing contains relatively rigid torsion cells so that wing twist is resisted by shear in the walls of these cells rather than differential bending of the beams. Experience indicates that this simple bending theory is satisfactory for the practical design of shell wings if allowances or corrections are made for the following conditions:

(1) Excessive shear lag, or shear deflection, in the shell between various bending elements. Such deflections cause the actual stresses in elements remote from the vertical shear webs to be less than, and the stresses in elements adjacent to the shear webs greater than, the values indicated by the simple bending theory. In some types of structures as described in section 3.1330 (5), these deflections may be considered negligible in the design of the wing as a whole. Since the bending elements receive and give up their axial loads through shear in the webs or skin to which they are attached, local shear stresses and deflections will be intensified in the region of discontinuities in the bending or shear elements. Shear lag is therefore likely to be appreciable in such regions. A convenient method of allowing for shear lag is to assume a reduced effective area for the bending elements affected, in computing the section properties as described in section 3.1330. The stresses computed for such elements by the bending theory will then be too high, and, to be consistent, should be reduced in the same ratio as the areas used in the section properties.

(2) The effects of torsion on the bending stresses at the corners of a box beam.

This condition is usually dealt with after the bending stresses and shear distribution have been determined on the basis of the simple theory. See section 3.1370 for discussion.

3.1330. *Section properties.* A sufficient number of stations along the wing should be investigated to determine the minimum margins of safety. The information necessary to compute the section properties at each station selected for investigation may be conveniently obtained from a scale diagram of the wing section. Such a diagram (fig. 3–12) and accompanying data should show the following:

(1) All material assumed acting in shear or bending (sec. 3.138) divided into suitable elementary strips and areas, with each such element designated by a suitable item number for use in tabular computations.

(2) Thicknesses of skin and web elements, area and center of gravity of stiffeners and flanges, and the relative moduli of elasticity of all elements, normal to the section (secs. 2.1210, 2.52, and 3.138, or table 2–13). For example, the modulus of the beam flanges might be taken as a basic in tension and the moduli of other elements expressed as ratios thereto.

(3) Reference axes from which the various elements are located. The amount of calculation will generally be less if the reference axes are made parallel to the beam and chord directions used in the loading curve determinations.

(4) Effective widths of skin assumed acting in compression in conjunction with stiffeners or flanges. These should be consistent with the methods used in determining allowable stresses, in accordance with section 3.138.

(5) Effectiveness factors for bending elements which have elastic modulus different from the basic value selected for the wing, or which are effected by shear lag. The final factor, e, includes both effects, and may be expressed as: $e=e_1\times e_2$, where e_1 is equal to $\frac{E_{element}}{E_{basic}}$, and e_2 is the shear lag factor.

A value of $e_2=1.0$ indicates that the effectiveness of an element is not considered reduced by shear lag, while $e_2=0$ indicates that it is completely ineffective. Shear lag may be general or local or a combination of both. General shear lag is greatest in a shell wing which has a major portion of the bending elements remote from the shear webs, relatively thin skin, and little or no taper in plan and front views. The general shear-lag effectiveness factors for such wings should be based on rational analysis or test data for similar wings, unless the spar web flanges can withstand stresses considerably higher than those computed by the simple bending theory (refs. 3–4, 3–9, and 3–13). In a wing having characteristics opposite to those described, general shear

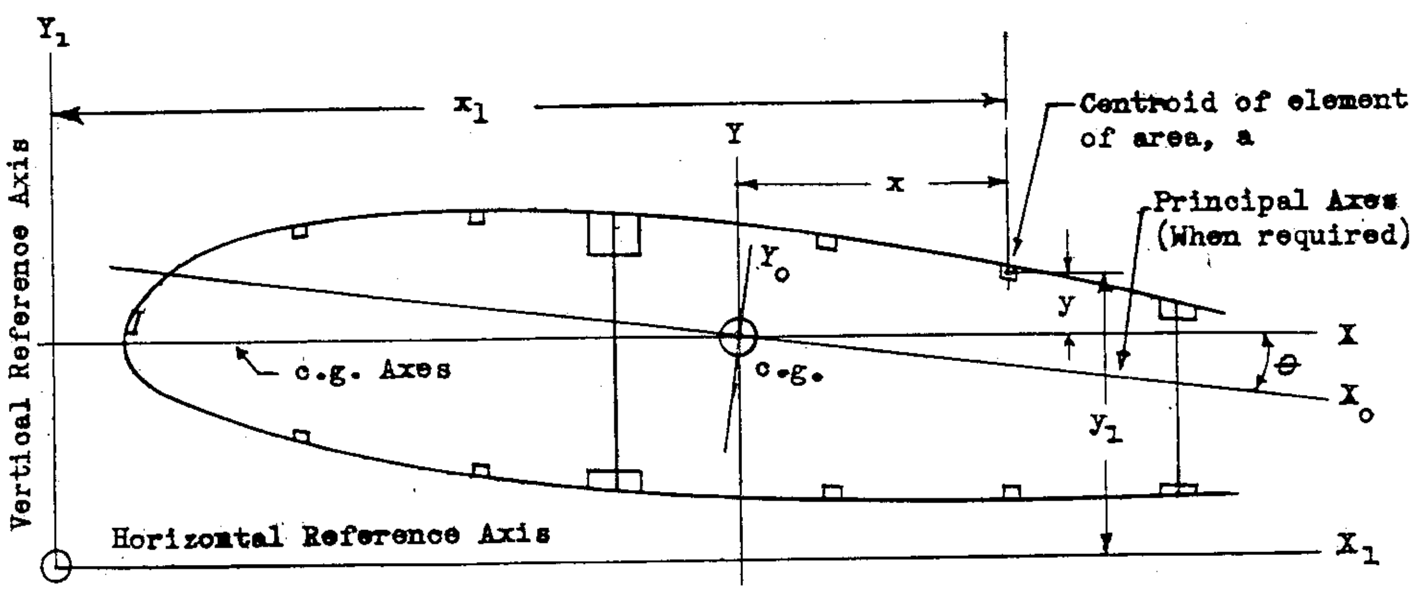


Figure 3–12. Diagram for computation of section properties.

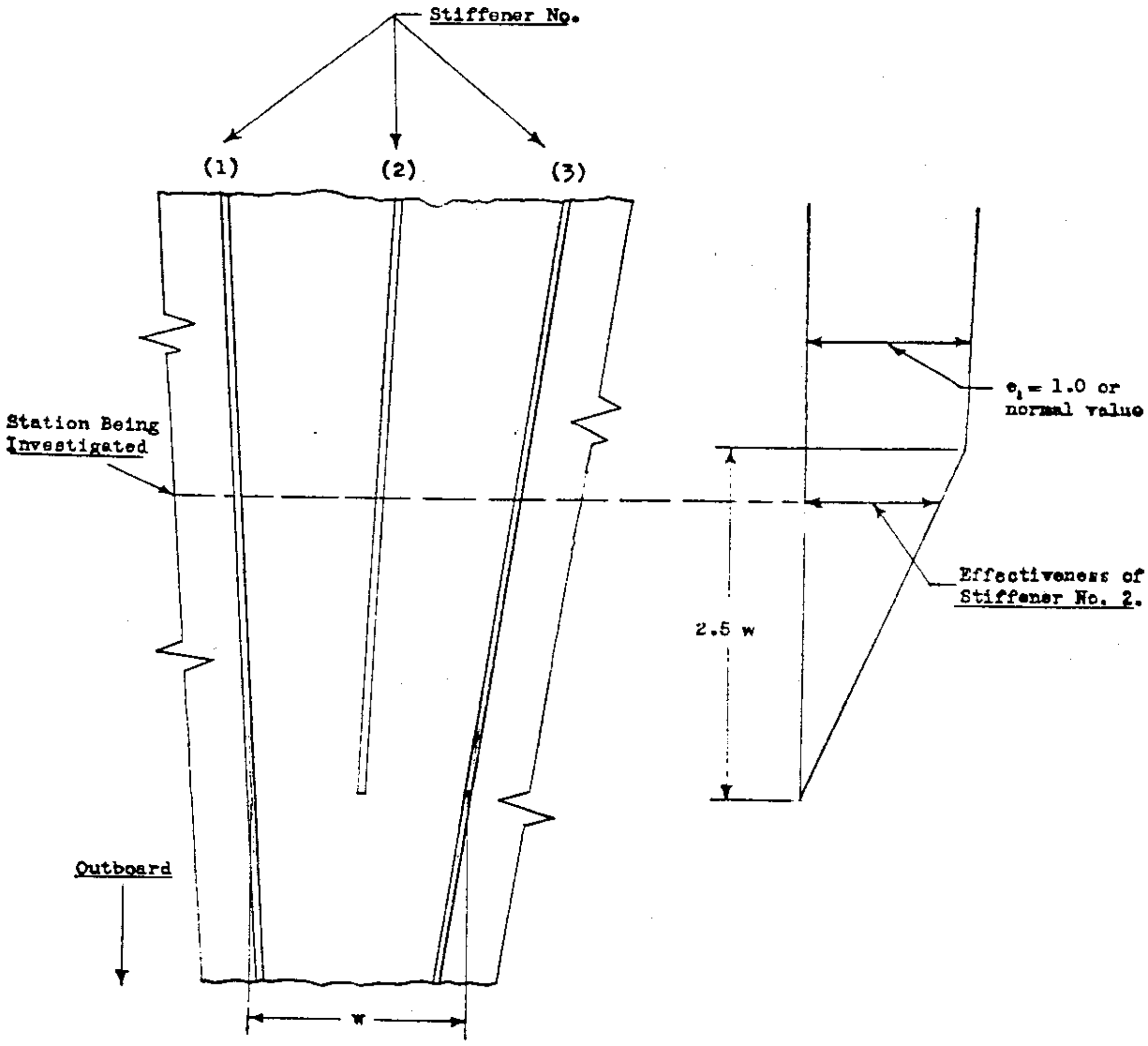


Figure 3–13. Effectiveness of discontinuous stiffener.

lag may be neglected if the spar flanges can withstand stresses slightly larger than those computed by the simple bending theory. Local shear lag due to discontinuities and cutouts may be estimated by determining e_2 from figures 3–13 and 3–14, or computed by methods of reference 3–13.

In using figure 3–14, L may be taken as $2.5W$ for conventional constructions employing stiff 45° plywood skin. A more rational value for L, applicable to all grain directions, may be computed from the following formula which takes into account the shear rigidity of the skin in relation to the axial load:

$$L=\frac{1.25\,W}{\sqrt{\frac{G\,t\,W}{E'A}}} \qquad (3:17)$$

where

W = width of cutout or free end.

G = effective shear modulus of skin.

t = thickness of skin.

E' = effective modulus of elasticity of composite section in tension or compression.

A = total effective area of skin and stiffeners in tension or compression.

With the foregoing information available, the wing-section properties may be computed in a tabular form, such as shown on table 3–5, the column headings meaning:

(1) Effectiveness factor for item, e.

(2) (*a*) Geometrical area of item, (A).
 (*b*) Effective area of item, (a), $=eA$.

(3) Beam distance of item from reference axis (y_1).

(5) Beam moment of area about the reference axis, (ay_1).

The location of the X axis, passing through the center of gravity and parallel to the horizontal reference axis, should next be determined by dividing $\sum$ *col. (5)* by $\sum$ *col. (2b)*.

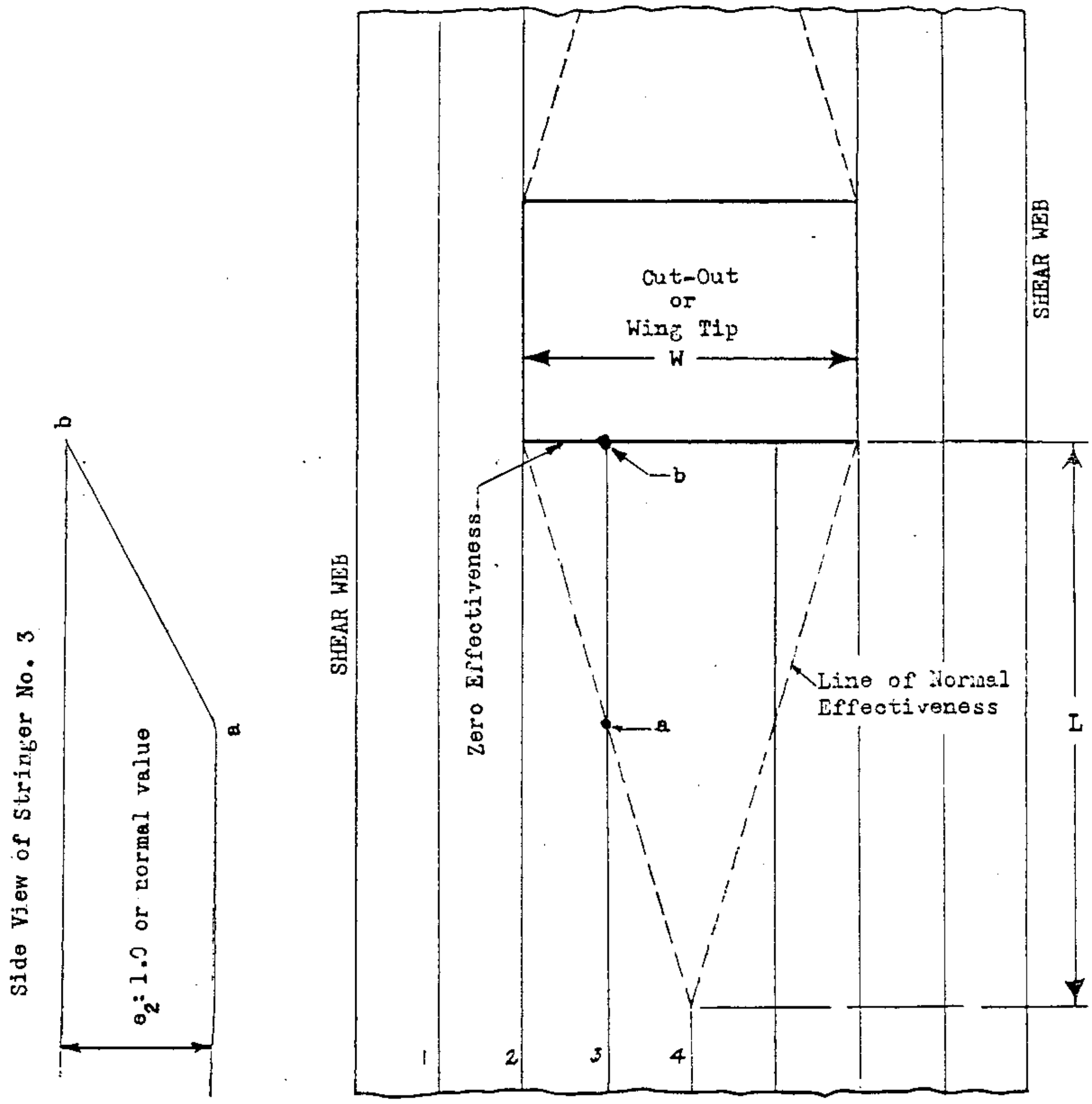


Figure 3–14. Effectiveness of stringers at cutout.

(7) Beam distance of item from the X axis passing through the center of gravity (y).

(9) Beam moment of the area about the X axis, (ay).

(11) Second beam moment of area about the X axis, (ay^2).

(13) Individual moments of inertia of items which are of sufficient magnitude to be included.

The sum of the items in column 9 for all of the wing elements above or all of the wing elements below the X axis is equal to the static moment of the section Q_x. The sum of items in columns 11 and 13 is equal to the moment of inertia of the wing section about the X axis. By a similar process, the wing-section properties about the Y axis can be determined by filling out the remaining columns in table 3–5 pertaining to chord distances and moments. The X and Y axis are *not* necessarily the principal axes.

The sum of all of the items in column 15 is equal to the product of inertia of the section about the center of gravity axes. Careful attention should be paid to the use of the proper signs in computing the products of inertia and in the subsequent stress calculations.

When effective widths are used for skin in compression, it is evident that the section properties may change for inverted loads, and in such cases the necessary computations should be repeated accordingly.

3.1331. *Bending stress formulas.* The following formulas may be used for the computation of the bending stresses at any point on the wing section. These formulas are similar to those described in section 6:6 of reference 3–15, and permit the stresses to be computed without determining the principal axes of inertia or the section properties relative thereto.

$$f'=-\frac{\overline{M}_b y}{I_x}-\frac{\overline{M}_c x}{I_y} \qquad (3:18)$$

where

$$\overline{M}_b=\frac{M_b-M_c\frac{I_{xy}}{I_y}}{I-\frac{(I_{xy})^2}{I_xI_y}}, \text{ and } \overline{M}_c=\frac{M_c-M_b\frac{I_{xy}}{I_x}}{I-\frac{(I_{xy})^2}{I_xI_y}}$$

The values of M_b and M_c are the values of the bending moments about the X and Y axes, respectively, used in the section properties computations; the I values are determined by the methods outlined in table 3–5, and the x and y values are the distances to the points at which the bending stresses are desired.

If the analysis of some of the wing sections indicates that the value of I_{xy} is approaching zero, it is apparent that the reference axes chosen are nearly parallel to the section principal axes, and the analysis of similar wing sections may be simplified by omitting the computation of the product of inertia in table 3–5. The expression for the stress at any point in this case simplifies to:

$$f'=-\frac{M_b x}{I_x}-\frac{M_c y}{I_y} \qquad (3:19)$$

When desired, the angle of inclination of the principal axes of inertia to the XY axes is given by the following relation (fig. 3–12):

$$\text{Tan } 2\theta=\frac{2I_{xy}}{I_x-I_y} \qquad (3:20)$$

where the values on the right side of the equation are obtained from table 3–5.

The stress f' computed by the formulas applies directly only to elements having the elastic modulus selected as basic for the section, and a shear lag effectiveness factor of 1.0. The actual stress f for other elements is obtained by multiplying f' from the formulas by the proper effectiveness factor from table 3–5.

3.134. *Secondary stresses in bending elements.*

(*a*) *Air loads and bending deflections.* Stiffeners are normally subjected to combined compression and bending. The compression results from the stiffener acting as a part of the flange material of the entire section. Two of the conditions producing bending in the stiffeners are: Part of the normal airload on the skin being carried to the ribs by the stiffeners, and curvature of the stiffeners due to bending deflection of the entire wing. Allowance for these bending loads may be made by using conservative values for the allowable compressive stress or, in relatively large rib spacings, by suitable computations and tests.

(*b*) *Diagonal tension field effects.* When the wing covering buckles in shear, additional stresses may be imposed on the spanwise stiffeners by the diagonal-tension field effects in the skin. If the initial buckling shear stress is greatly exceeded, it may be necessary to make additional analyses to account for the increased stiffener stresses. Shear buckles (diagonal tension fields) in curved skin tend to produce bending or sagging of the stiffeners between the ribs. Particular attention should be paid to the possibilities of the sagging type of failure in spanwise leading-edge stiffeners, especially when they are also subjected to combined beam and chord compressive loads. Combined loading tests or conservative allowable stresses based on simple tests in accordance with section 3.1381 should therefore be employed for D-nose spar and similar types of wings.

(*c*) Bending stresses due to torsion are discussed in section 3.1370.

3.135. *Computation of shear flows and stresses.*

3.1350. *General.* The methods outlined herein are based on the following principles: (refs. 3–5 and 3–11).

(1) The shear flows producing bending in the wing (direct shear) are distributed by the various shear elements to each ending element in such a manner as to produce the increase in axial load per unit of span required by the bending theory. In applying this principle, use is made of the computations performed in determining the bending stresses, and the results are affected by the same basic assumptions and limitations.

(2) The shear flows in the various shear elements of a torque box or cell are assumed to produce (or resist) torque about a reference point in accordance with the

Table 3-5. Wing-section properties computations.

	(1)	(2)		(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)	(15)
Items	Effectiveness Factor, e.	Area		Distance from Ref. Axis		Moment about Ref. Axis		Distance from Axis Passing Through C. G.		First Moment about C.G. Axis		Second Moment about C.G. Axis		Individual Moments of Inertia		Product of Inertia
		(a)	(b)	Beam	Chord	Beam	Chord	Beam	Chord	Beam	Chord	Beam	Chord	Beam	Chord	
		Actual	Effective	y_1	x_1	ay_1	ax_1	y	x	ay	ax	ay^2	ax^2	I_{ox}	I_{oy}	axy
			$\Sigma a =$			$= \Sigma ay_1$	$= \Sigma ax_1$			$= Q_x$	$= Q_y$	$= \Sigma ay^2$	$= \Sigma ax^2$	$= \Sigma I_{ox}$	$= \Sigma I_{oy}$	$= \Sigma axy$

$$I_x = \Sigma ay^2 + \Sigma I_{ox}$$

$$I_y = \Sigma ax^2 + \Sigma I_{oy}$$

elementary principles of shear flows, as illustrated in figure 3–15. This assumption is valid only where: The ribs and bulkheads are rigid in shear in their own plane, particularly at concentrated loads; the length of the torque box, or the distance from the section where a large concentrated torque, applied to the section where it is reacted, is relatively greater than the cross-sectional dimensions of the box; and where the cross sections of the wing are free to warp when the wing twists, as in a wing panel which is so joined to the center section that only the main beam can transmit bending, the remaining webs being pin-jointed. When any of these conditions are seriously violated, conservative overlapping assumptions should be made as to the shear in the various elements.

3.1351. *Shear flow absorbed by bending elements.* The rational methods for shear distribution first require the determination of the shear flows absorbed by the individual bending elements which may be determined by one of the following methods:

(1) *Spanwise method.* The spanwise method requires the calculation of the total axial load in each bending element at various stations along the span. The change in axial load per inch of span at any point is then equal to the shear flow being absorbed by the element at that point.

This method takes account of beam taper, discontinuities and redistribution of bending material, and is therefore particularly applicable to complex structures where these conditions are involved to a considerable degree. The average axial stress, f', (in terms of the "basic" elastic modulus) in each element having small depth compared to the whole section at a particular station may be obtained by substituting the x and y coordinates of the centroid of the element in the bending stress formula of section 3.1331. The total axial load, P, equals $f' \times a$, where a is the effective area of the element from the section properties computations. The shear flow, Δq, absorbed by the element is:

$$\Delta q = \frac{dP}{dZ} \tag{3:21}$$

where $\frac{dP}{dZ}$ is obtained by plotting P against the distance, Z, along the span, and finding the slope of the tangent at desired points. Δq may be most conveniently found by tabular methods, that is: $\Delta q = (P_2 - P_1)/\Delta z$, where P_1 and P_2 are the axial loads at two adjacent stations and Δz is the distance between them. Δq is considered positive when it tends to increase the tension on an element, proceeding from outboard to inboard, as shown in figure 3–16. A more complete description of this method is given in reference 3–18.

(2) *Section method.* The section method determines the shear flow absorbed by the bending elements by considering one section at a time under the external shears at that section, with separate corrections, if desired, for the effects of wing taper. This method is obviously not correct for sections in the vicinity of cutouts on wings having distributed bending material. It is, therefore, more applicable to wings where the bending material is concentrated in beams which taper uniformly. The shear flow absorbed by any bending element is obtained from formulas similar to those for the bending stresses (equation 3:18), using the same section properties computations, as follows:

$$\Delta q = a\left[-\frac{V_y}{I_x} - \frac{D_x}{I_y}\right] \tag{3:22}$$

$$V = \frac{S_b' - S_c' \frac{I_{xy}}{I_y}}{1 - \frac{(I_{xy})^2}{I_x I_y}} \tag{3:23}$$

$$D = \frac{S_c - S_b' \frac{I_{xy}}{I_x}}{1 - \frac{(I_{xy})^2}{I_x I_y}} \tag{3:24}$$

where:

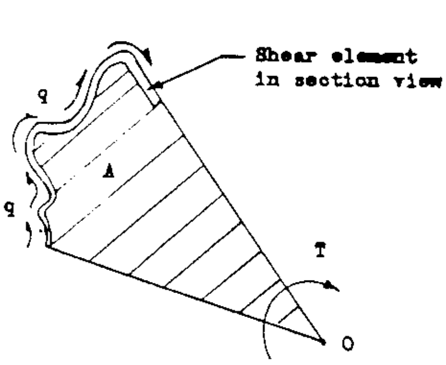
a = effective area of element.

x and y are coordinates of controid of element from section diagram. Deep elements, such as solid spars, should be broken into smaller elements.

I_x, table 3–5.

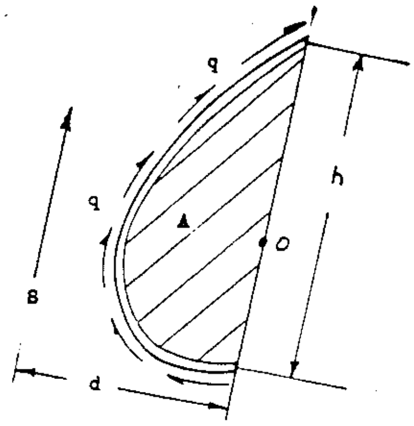
I_y, table 3–5.

$I_{xy} = \sum axy$, col. 15, table 3–5.



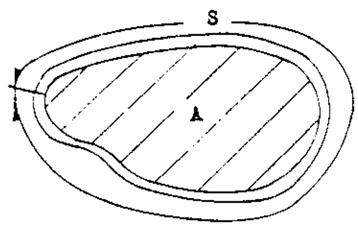
$T = 2qA$

(a) TORQUE



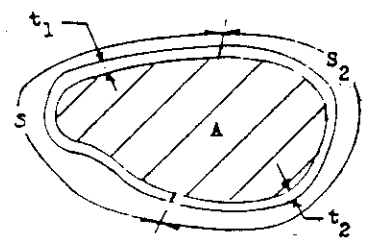
$S = qh$
$Sd = T = 2qA$
$qhd = 2qA$
$d = \frac{2A}{h}$

(b) RESULTANT SHEAR



G constant
t constant

$\theta = \frac{1}{2GA} \, q \, \frac{s}{t}$



$\theta = \frac{1}{2A} \sum q \, \frac{s}{Gt}$

(c) TWIST OF SHEAR CELL

SYMBOLS

q = shear applied per inch of shear element in section view. (Lb. per in.)
S = resultant of total shear acting on shear element.
s = length of median line of shear element in section view.
t = thickness of shear element.
f_s = shear stress (psi.) = $\frac{q}{t}$
h = length of chord joining ends of shear element.
o = reference point about which torque is taken.
A = area enclosed between median line of shear element and radii drawn from extremities to O.
θ = angle of twist of shear cell (radians) per inch of length normal to the section.
G = modulus of rigidity of portion of cell wall.
T = torque about reference point.

Figure 3–15. Properties of shear flows.

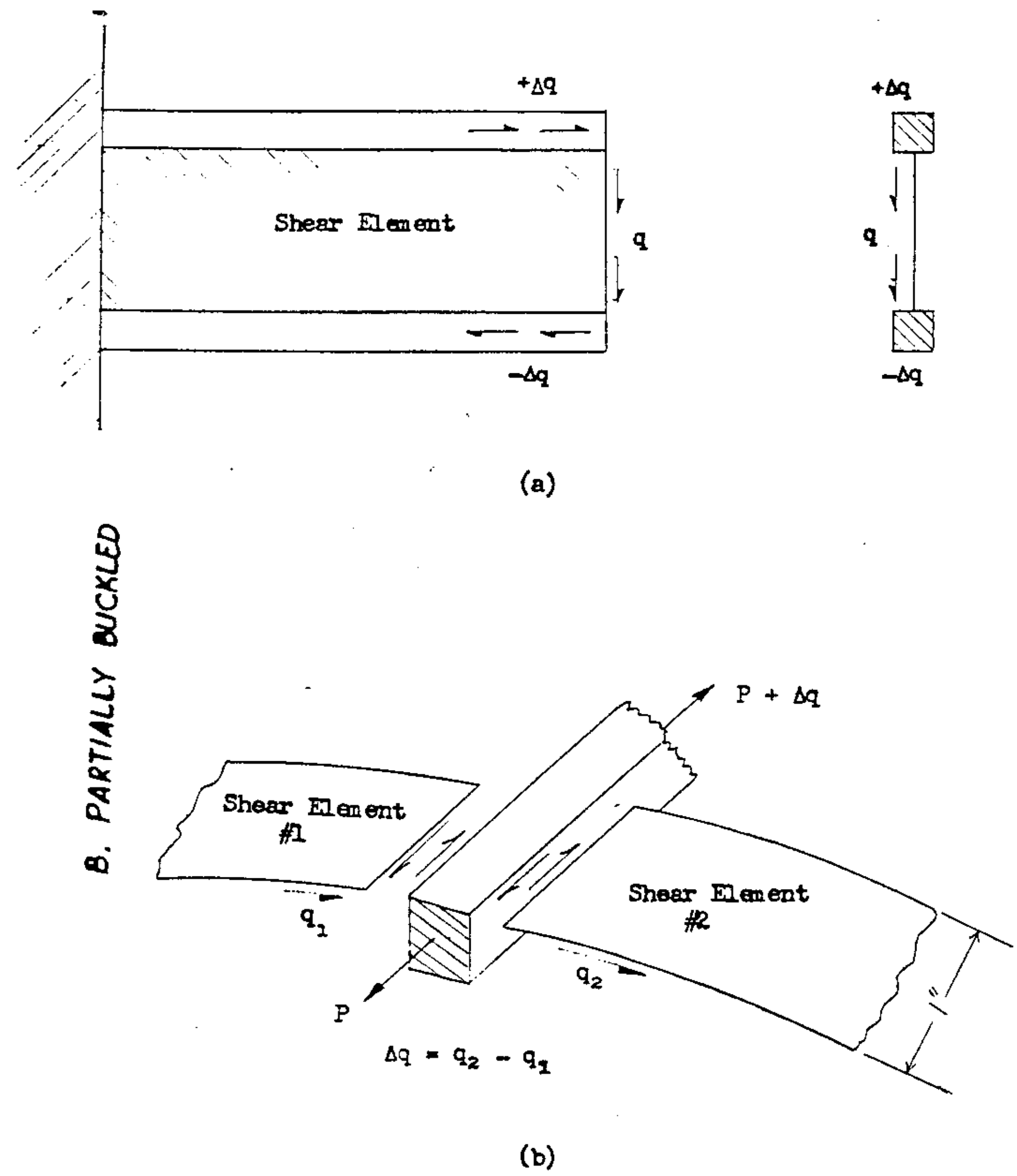


Figure 3–16. Sign conventions for shear flows.

S_b' = the total external beamwise shear (parallel to the Y reference axis for the section) resisted by the shear elements at the section, positive upward. It may include a shear correction due to taper in depth, as described in section 3.1352.

S_c' = the total external chordwise shear (parallel to the X axis) resisted by the shear elements at the section, positive rearward. It may include a shear correction due to taper in plan view.

3.1352. *Shear correction for beam taper.* When a beam having concentrated flanges is tapered in depth, a part of the external shear at any station is resisted by components of the axial loads in the flanges, as shown in figure 3–17. That part of the shear resisted by the flange axial loads is: $\Delta S = \frac{M}{L_o}$, where M is the moment at the station and L_o is the distance from the station to the point where centerlines of the flanges would meet if prolonged. The shear resisted by the shear elements is then: $S'_b = S_b - \Delta S_b$. If the flange material is distributed over the wing surface a conservative average taper may be assumed. These corrections for taper should not be used with the spanwise method of determining shear flow absorbed by bending elements.

3.1353. *Simple D spar.* The type of structure considered under this heading is shown in figure 3–18. The method described herein is rational in regard to beamwise shear and torque if the following idealizing assumptions are applicable. The beamwise bending material is assumed concentrated in flanges at the vertical web; the leading edge is assumed to be thin, that is, not capable of carrying beamwise bending, and the leading edge strip (or equivalent material resisting chordwise

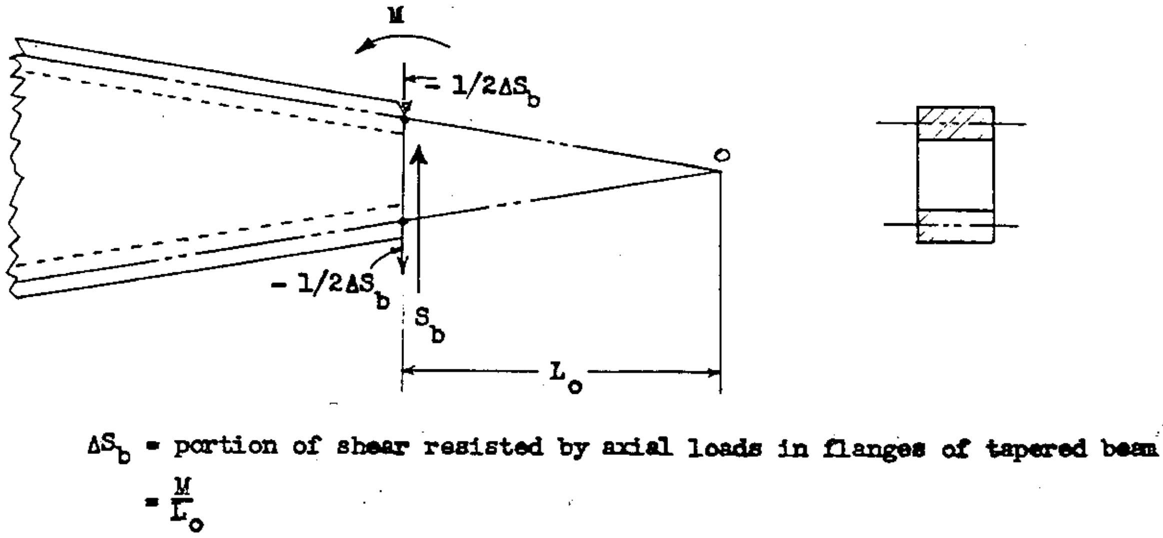


Figure 3–17. Shear correction for tapered beam.

bending), is assumed to be located so as not to be affected by beamwise bending nor to incline the principal axes to the vertical web. As in any single cell, the shear flow is statically determinate, and, under the above assumptions, readily apparent. If the external loads are transferred to a point on the neutral axis in the vertical web, as shears parallel and perpendicular to the web, and a torque about the point, as shown in figure 3–18 the parallel shear, S'_b, is resisted entirely by the vertical web, so that $q_b = S'_b/h$, where h is the height between the centroids of the flanges. The torque, M_t, is resisted by the torsion cell, requiring a shear flow around the periphery: $q_t = \frac{M_t}{2A}$, where A is the enclosed area.

The shear S'_c is assumed resisted equally by the upper and lower skin, so that: $q_c = S'_c/2d$, where d is the distance from the vertical web to the leading edge strip.

Then: q_w (vertical web)$=q_b-q_t$; and $q_{L.E.}=q_t \pm q_c$, with the sign conventions shown on the diagram.

If the bending material of a D-spar is largely distributed around the periphery in the form of a thick skin or spanwise stiffeners, the general rational method for single cells, described in the following, is more applicable.

3.1354. *Rational shear distribution.*

3.13540. *Single cell—general method.* The following method is applicable to single cell structures having the bending material distributed in the form of a thick skin or any number of concentrated flanges or stiffeners. However, when such material is in the form of thick skin, it is assumed divided into strips each of which is considered a concentrated element. Since the single cell is statically determinate, the elastic properties of the shear material are not necessarily involved in determining the stress distribution, although they are required in determining the twist or shear center. For simplicity, the shear center will not be used in computing shear flows and stresses. Its location may be readily determined after the shear flows are known. The method of computing shear flows is briefly outlined as follows: Referring to figure 3–19, the shear flow in the main vertical web is considered as an unknown, q_m, and the shear in each successive shear element around the periphery of the cell is expressed in terms of q_m by successively adding (algebraically) the shear flows, Δq_n, absorbed by the bending elements. The sum of the torques due to each shear element, about reference point O in the main vertical web, is then computed from the principles of shear flows (figure 3–15) and equated to the external

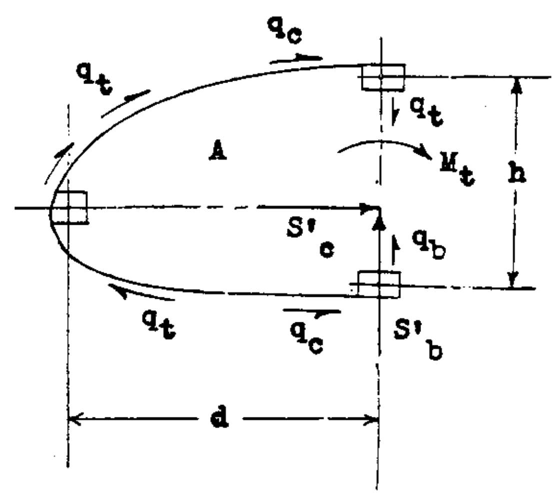


Figure 3–18. Shear in simple D-spar.

torque, M_t. This equation is solved for q_m, and the numerical values of the remaining shear flows obtained by successive addition of the Δq values, as explained. By using a suitable notation, the computations may be reduced to a simple tabular form as shown on table 3–6.

Such a notation is described as follows, and is illustrated in figure 3–19, where the assumed positive directions of quantities are as shown:

M_t = the resultant external moment applied at point O when the external shear S_b' and S_c' have been transferred to that point.

q_m = shear flow in main web.

q_1, q_2, q_3, etc., are shear flows in successive shear elements numbered clockwise around the section, as shown.

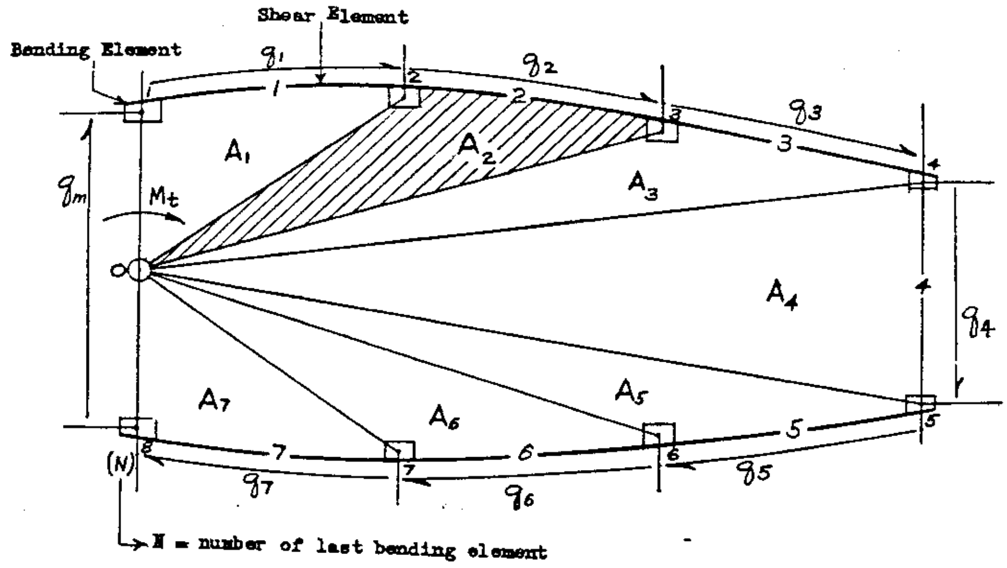
Table 3–6. Shear-flow computations for single cell

(1)	(2)	(3)	(4)	(5)	(6)
n	Δq_n	$\sum_1^n \Delta q_n = \sum_1^n (2)$	A_n	$A_n \sum_1^n \Delta q_n = (4) \times (3)$	$q_n = q_m + (3)$
1	Δq_1		A_1		
2	Δq_2		A_2		
3	Δq_3		A_3		
N − 1					
N					
	$\sum_1^N (2)$		$\sum_1^{N-1} (4)$	$\sum_1^{N-1} (5)$	

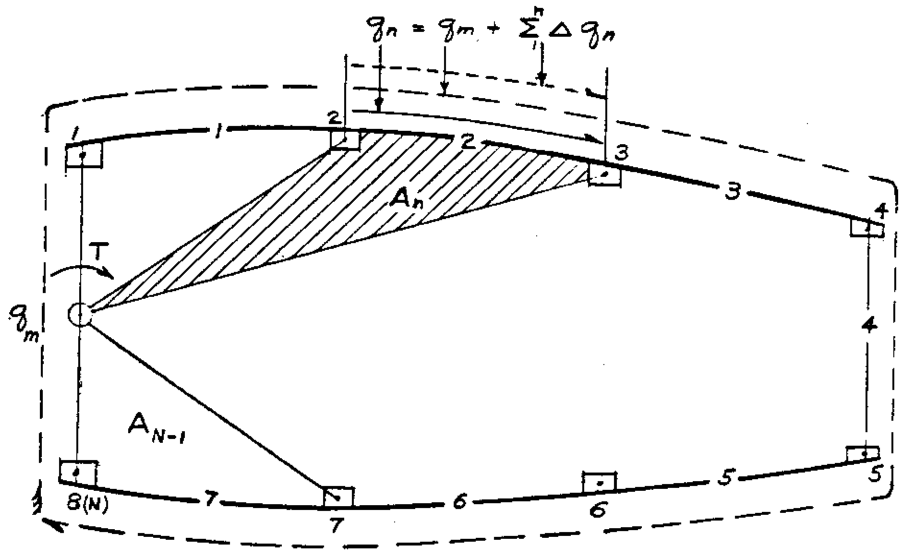
$$q_m = \frac{M_t}{2A} - \frac{1}{A} \sum_1^{N-1} (5)$$

Note: $\sum_1^N (2)$ should approximate 0.

$\sum_1^{N-1} (4)$ should approximate total area = A.



(a)



(b) GRAPHIC REPRESENTATION OF SHEAR FLOW EQUATIONS

Figure 3–19. Rational shear flow—single cell.

q_n=shear flow in *nth* shear element.

Δq_1, Δq_2, Δq_3, etc., are shear flows absorbed by bending elements correspondingly numbered. Δq is positive when it tends to produce tension in the bending element, as shown in figure 3–16. It is produced by (or requires) a resultant shear flow directed away from the element in section view. The values of Δq are assumed to have been determined by methods such as those of section 3.1351.

Δq_n=shear flow absorbed by *nth* shear element.

A_1, A_2, A_3, etc., are the areas enclosed between shear elements and radii from the reference point, O, to centroids of the bending elements.

A=enclosed area of entire section.

T=total torque of shear elements about point O.

$\sum_1^n$=summation of quantities for elements 1 through n, where n=1, 2, 3, etc.

N=number of last bending element (lower main flange).

$N-1$=number of last shear element (not counting main web).

The expressions for shear flow in any element in terms of q_m, using sign conventions of figure 3–16, are:

$$\Delta q_1 = q_1 - q_m \longrightarrow \Delta q_1 = q_m + \Delta q_1$$
$$\Delta q_2 = q_2 - q_1 \longrightarrow \Delta q_2 = q_m + \Delta q_1 + \Delta q_2$$
$$q_n = q_m + \sum_1^n \Delta q_n \qquad (3{:}25)$$

Equation (3:25) is represented graphically on diagram (*b*) figure 3–19 by a flow q_m around the entire section, to which is added flow $\sum_1^n \Delta q_n$ at any shear element to obtain the total flow q_n acting in that element.

The expression for the total torque of the shear elements about point *O*, figure 3–19(a), is:

$$T = \sum 2A_n q_n$$

or

$\frac{T}{2} = \sum A_n q_n$, which, from diagram (b) of figure 3–19

$$= Aq_m + \sum_1^{N-1}\left(A_n \sum_1^n \Delta q_n\right)$$

$= \frac{M_t}{2}$ (equilibrium of internal and external loads

$$Aq_m = \frac{M_t}{2} - \sum_1^{N-1}\left(A_n \sum_1^n \Delta q_n\right)$$

$$q_m = \frac{M_t}{2A} - \frac{1}{A}\sum_1^{N-1}\left(A_n \sum_1^n \Delta q_n\right) \qquad (3{:}26)$$

Equations (3:25) and (3:26) may be represented in the tabular form shown by table 3–6. Equations (3:25) and (3:26) and table 3–6 are directly applicable to stiffened-*D*-nose type wings if the sign conventions and numbering shown in figure 3–20 are employed.

3.13541. *Two cell—general method.* The following method is an extension of the general method for single cells. The two-cell structure is statically indeterminate since the division of the total torque between the two cells depends upon their relative torsional stiffnesses. A shear flow in an element of the front cell and a flow in an element of the rear cell are therefore considered as unknowns, and the flows in the remaining elements expressed in terms of these two unknowns. One independent equation is obtained from $\sum$ torques=*O*, and another from the fact that the twist of the front cell equals the twist of the rear cell. The two unknown shear flows are obtained by simultaneous solution of these equations, and the reremaining flows computed by successively adding or subtracting the shear flows absorbed by the bending elements. The notation is illustrated in figure 3–21, where the following symbols are additional to those described in section 3.13540 for single cells.

q_m=shear flow in main web.

q_f=shear flow in first shear element (numbered 0) of front web.

$s_0, s_2, s_3, \ldots s_n$, are lengths of shear elements.

$c_0, c_1, c_2, \ldots c_n$, are elastic constants of the shear elements.

$c = \frac{s}{t_e}$, where t_e is the effective thickness of the shear element, that is: $t_e = t_1 \times \frac{G_1}{G}$, where t_1 is the geo-

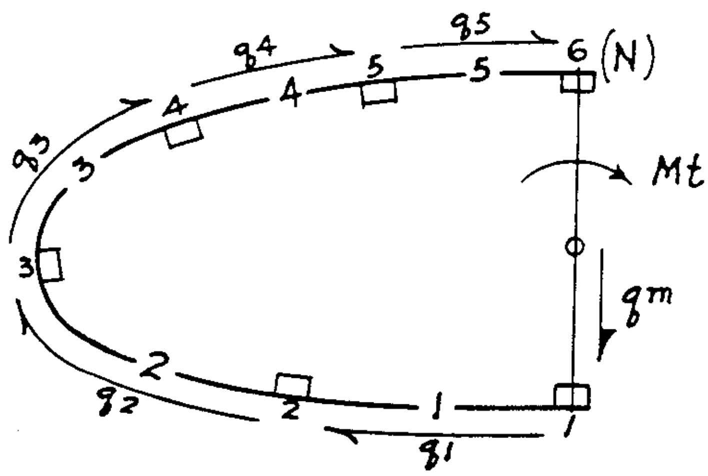


Figure 3–20. Conventions for stiffened-D nose section.

metrical thickness of the element, G_1, the shear modulus of the element, and G the shear modulus of the material considered basic for the section (sec. 2.52). If a particular element is expected to buckle appreciably in shear, the value of G_1 should be reduced accordingly.

A_F=enclosed area of front cell.

A_R=enclosed area of rear cell.

$A = A_F + A_R$.

$R = \frac{A_F}{A_R}$

$\sum_1^n$=summation of quantities for elements 1 through *n*, where $n=1, 2, 3$, etc.

N=number of upper flange of main web.

M=number of lower flange of main web.

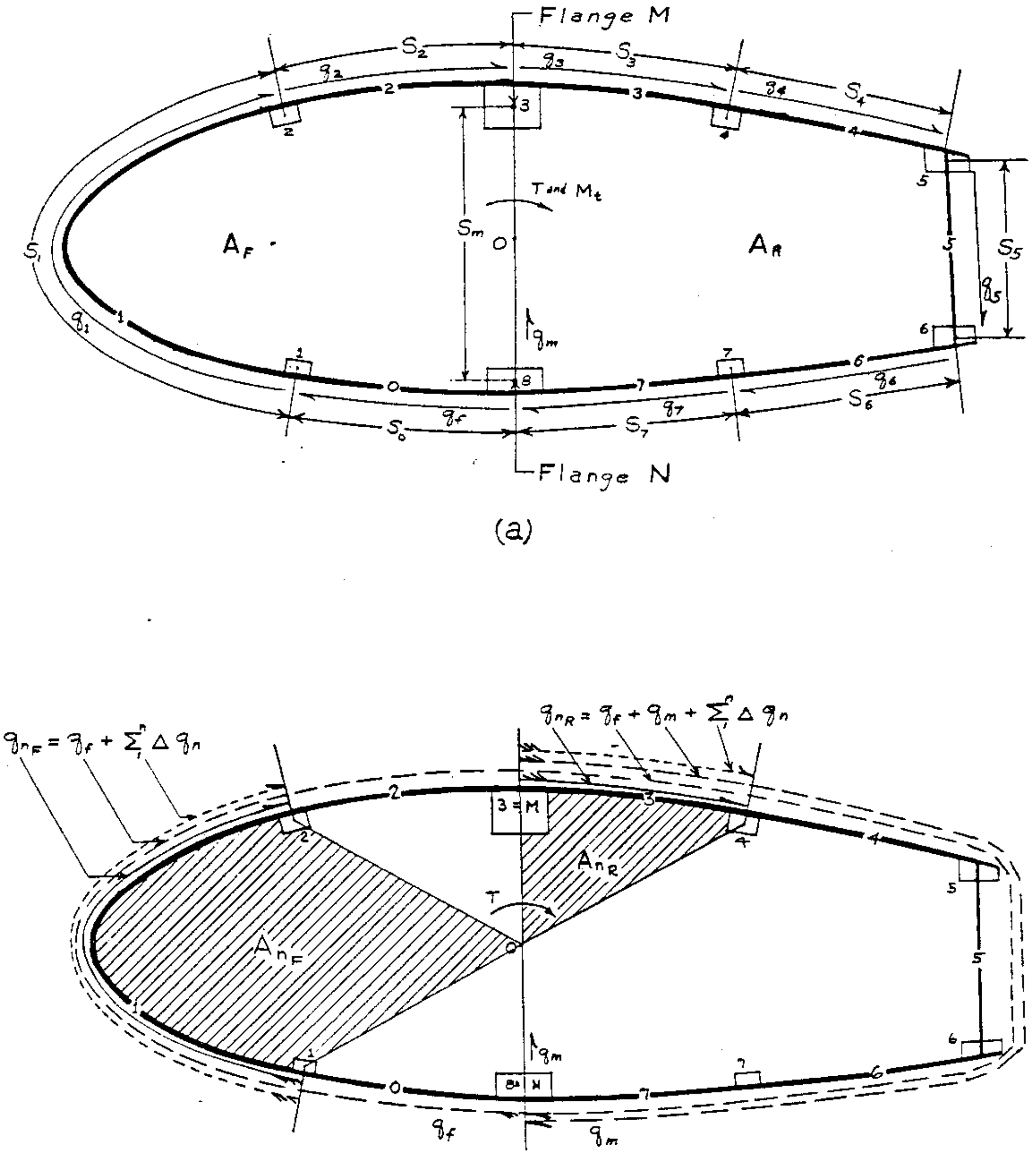


Figure 3-21. Rational shear flow; two-cell wing.

Subscripts $_F$ and $_R$ refer to front and rear cells, respectively.

Shear flow in any shear element (see derivation for single cell).

Front cell:

$$q_{nF}=q_f+\sum_1^n \Delta q_n \qquad (3:27)$$

Rear cell:

$$q_{nR}=q_f+q_m+\sum_1^n \Delta q_n \qquad (3:28)$$

Equation 3:27 and 3:28 are represented graphically on diagram (b) of figure 3-21.

Torque about point 0.

$\frac{T}{2}=\sum A_n q_n$, which from diagram (b), figure 3-21,

$$=q\cdot A+q_m A_R+\sum_1^{N-1}\left(A_n\sum_1^n \Delta q_n\right)$$

$$=\frac{M_t}{2}\text{ (External torque)}$$

$$q_f A+q_m A_R=\frac{M_t}{2}-\sum_1^{N-1}\left(A_n\sum_1^n\Delta q_n\right)$$

$$q_f+\frac{A_R}{A}q_m=\frac{M_t}{2A}-\frac{1}{A}\sum_1^{N-1}\left(A_n\sum_1^n\Delta q_n\right) \qquad (3:29)$$

which may be written in the form

$$X_2 q_f + Y_2 q_m = Z_2 \qquad (3:30)$$

Where X_2, Y_2, and Z_2 are numerical constants, and q_f and q_m are unknown quantities.

Consistent deformations. The angle of twist θ is the same for front and rear cells.

Therefore,

$$\theta = \frac{1}{2A_{cell}} \sum q \frac{s}{Gt_e} \qquad (3:31)$$

for each cell, where the summation is taken entirely around the cell (fig. 3–15).

$$2G\theta = \frac{1}{A_{cell}} \sum qc \qquad (3:32)$$

G is taken out of the summation sign as a constant, since all elements are reduced to a common basic shear modulus by use of effective thicknesses. Therefore:

$$\frac{1}{A_F} \sum^{F} qc = \frac{1}{A_R} \sum^{R} qc$$

$\sum^{F} q_c = R \sum^{R} qc$, which is from diagram (b) of figure 3–21:

$$q_f \sum_{0}^{M-1} c_n + \sum_{1}^{M-1} \left(c_n \sum_{1}^{n} \Delta q_n \right) - q_m c_m = R q_f \sum_{M}^{N-1} c_n +$$

$$R \sum_{M}^{N-1} \left(c_n \sum_{1}^{n} \Delta q_n \right) + R q_m \sum_{M}^{N-1} c_n + R q_m c_m \qquad (3:33)$$

or

$$q_f \left(\sum_{0}^{M-1} c_n - R \sum_{M}^{N-1} c_n \right) - q_m \left(c_m + R c_m + R \sum_{M}^{N-1} c_n \right) =$$

$$R \sum_{M}^{N-1} \left(c_n \sum_{1}^{n} \Delta q_n \right) - \sum_{1}^{M-1} \left(c_n \sum_{1}^{n} \Delta q_n \right) \qquad (3:34)$$

which may be written in the form

$$X_1 q_f + Y_1 q_m = Z_1 \qquad (3:35)$$

The quantities q_f and q_m are then determined by solving equation (3:30) and (3:35) simultaneously. The summation terms in these equations may be computed in a form similar to table 3–7.

3.13542. *Two-cell, four-flange wing.* If it is assumed for this type of wing (fig. 3–22) that the skin and web members carry shear only, the general equations given in section 3.13541 can be written in the following form:

$$q_f + \frac{A_R}{A} q_m = \frac{M_t}{2A} - \frac{1}{A} \sum_{1}^{3} \left(A_n \sum_{1}^{n} \Delta q_n \right) \qquad (3:36)$$

$$q_f \left(c_o - R \sum_{1}^{3} c_n \right) - q_m \left(c_m + R c_m + R \sum_{1}^{3} c_n \right) =$$

$$R \sum_{1}^{3} \left(c_n \sum_{1}^{n} \Delta q_n \right) \qquad (3:37)$$

These equations may be expressed as follows:

$$X_2 q_f + Y_2 q_m = Z_2 \qquad (3:38)$$

$$X_1 q_f + Y_1 q_m = Z_1 \qquad (3:39)$$

where:

$$X_2 = 1 \qquad (3:40)$$

$$Y_2 = \frac{A_R}{A} \qquad (3:41)$$

$$Z_2 = \frac{M_t}{2A} - \frac{1}{A} \sum_{1}^{3} \left(A_n \sum_{1}^{n} \Delta q_n \right) \qquad (3:42)$$

$$X_1 = c_o - R \sum_{1}^{3} c_n \qquad (3:43)$$

$$Y_1 = -\left(c_m + R c_m + R \sum_{1}^{3} c_n \right) \qquad (3:44)$$

$$Z_1 = R \sum_{1}^{3} \left(c_n \sum_{1}^{n} \Delta q_n \right) \qquad (3:45)$$

Then, solving (3:38) and (3:39) simultaneously,

$$q_f = \frac{\dfrac{Z_1}{Y_1} - \dfrac{Z_2}{Y_2}}{\dfrac{X_1}{Y_1} - \dfrac{X_2}{Y_2}} \qquad (3:46)$$

$$q_m = \frac{\dfrac{Z_1}{X_1} - \dfrac{Z_2}{X_2}}{\dfrac{Y_1}{X_1} - \dfrac{Y_2}{X_2}} \qquad (3:47)$$

Example: Referring to figure 3–22 and table 3–7, it is assumed that the following data have been previously determined or given:

$S_b' = +100{,}000$ pounds.

$S_c' = -10{,}000$ pounds.

$M_t = -500{,}000$ inch-pounds.

Δq values, as listed in table 3–7 (determined by sec. 3.1351 (2))

S, t_e, and A values as listed in table 3–7.

$A_F = 2{,}288$ square inches.

$A_R = 2{,}912$ square inches.

$A = 5{,}200$ square inches.

Shear flow values, obtained by substitution of

Table 3-7. Shear-flow computations for typical two-cell, four-flange wing section

	(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)
	n	S	t_e ①	c_n	Δq_n	$\sum_1^n \Delta q_n$	$c_n \sum_1^n \Delta q_n$	A_n	$A_n \sum_1^n \Delta q_n$
				(2)/(3)		$\sum_1^n (5)$	(4) × (6)		(6) × (8)
FRONT CELL	0	157.0	.250	628.0				2288.0	
REAR CELL	1	85.5	.250	341.8	-2383.7	-2383.7	-814,749	839.0	-1,999,924
	2	26.0	.375	69.3	-459.1	-2842.8	-197,006	1033.5	-2,938,034
	3	82.3	.250	329.2	+584.9	-2257.9	-743,301	1040.0	-2,348,216
				$\sum_1^3 c_n = 740.3$			$\sum_1^3 (c_n \sum_1^n \Delta q_n) = -1,755,056$	$A_R = \sum_1^3 A_n$	$\sum_1^3 (A_n \sum_1^n \Delta q_n) = -7,286,174$
MIDDLE WEB	m	36.9	1.000	36.9					

NOTE:

①-Shear Modulus of all shear elements assumed to be the same in this case.

$$A_F = 2288$$
$$A_R = 2912.5$$
$$R = A_F/A_R = .7855$$

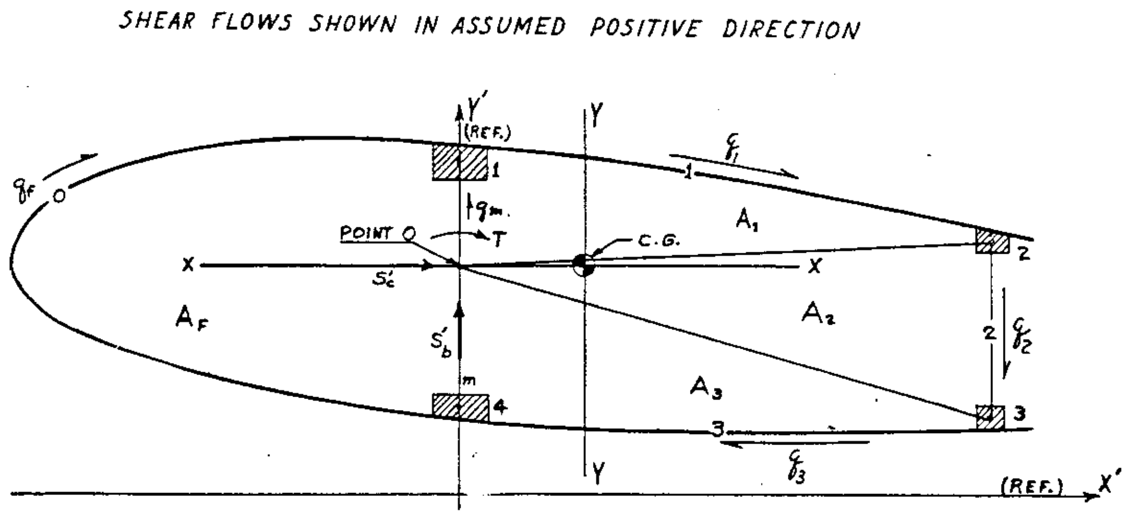


Figure 3-22. Typical two-cell, four flange wing section.

the summations from table 3-7 in equations (3:40) to (3:47) are as follows:

$q_f = 154.3$ pounds per inch.

$q_m = 2,140.4$ pounds per inch.

The remaining shear flow values are then determined from equations (3:27) and (3:28):

$q_1 = -89$ pounds per inch.

$q_2 = -548.1$ pounds per inch.

$q_t = 36.8$ pounds per inch.

3.13543. *Shear centers.* For some purposes, it is desirable to determine the shear center of a wing section. As derived herein, the shear center is defined as the point on a wing section at which the application of a shear load will produce no twist in a differential length of the structure beyond the section. A point so determined is a true shear center for the wing as a whole only if the wing is of constant section throughout the span, or tapers in a manner so that all sections are geometrically similar.

In the following formulas, symbols not expressly defined are the same as in sections 3.13540 and 3.13541.

(*a*) *Single cell.* Assume that a V load of value P has been applied to the section and the values of Δq for the bending elements computed according to section 3.1351:

$$\text{Twist} = \theta = 0 = \frac{1}{2AG} \sum qc \qquad (3:48)$$

$\sum qc$ is found by inspection of figure 3-19 and equation (3:25), resulting in:

$$\theta = 0 = \frac{1}{2AG}\left[q_m c_m + q_m \sum_1^{N-1} c_n + \sum_1^{N-1}\left(c_n \sum_1^n \Delta q_n\right)\right] \qquad (3:49)$$

Equation (3:49) is solved for the value of q_m which will produce no twist:

$$q_m = -\frac{\sum_1^{N-1}\left(c_n \sum_1^n \Delta q_n\right)}{c_m + \sum_1^{N-1} c_n} \qquad (3:50)$$

Let x = the horizontal distance from the origin O to the load P for the condition of no twist. (That is, x = distance to shear center). Since $Px = M_t$, x may be determined from equation (3:26), as follows:

$$q_m = \frac{P_x}{2A} - \frac{1}{A} \sum_1^{N-1}\left(A_n \sum_1^n \Delta q_n\right) \qquad (3:51)$$

$$x = \frac{2A}{P}\left[q_m + \frac{1}{A} \sum_1^{N-1}\left(A_n \sum_1^n \Delta q_n\right)\right] \qquad (3:52)$$

where q_m is from equation (3:50) and other terms are computed as in table 3-6.

The vertical location of the shear center may be determined, if desired, by applying a drag load and proceeding as has been shown.

(*b*) *Two-cell.* It is assumed that a V load of value P has been applied at the shear center which is at an unknown horizontal distance x from the origin O, and that Δq values corresponding to load P have been computed for the bending elements. Since the twist of both cells is zero:

$$\theta_F = 0 = \frac{1}{A_F} \sum^F qc \qquad (3:53)$$

$$\theta_R = 0 = \frac{1}{A_R} \sum^R qc \qquad (3:54)$$

substituting for $\sum qc$, according to section 3.13541:

$$\theta_F = 0 = q_f \sum_0^{M-1} c_n - q_m c_m + \sum_1^{M-1}\left(c_n \sum_1^n \Delta q_n\right) \quad (3:55)$$

$$\theta_R = 0 = q_f \sum_M^{N-1} c_n + q_m\left(\sum_M^{N-1} c_n + c_m\right) + \sum_M^{N-1}\left(c_n \sum_1^n \Delta q_n\right) \quad (3:56)$$

Solving equations (3:55) and (3:56) simultaneously for q_f and q_m will give the values necessary for the condition of no twist. Since Px is the torsional moment about the origin O, this moment and the value of x may be found from the derivation of equation (3:29), as follows:

$$\frac{M_t}{2} = \frac{Px}{2} = q_f A + q_m A_R + \sum_1^{M-1}\left(A_n \sum_1^n \Delta q_n\right) \quad (3:57)$$

where the values of q_f and q_m are from equations (3:55) and 3:56). The vertical location of the shear center may be determined, if desired, by applying a drag load and proceeding as in the foregoing.

3.136. *Ribs and Bulkheads.*

3.1360. *Normal ribs.* Normal ribs (those subjected primarily to airloads), in a shell wing, receive the airloads from adjacent skin and stiffeners and redistribute them to the various shear elements of the wing section. The strength of such ribs is always proven by strength tests, but a picture of the stress distribution is useful in rib design and in devising suitable test set-ups. The required airloads, distributed in accordance with the airfoil chordwise pressure distribution, may be considered as the applied loads on the rib, and the shear flows applied by the rib to the various wing section shear elements, oppositely directed, as the reactions. Such shear flows may be determined by performing computations similar to those for the shear flow distribution (using the section method, sec. 3.1351 (2)), after resolving the airloads into resultant forces and a moment, at a convenient reference point.

These conditions may be simulated in a test by constructing a short spanwise section of the wing in which the test rib at one end forms the loading bulkhead, while a bulkhead at the opposite end supports the whole section. The spanwise length, and the attachment of stiffeners and skin to the support bulkhead, should be such that the rib loads are not transmitted directly to the support bulkhead by these elements acting as cantilever beams.

Normal ribs are also subject to a variety of secondary loads, for example: Loads resulting from their function as compression elements when the skin buckles into diagonal-tension fields due to shear; and loads resulting from the axial forces in stiffeners and skin while the wing is deflected in bending.

3.13600. *Rib-crushing loads.* Compressive forces in the upper surface material of the wing, while it is curved upward by bending deflections, produce downward acting loads in the ribs, while the tensile forces in the lower surface produce upward loads, thus subjecting the ribs to compression or crushing in the vertical direction. Where an appreciable portion of the wing-bending material is distributed in the form of skin and stiffeners remote from the beam webs, the rib-crushing loads should be investigated by methods such as reference 3–10 or the following:

$$w = \frac{PL}{R} = \frac{PLM}{EI} \quad (3:58)$$

where:

w=vertical crushing load on rib flange, in pounds per inch of chord.

P=spanwise axial load: in wing surface material due to bending, in pounds per inch of chord, at given point on wing section.

L=rib spacing, inch.

R=radius of curvature of wing due to bending.

M=bending moment on wing section. (M_b from section 3.1331 may be used as an approximation.)

I=moment of inertia of wing section. (I_x from table 3–5 may be used as an approximation.)

E=basic modulus of elasticity used in computing section properties (sec. 3.1330).

3.1361. *Bulkhead ribs.* Bulkhead ribs are described as those that distribute loads of appreciable magnitude, other than air loads, to the wing-section shear elements; for example, fuselage, landing gear, and fuel tank reactions. Such loads, as well as the airloads, may be considered as external loads applied to the rib, and the shear flows applied by the rib to the shear elements, oppositely directed, as the reactions. Here, however, one or more of the conditions required by the shear-flow theory (sec. 3.135) will generally be violated. For example, a larger amount of shear may be absorbed by the elements nearest a concentrated load, depending on their rigidity relative to that of the bulkhead. Conservative overlapping assumptions should therefore be made.

Bulkhead ribs may also perform the function of redistributing shear among the shear elements of a wing wherever some of these elements are discontinued or bending elements redistributed. The shear flows from the outboard wing section may then be considered as the applied loads on the rib, and the shear flows applied to the inboard section, oppositely directed, as the reactions.

Likewise, at a rib where any wing element carrying an appreciable axial load changes direction, the axial loads in the inboard and outboard portions of such an element should be resolved into components parallel and perpendicular to the plane of the rib. The resultant of the components in the plane of the rib may then be considered as a load applied to the rib, with reactions supplied by the wing-section shear elements as described previously.

As a result of the bulkhead analysis, it may be necessary to revise the shear distribution determined in the general shear analysis (sec. 3.135) for local conditions.

3.137. *Miscellaneous structural problems.*

3.1370. *Additional bending and shear stresses due to torsion.* The corner flanges of a box beam are theoretically free from axial (bending) stresses under a pure torque loading, if the cross sections are free to "warp" as the box twists. However, in a shell wing where more than one beam is continuous through the fuselage either directly or through an equivalent structure, bending stresses will be induced in the corner flanges since the opposing action of the opposite wing will restrain the root sections from warping. Additional shear in the short sides of the box is also induced at restrained sections.

In wings not subjected to unusual torque loads and in which the torque cells are continuous and enclose a large part of the sectional area of a reasonably thick wing, the bending stresses at the root due to torsion should be small compared to the total bending stresses for the loading conditions producing maximum bending in the wing.

Analytical methods for computing the bending stress due to torsion in various types of box wings are described in references 3–8 and 3–12. Where the shear rigidity of one wall of a box wing is greatly reduced by a cut-out, the wing torsion should be assumed to be carried as differential bending in the spars in the region of the cut-out. Rational solution of the general case is given in reference 3–6.

Wings in which the torsional stiffness of the torque cells is relatively small because of the small enclosed area or because of many large cut-outs may be conservatively designed as independent spar wings. The effect of the torque cell in relieving the critically loaded spar by transferring part of the load to the other spars may, however, be estimated according to reference 3–7.

3.1371. *General instability.* Reference to section 3.1381 shows that the column length of spanwise stiffeners is generally taken equal to the rib spacing. Such an assumption is valid only when the ribs act as rigid lateral restraints for the stiffeners at the points of intersection. If the ribs lack rigidity in their own planes, allowing the stiffeners to deflect laterally, the axial compressive loads in the stiffeners tend to further increase such deflections because of the resulting eccentricities. If the rib rigidity is too low relative to the axial stiffener (or skin) compressive loads, a state of equilibrium will not be reached, and the ribs and stiffeners will collapse simultaneously. In conventional wings with full depth ribs, the condition described above, known as general instability usually need not be considered. If shallow ribs (at tank bays and wheel wells) or truss-type ribs having shallow flanges are used in wings where a large part of the bending compressive loads are carried in surface material remote from the wing beams, analysis or tests for this condition should be made (ref. 3–14).

3.138. *Strength Determination.* The analytical determination of the strength of the structure is based on a comparison between the computed internal stresses, and the allowable stresses obtained by static test or calculated from the material properties by methods such as those of chapter 2. In order that the computed margins of safety so obtained may represent the strength of the structure with respect to the specified external loads, as accurately as possible, all conditions and assumptions on which both the internal and allowable stresses are based should be reviewed, and any necessary adjustments or allowances made, prior to the final comparison showing the margins of safety. Such allowances may be made by arbitrarily increasing the originally computed internal stresses or decreasing the allowable stresses, in the light of the review.

Some of the factors to be considered in the strength determination are discussed under the following subsections.

3.1380. *Buckling in skin.* For a structure in which the major portion of the compressive loads due to bending are intended to be resisted by the skin, with the shape being maintained by comparatively light reinforcing structure, the critical buckling and ultimate stresses for the skin, whichever is lower, should be considered as the allowable stress. When buckling does not occur, the ultimate allowable stresses may be computed by the methods of sections 2.60 and 2.61. The criteria of sections 2.70, and 2.80 may be used as guides in predicting the occurrence or nonoccurrence of buckling, but the strength of such structures should be substantiated by static tests of the complete structure, or of a closely similar structure, to ultimate load, because of the uncertainties of buckling phenomena.

For structures in which the supporting and stiffening members are capable of withstanding a major portion of the compressive loads, buckling of the skin does not necessarily result in failure, as discussed in the following subsections on stiffened panels and shear elements. Sharply curved skin panels have much higher critical buckling stresses than flat panels of the same dimensions, but failure in curved panels usually occurs immediately after buckling begins.

3.1381. *Compression elements.* Where secondary stresses, such as those described in sections 3.1330 (5), 3.134, and 3.1370 have not already been taken into account, a reasonable increase in internal stresses should be assumed for critical elements affected thereby. Although wood will yield slightly in compression, tending to relieve the highly stressed fibers, elements which have undergone some crushing in compression may fail at unexpectedly low tensile stresses when the load is reversed.

When light spanwise stiffeners are used to reduce the size of the skin panels rather than to resist the wing bending loads, they need not be designed to withstand the stresses which would be assigned to them as isolated structural elements by the bending theory, provided that such stiffeners are designed to accommodate themselves to the spanwise shortening of the compression side of the wing without failing. At locations remote from the spars, this can be accomplished by making the stiffeners sufficiently flexible so that they can bow between the ribs without failing. Such stiffeners may tend to separate from the skin, however, unless special precautions are taken. At locations adjacent to highly stressed spar flanges this accommodation may be obtained by using a cross section and material such that local crippling and crushing failure will not occur.

3.1382. *Stiffened panels.* In structures where the skin is expected to buckle below ultimate load and the reinforcing structure is designed accordingly, the allowable compressive stresses may be obtained from section 2.77 or from tests on stiffened panels.

(*a*) *Effective widths.* In both the allowable and the internal stress computations, an effective width strip of skin adjacent to each stringer is assumed fully effective in compression. The width is often selected arbitrarily, and it is sometimes assumed that the value selected makes little difference so long as the value used in the section-properties computations is consistent with that used in computing allowable stresses from the total load supported by a test panel. This assumption would be true if the upper and lower bending material of the wing consisted only of two symmetrical panels (with the same effective widths in tension as compression) but it may lead to some error if the bending material is not structurally symmetrical and the usual methods of computing section properties are used. Therefore, for structures in which the skin carries a considerable portion of the bending load, the effective widths should be determined as accurately as possible, either by theoretical methods, such as those of section 2.72 and 2.774 or by accurate strain-gage measurements on the test panels. The effective width, $2w$, of plywood panels, is usually expressed as a strip that is considered to act at a stress corresponding to that of the unbuckled plywood at the same *deformation* as the stiffener.

The effective width of metal panels is usually expressed as a strip acting at the same *stress* as the stiffener. The basis for the effective widths indicated in a particular analysis should, therefore, be clearly stated.

(*b*) *Allowable compressive stresses.* In determining the allowable compressive stress, the various possible modes of failure discussed in section 2.775 should be considered. When the allowable stress is computed by section 2.72, the stiffener plus effective width of skin is considered as one composite element having an effective modulus of elasticity E'. This procedure was arranged to facilitate checking the stress in any ply or fiber of either plywood or stiffener. Such a composite element may be considered as one item in the section-properties computations (sec. 3.1330), where e_1 will equal $\frac{E'}{E}$ basic.

The computed internal stress, f, for comparison with the allowable will then be: $f=f'\times e$, where f' is the fictitious basic-modulus stress obtained by the bending formulas in section 3.1331, and e is the total element effectiveness factor in accordance with section 3.1330 (5).

When the ribs are sufficiently rigid in their own planes (sec. 3.1371) the column length of the stiffened panels is taken as equal to the rib spacing. In regard to the column-fixity coefficient to be used in conjunction with this column length, it is noted that typical structures show a general tendency to bow inward in the bays between ribs, but a few bays will tend to bow outward. Where one bay bows in and the next out, a fixity of approximately $c=1.0$ is developed, depending on the rotational fixity furnished by the fibs and the degree of buckling and plate or curvature effect of the skin. A value of $c=1.5$ may be assumed if the stringers are fixed to ribs having appreciable bending stiffness in a vertical plane parallel to the stringers. Higher values should not be used in design unless substantiated by tests on a complete structure.

In flat-ended-panel tests, a value of $c=3.0$ or more is usually developed. The results of such tests must therefore be corrected to the fixity values used in the design of the structure.

(c) *Combined stresses.* A convenient method of considering the effects of combined compression and shear in stiffened panels is the stress ratio or interaction curve method, that is, $R_c^m+R_s^n=1.0$, where R_c is based on the allowable compressive stress discussed in paragraph (b), and R_s is based on the strength of the panel in pure shear.

The exponents m and n may be assumed equal to 2.0 for panels which are substantially flat, but not more than 1.0 for sharply curved panels, such as in D-nose spars, unless tests are made under combined loads to determine points on the interaction curve. For D-nose spars, tests to ultimate load should be made. A portion of the spar of sufficient length to eliminate end effects, may be used in such tests.

3.1383. *Tension elements.* Tension elements of wood yield very little, compared to metals, before reaching their ultimate strength (sec. 2:16). Unaccounted-for secondary stresses or unconservative assumptions in the stress analysis are therefore likely to cause failures. Since the plywood skin, stiffeners, and spar flanges on the tension side of a wing may not reach their ultimate strengths at the same time, the stresses in each element should be determined and compared with the corresponding allowables. For plywood having the face grain parallel or perpendicular to the spanwise direction, the modulus of elasticity for use in determining section properties and internal stresses may be obtained from section 2.52, or table 2–13, and the allowable tensile stress from section 2.601, and table 2–13. For plywood having the face grain at an angle to the spanwise direction, the spanwise modulus of elasticity may be obtained from section 2.56.

The allowable tensile stress for such 45° plywood may be obtained from section 2.611 and table 2–13.

When the plywood on the tension side does not buckle due to shear, which is usually the case on a wing (sec. 2.73), the condition for failure under combined tension and shear may be determined by stress ratios in accordance with section 2.613.

3.1384. *Shear elements.* When the shear flow, q, has been determined, the internal shear stress is obtained by dividing q by the actual thickness of the element, even though an effective thickness based on relative moduli of rigidity was used in the shear distribution analysis. The allowable shear stress values given by section 2.73 are directly applicable to beam webs and allow for the effects of the beam bending stresses near the flanges.

These allowable shear stresses should also be applicable to substantially flat wing skin panels in the same range with respect to buckling. The ultimate strength of curved panels in shear must at present be obtained from tests on specific structures as described in section 3.1382 (*c*), since buckling usually precipitates failure.

3.2 Fixed Tail Surfaces

The procedures applicable for use in the stress analysis of fixed tail surfaces (fin and stabilizer) are analogous to those described in section 3.1 for the analysis of wings. The nature of the applied loads is necessarily similar in that the source is principally aerodynamic and the spanwise and chordwise distributions of the same are similar to those over wing surfaces. The loads resulting from inertia effects require a consideration similar to that employed in the analysis of wings. The dependence of the applicable type of analysis upon the structural arrangement of the material is also similar to that encountered with wings and this consideration is treated in section 3.1. The strength of the structure is determined by comparison of the calculated internal loads and stresses with the allowables which are obtained either from tests or from the information given in chapter 2. The determination of the strength of shell structures, including reinforced shells, is presented in detail in section 3.138.

3.3 Movable Control Surfaces

The movable control surfaces are ordinarily comprised of the ailerons, elevator, and rudder. The analysis of each of these surfaces is fundamentally the same basic problem. Each movable surface consists of an airfoil free to rotate about a hinge axis fixed on the supporting structure except as restrained by the control system at its attachment point (control horn). The essential structure is made up of the:

(1) *Airfoil surface* (fabric or plywood plating) upon which the air forces act and are transmitted through
(2) *Surface attachment means* (lacing, nails, or glue) to the
(3) *Ribs*. The ribs transfer the air loads through shear and bending to the
(4) *Main beam* and
(5) *Torque tubes*. The beam and torque tube are supported by the fixed surface structure at the
(6) *Hinges* where the transverse shear is transmitted to the fixed surface. The torque tube carries the torque resulting from the air loads and hinge support reactions to the
(7) *Horn*, where it is balanced by the control system reactions.

A satisfactory analysis should include a check of the plating material (fabric, plywood) under the imposed design pressure loading. Unit pressure loadings, consistent in magnitude with those encountered over deflected control surfaces should be considered in such a check. The strength of the surface attachments should be checked in combination with that of the surface material itself. The most satisfactory method of determining the strength of such structural items is by "blow-off tests" of panels representing the type of construction employed (simulating rib spacing, surface attachments) when subjected to test pressures representing the design loadings. Critical surface pressures are usually negative (tending to blow the surface outward).

Ribs may be considered as cantilever beams supported at the main beam or torque tube and supporting the pressure loading over the area extending approximately midway to adjacent ribs. Here again static tests of representative structures constitute the preferred basis for proof of satisfactory strength.

The main beam and torque tube should be checked under the shear, bending, and torsional loads resulting from the rib loadings, and the reactions at the hinge supports and the control horn. When the main beam or torque tube is continuous over three or more hinge supports, the deflection of the fixed surface or wing under flight loads should be taken into account by introducing suitable deflections of the supports into the three moment equations or by conservative overlapping assumptions. Irregularities and discontinuities of such structures are often encountered because of the cut-outs necessary for the control surface hinges. Care should be exercised to provide adequate strength and rigidity in way of such cutouts by means of proper reinforcing and by use of conservative assumptions both as to stresses developed and stresses allowed. This is especially necessary in wood structures because of the inherent inability of wood to equalize stress concentrations through considerable plastic deformation.

3.4 Fuselages

3.40. General. Most of the commonly used types of wood fuselage construction fall within one of the following:

(1) Four-longeron type.
(2) Reinforced shell (semimonocoque) type.
(3) Pure shell (monocoque) type.

Examples of these types are included in the sketches shown herein under the pertinent subheadings. A particular airplane fuselage need not necessarily be confined to one type of construction but may employ any applicable combination. For example, the stiffened-shell type may revert to the four-longeron type in way of large cut-outs such as cockpit openings, or bomb bays.

3.41. Four-Longeron Type. The treatment of the four-longeron type is somewhat analogous to that of the D-section and single-cell shells as described in section 3.13 with the additional simplification that results from the inherent symmetry of the typical fuselage section. In both, the material effective in bending is concentrated into a small number of locations and the section properties for use in a bending analysis may be calculated in the normal manner as based upon such an assumption. The plywood shell material will actually contribute in some indeterminate extent to the bending strength of such four-longeron-type sections as are illustrated in figures 3–23 and 3–24. However, it is probable that, on the compression side, this contribution will be limited to approximately that corresponding to the buckling load for the plywood panels as determined from the transverse frame spacing, panel thickness, species, arrangement of plies, and curvature according to the methods described in chapter 2. In this type of construction, the unit deformation corresponding to the maximum design stress in the longerons very probably exceeds by far that corresponding to the buckling stress of the adjacent plywood material and of that farther removed from the neutral axis. Also, without curvature and without longitudinal stringers between longerons and the smaller plywood panel expanses and greater buckling stresses resulting therefrom, the design shearing stress in the sides of the four-longeron-type section will also probably exceed the buckling values by a considerable amount.

Both of these tendencies lead to the conclusion that it is satisfactorily conservative to neglect the contribution of the plywood shell to the bending properties in cases where the buckling stresses of such shell material is considerably exceeded by the longeron stresses and shear web stresses calculated on the basis of zero contribution (fig. 3–23). In any event, the optimum contribution of the shell material that could be expected would be that corresponding, on the compression side, to the buckling stress of the panels and, on the tension side, full effectiveness. In this connection, the designer's attention is directed to the existing knowledge of the behavior of thin panels subsequent to buckling. With flat panels and panels of slight curvature (that is, those in which the contribution of curvature to the buckling load is not significant) a load approximately equal to the buckling load is maintained after buckling. With thick plates of considerable curvature (that is, those in which the contribution of curvature to the buckling load is appreciable) the load tends to drop off after buckling. In such panels, rupture is also much more likely to result at buckling. For these reasons, it is desirable that under the ultimate design loads, the stresses resulting in such a portion of a compression flange do not exceed the critical buckling stresses. On the tension side, the contribution of the plating should be taken as that corresponding to an equivalent area of the plywood flange in terms of the longeron material (fig. 3–24). For purposes of calculation, the equivalent effective area (or thickness) of the tension plywood flange would be equal to $t\times\frac{E_2}{E_1}$ where t=plywood thickness, E_1=modulus of elasticity of longeron material in a direction normal to the section, and E_2=modulus of elasticity of plywood material in a direction normal to the section. These definitions are different from those used in chapter 2.

In determining the optimum effectiveness of the compression plywood material, it is emphasized that the total load carried by the material would be approximately limited to the buckling load rather than being proportional to the total load upon the section. If it is considered permissible for the subject panel to buckle at the design load, the effective thickness for use in computing section properties may be taken as approximately $t\left(\frac{F_{cr}}{F}\right)\left(\frac{y_1}{y_2}\right)$, defined in figure 3–24. If it remains unbuckled the corresponding effective thickness may be taken as $t\left(\frac{E_2}{E_1}\right)$. The applicable pro-

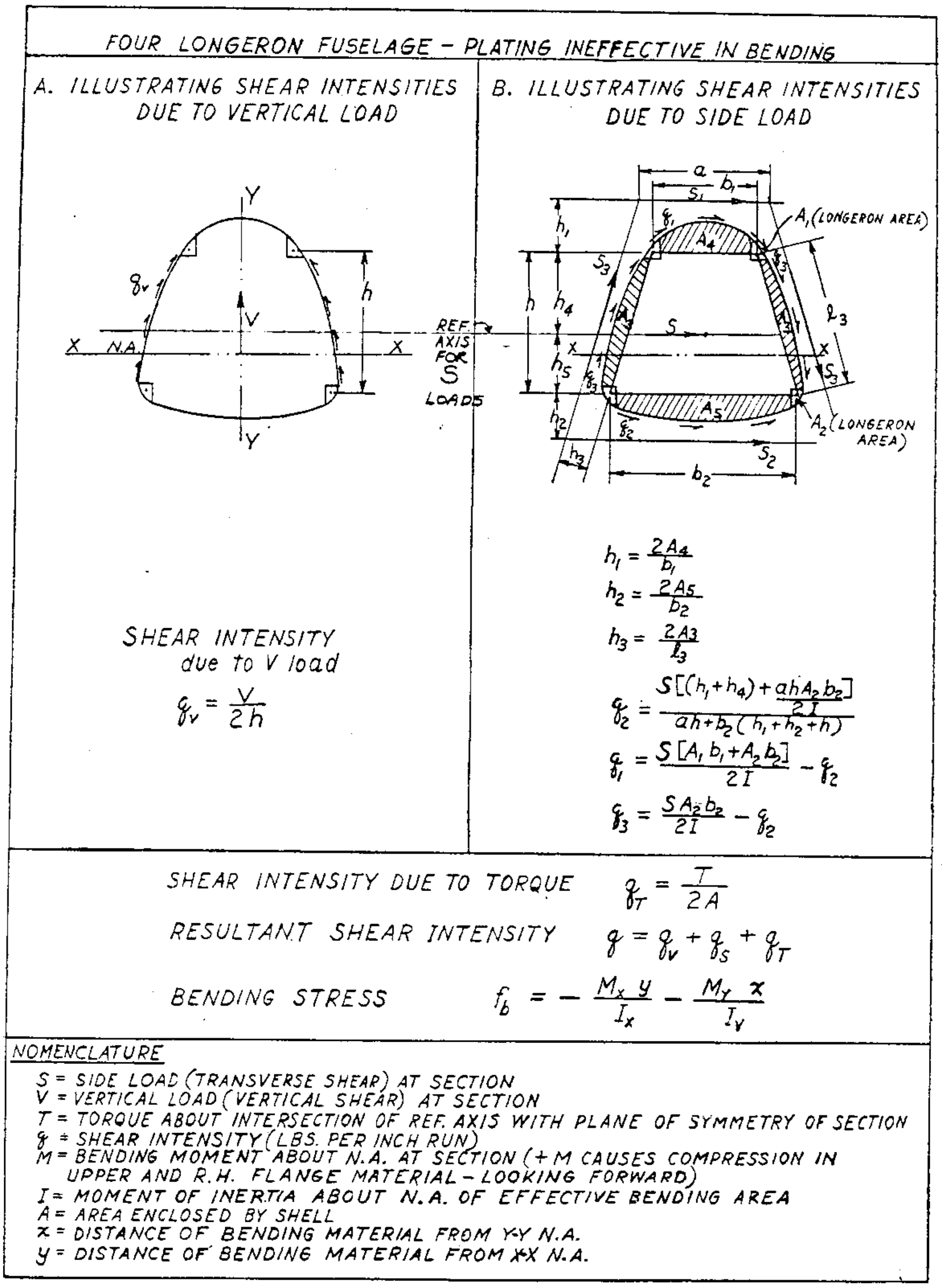
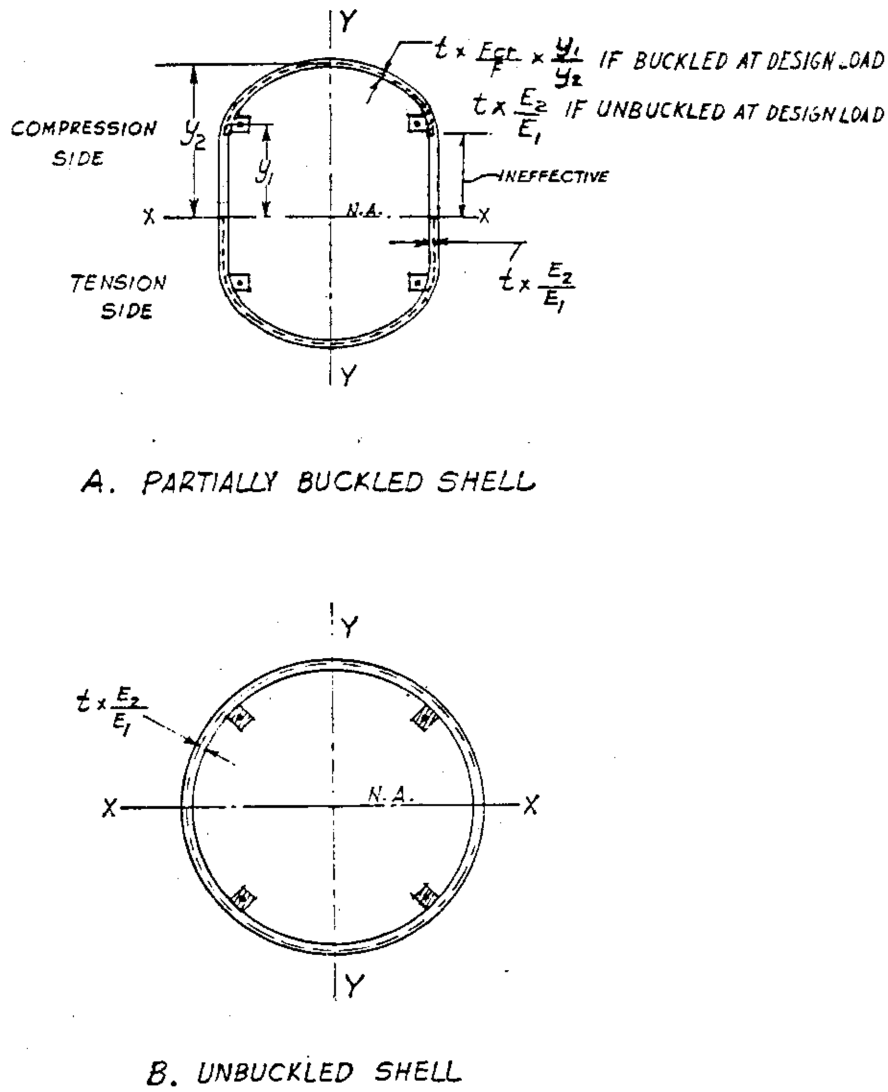


Figure 3–23. Four-longeron fuselage—plating ineffective in bending.

cedure must be checked by computing the actual stress in the plating and comparing it with $F_{c_{cr}}$. The resultant external applied loads on the section in question should be resolved into:

(1) Vertical shear (in plane of symmetry).
(2) Transverse shear (at reference point determined by fig. 3–23).
(3) Moment about each of the principal section axes.
(4) Torque about reference axis (for example, the intersection with the plane of symmetry of the transverse reference axis defined by fig. 3–23).

The plywood panels (sides, top, and bottom)



F_{cr} = PANEL BUCKLING STRESS

F = LONGERON ALLOWABLE STRESS

E_2 = MODULUS OF ELASTICITY OF PLYWOOD NORMAL TO SECTION

E_1 = MODULUS OF ELASTICITY OF LONGERON NORMAL TO SECTION

Figure 3–24. Four-longeron fuselage—plating effective in bending.

can be considered to carry the shear upon the section, both that due to the vertical and transverse loads and that resulting from torsion. When the flange material is concentrated in the longerons, the shear intensity (pounds per inch run) can be considered constant between adjacent flanges. The shear intensity, and thus the shear stress, may then be determined by figure 3–23 without the use of the shear center. Such center may be determined, if desired, by the methods of reference 3–11. Calculations made in connection with the application of the thin-shell theory, developed primarily for use with isotropic metal materials, should be modified to account for variations in the modulus of rigidity (G) for the various wood panels as affected by wood species, direction of grain, relative thickness and arrangement of plies, according to the methods described in chapter 2.

If the shell thickness, curvature, and frame spacing are such that the buckling stresses will not be exceeded under conditions of maximum loading, the section properties may be calculated using the full shell area as modified to correspond to equivalent longeron material, that is, the proportionate amount of effective shell material, in terms of longeron material, is equal to $\frac{E_2}{E_1}$ as previously described. When the section properties are thus calculated on the basis of longeron material, the bending stress in the longerons is determined in the usual manner.

$$f_1 = \frac{My_1}{I} \tag{3:59}$$

where y_1 is the distance of the longeron material from the neutral axis. The bending stress in the plywood material, however, is determined as

$$f_2 = \frac{My_2}{I} \times \frac{E_2}{E_1}. \tag{3:60}$$

where y_2 is the distance of the subject material from the neutral axis.

The possible variety of assumptions made to facilitate analysis can be considerable and will, to a large extent, be determined by the individual details of the problem together with the designer's experience, judgment, and discretion. An adequate supplementary static-test program is required, and it is also essential that the assumptions used in converting the test data into allowable loads and stresses be duplicated in the stress analysis of the flight article.

3.42. REINFORCED-SHELL TYPE. This type of construction is very broad in nature and covers the field extending from the longeron type with large longerons and thin shell to the type approaching the pure shell, that is, small longitudinals and thick shell.

3.421. *Stressed-skin fuselages.* Stressed-skin fuselages usually are structures of the reinforced-shell, single-cell type, and the basic methods of wing analysis, as described in section 3.13 generally can be applied directly to the analysis of such fuselage structures. Due to variations of the type of loading and certain other structural problems, however, it is considered advisable to review the fuselage analysis problem as a separate subject.

Unless a fuselage of this nature conforms closely to a previously-constructed type, the strength of which has been determined by test, a stress analysis is not considered as a sufficiently accurate means of determining its strength. The stress analysis should be supplemented by pertinent test data. Whenever possible, it is desirable to test the entire fuselage for bending and torsion, but tests of certain component parts may be acceptable in conjunction with a stress analysis.

3.422. *Computation of bending stresses.* Prior to computing the bending stresses, it is necessary to compute the fuselage-section properties. As was previously recommended in section 3.13, it is considered advisable to make a sketch of the fuselage section considered. This sketch should indicate all of the material assumed to be effective. Figure 3–25 is a sketch of a fuselage cross section of the subject type.

On the tension side of the fuselage the skin material may be assumed to be acting as discussed in the following, while, on the compression side, only the effective width of skin (section 3.1382) adjacent to the stiffener should be assumed to be acting. In general, the modulus of elasticity of the plywood plating will differ from that of the stiffener material. Account of this fact must be taken in calculating the section properties. This may be done by converting the actual area of the plating on the tension side into that of equivalent stiffener material, either in terms of equivalent thickness or equivalent widths—the latter being somewhat analogous to the effective width as used on the compression side. The geometrical shape of the section contour together with the arrangement and spacing of stiffener material will dictate which method of treatment is analytically simpler or more accurate. The proportionate effectiveness of the plating material in tension may be taken as $\frac{E_2}{E_1}$ as described previously under section 3.42.

Proper account for wood species, plywood grain attitude and arrangement, and veneer thicknesses should be taken into account according to the procedures described under section 2.77. The determination of bending stresses by means of the $\frac{My}{I}$ formula implies the assumption of plane sections remaining plane sections. Hence, the calculated stresses in the plating material, as based upon section properties determined by conversion of plating material into equivalent stiffener material, must also be modified in the ratio $\frac{E_2}{E_1}$. The resulting corrected stresses in the plating must be compared with the allowable tensile stresses in the plating material as described in section 3.1383. Such a check should always be made of plywood material adjacent to highly stressed stiffener material, even where the contribution of such plywood material has been completely neglected in

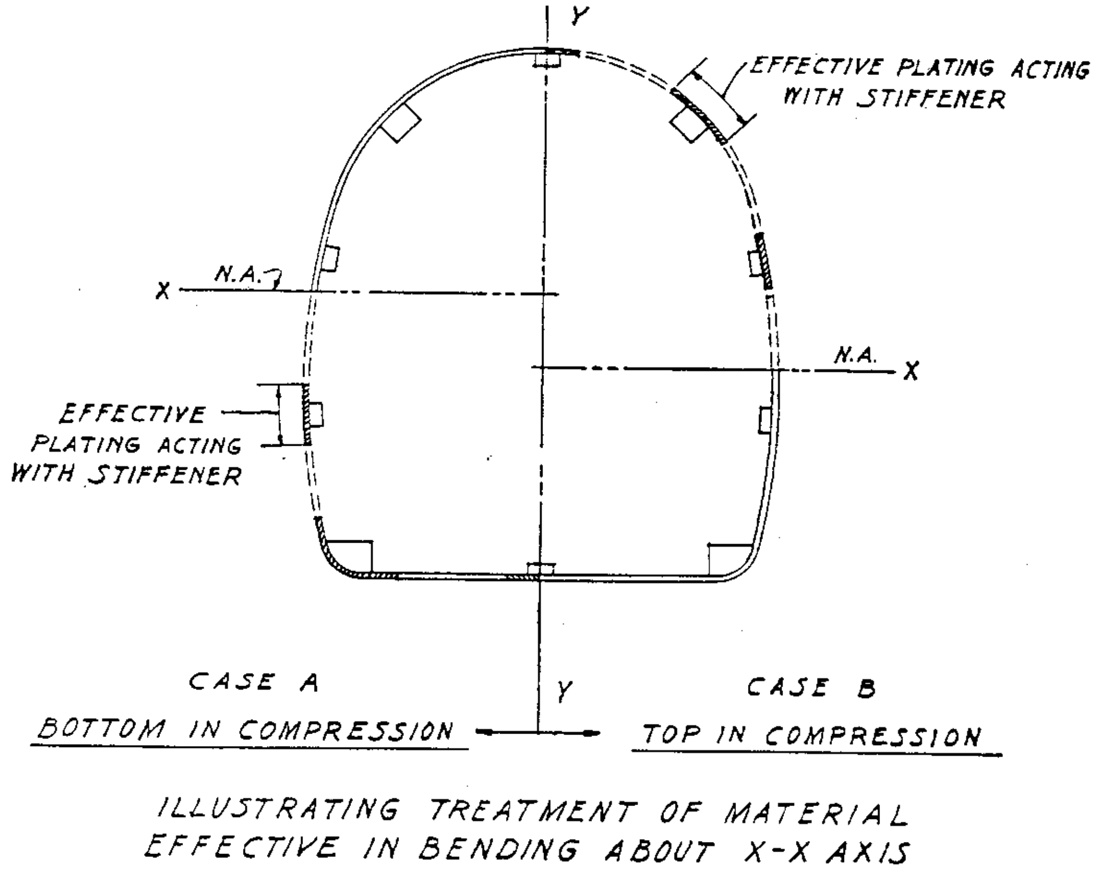


Figure 3–25. Reinforced shell fuselage.

the determination of section properties. In order to account for the effect of shear on the effective widths for stiffeners on the side of the fuselage, it is advisable to compute the effective widths for *all* stiffeners on the compression side on the basis of a panel edge stress corresponding to the allowable stress of the stiffener, rather than the actual stress to which it may be subjected. It is customary to assign to each stiffener and adjacent skin an item number. Prior to actual computations, the designer should make an estimate of the neutral axis location, thereby dividing the elements into those on the compression side and those on the tension side. After the location of the true centroid of the section has been determined, the designer will be able to check the accuracy of his original assumptions as to neutral-axis location.

It usually will be found that no corrections for axis location are necessary if the final axis is located relatively close to the one originally assumed. A procedure similar to that described in section 3.1330 will be found convenient for computing the section properties. Distances and moments originally are taken from some conveniently located reference axis. The sum of moments about the reference axis, after being divided by the sum of the areas in the section, gives the location of the neutral axis of the section. Distances of the items from the neutral axis are then determined. The sum of the products of the areas located on either side of the neutral axis multiplied by the distances to the neutral axis is equal to the static moment of the section about the neutral axis, Q, and the sum of second moments of all of the elements of the section is equal to the moment of inertia of the section, I. Where the axial loads produce appreciable values of bending moments on the fuselage, these moments should be included in the bending moment, M, which is used to obtain the axial stresses due to bending.

Critical stresses commonly are assumed to occur at the stiffeners located farthest away from the neutral axis on the compressive side, and the stresses in these stiffeners resulting from bending are computed by the $\frac{My}{I}$ equation, M being the critical moment at the section and y being the distance of the stiffener from the neutral axis.

Although the bending theory indicates that the outermost fibers are the critical ones, it will often be found that stiffeners located near the top or bottom, on the shoulders of the section, are the ones which are liable to fail during tests if the skin buckles in shear. Such stiffeners usually are subjected to comparatively large direct stresses due to bending and, at the same time, may act as the stiffeners of the tension-field shear material transmitting the shearing stresses to the outermost stiffeners. Unless these stiffeners are of sufficiently large proportions to resist the bending loads imposed by the tension-field effects, failures of these stiffeners may occur at loads smaller than anticipated.

3.423. *Computation of shearing stresses.* The bending material in fuselage sections usually is distributed in such a manner that under symmetrical loadings it may be safely assumed that each side carries half of the vertical shear load, and the corresponding shearing stress, f_s, at any point is equal to $\frac{VQ}{2tI}$, where V=the shear force acting on the section, Q=static moment about the neutral axis of the areas located between the outermost fibers and a horizontal line through the point under consideration, I=moment of inertia of the section, and t=thickness of the skin at the point under consideration.

The sum value, Qx (table 3–5), should be used for determining the maximum shearing stresses that occur at the neutral axis of the fuselage. Although these methods pertain to the analysis of the fuselage for a shear load applied in a vertical direction, similar methods can be employed for a shear load applied horizontally, such as a side load on the vertical tail. If the structure is not too unsymmetrical about a horizontal plane, the shear center for application of the horizontal load may be estimated, using overlapping assumptions. If a more exact solution of shear distribution is desired, the methods of section 3.135 may be used. The total shear stress (or intensity) at any section is that obtained from the superposition of the component shear stresses (or intensities) resulting from vertical loads, transverse loads, and torsion.

Although the fuselage structure as a whole should be checked for the shear distribution as determined in the foregoing, it is recommended that certain sections of the fuselage be checked for other types of shear stress distribution that may be more in line with the actual load application. At the point of wing attachment to the fuselage, for example, very large loads are transmitted to the fuselage frame through the attachment fitting. It is reasonable to assume that high shearing stresses will be present near this fitting, gradually tapering to the extremity of the frame. Although this assumption is not in agreement with the conventional bending theory, it is recommended that it be considered in design to allow for probable shear concentrations.

Torsional shear stresses can be computed by the conventional formula $f_s=\frac{T}{2At}$ and should be combined with the stresses due to direct shear. The tendency of tension fields to sag the stiffeners also should be considered. Because similarity seldom exists between the geometric properties of different airplane structures, it is difficult to draw conclusions from one design as to the allowable shear stresses to be used for other designs. It is usually necessary, therefore, to conduct panel tests on representative curved shear panels.

3.43. Pure-Shell Type. By definition, the pure shell or monocoque type of structure incorporates no longitudinal stiffening members. Hence, the ultimate strength of such a structure may be taken as the critical buckling strength of its elements. As described in chapter 2, the buckling strength of a plywood panel may be estimated from its thickness, frame or stiffener spacing, wood species, arrangement of plies, and curvature. It is generally desirable that no portion of the structure become buckled prior to the application of the design load. In such a case, in the calculation of section properties, the material may be considered fully effective and the stresses determined according to the fundamentals of mechanics.

In a section such as shown in figure 3–26B, however, certain portions may become buckled at low loads without materially affecting the final load-carrying capacity of the total section. This may be exemplified by the buckling of flat panels on the compression side while the major portion of the total flange material is unbuckled by reason of its difference in curvature or thickness. It is generally satisfactorily conservative to omit the buckled material from consideration. Such a partially buckled structure must, of course, be adequately stiffened by frames.

3.431. *Monocoque-shell fuselages.* The basic principles of the design of thin-walled cylinders, as discussed in ANC–5 sections 1.63 and 1.64 can be applied to the design of monocoque fuselages. The monocoque portion of the fuselage structure usually is restricted to certain sections of the fuselage, such as the tail portion. In the center and in the forward portions of the fuselage, the reinforced-shell type of construction, which is more suited to the region where cut-outs are present, generally is used. Careful attention should be given to that part of the fuselage structure where two types of construction join. Adequate length and attachment of the reinforcing members to the shell should be provided. At the points where the monocoque section stops at cut-outs, transfer of the load from monocoque portion to the stiffeners around the cut-out should be investigated carefully (ref. 3–19).

Tests of monocoque fuselages have demonstrated that the strength is dependent to some extent on the smoothness of the plating. The designer should, therefore, be certain that the methods of assembly of monocoque fuselages in the shop will produce a satisfactory product. Where small margins of safety are present and when the effects of load concentrations have not been taken into account conservatively, strength tests should be carried to the full ultimate-load values, because the type of failure in this type of structure usually is elastic, and the appearance of the structure under proof load may be no indication of the ability of the structure to carry the required ultimate loads.

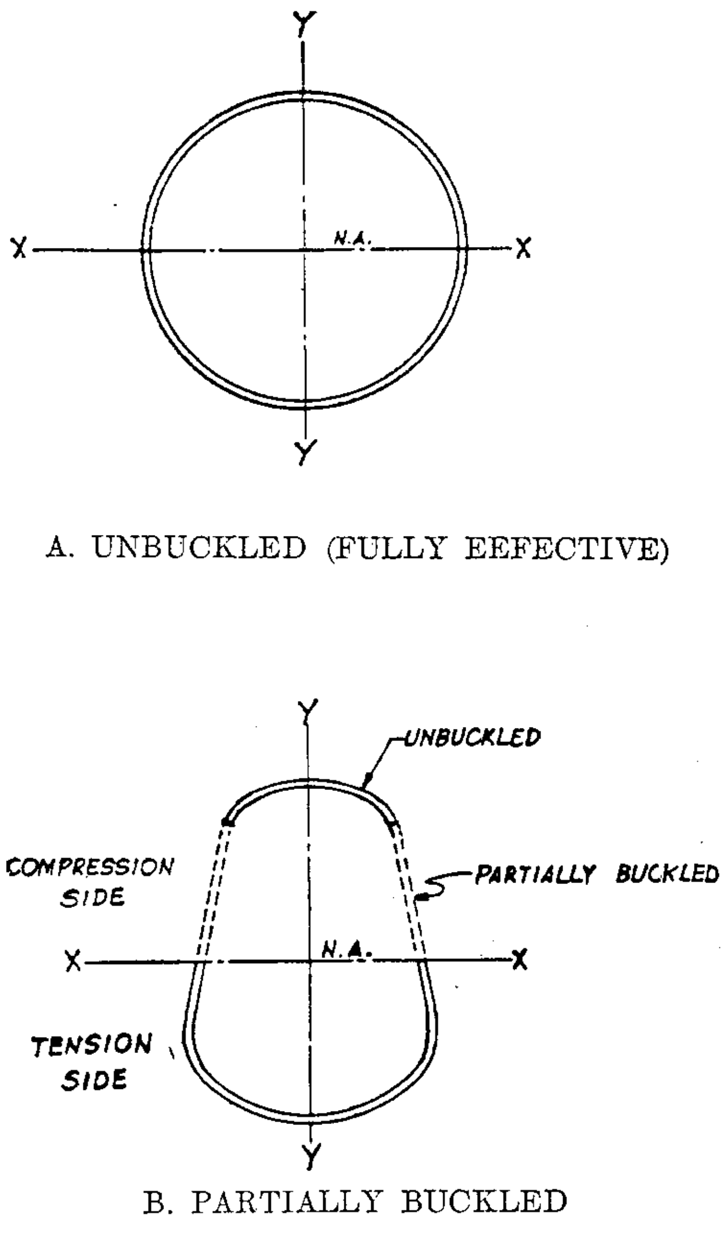


Figure 3–26. Pure shell-type fuselage.

3.44. Miscellaneous Fuselage Analysis Problems. Each new type of fuselage may present a new set of problems which has not occurred in other types. It is recommended, therefore, that every new type of fuselage be tested at least to the critical ultimate loads to determine the presence of possible stress concentrations and other effects which could have been overlooked in the most careful design. Some of the analysis problems which are somewhat common to all types of fuselages are discussed in the following sections.

3.441. *Analysis of seams.* The allowable loads of the seams should be computed and compared with the loads imposed by direct tensile stresses, by shear stresses, by any tension field effects of the shear stresses, and by combined stresses due to the action of all these stresses.

3.442. *Analysis of frames and rings.* The analysis of the fuselage frames constitutes a separate problem. Many manufacturers have adopted certain standard methods of frame analysis, which, although not necessarily mathematically rigorous for the types of the structures considered, have produced satisfactory designs. A general discussion of some of these methods is given.

3.4421. *Main frames.* Main frames are primarily for the purpose of distributing into the fuselage such concentrated loads as the loads from wings, tail surfaces, or landing gear, and those resulting from the local support of items of mass. Main-frame structures usually are of the redundant type and their analysis is based on the principles of least work and related or equivalent methods such as strain energy, column analogy, moment distribution, or joint relaxation (refs. 3–2 and 3–3).

Figures 3–27 A, B, and C show a fuselage main frame under a symmetrical loading condition. The loads from the wing (or landing gear) are shown applied at the applicable fittings and are resisted by shear forces in the fuselage skin. To agree with the elementary bending theory, these shear intensities should be distributed so as to conform to the $\frac{VQ}{2I}$ or $\frac{V}{2h}$ values of the fuselage section, as applicable, giving a distribution of shear forces of the type shown in figure 3–27 A or B. Some designers take into account the fact that, due to concentration of load where the frame is attached to the wing, the shear is carried mostly by the adjacent fuselage skin and the shear resistance of the skin is reduced arbitrarily, somewhat in proportion to its distance from the point of concentrated load application. This would yield a shear force distribution of the type shown in figure 3–27 C. In such cases, the fuselage skin should also be checked for the high stresses indicated.

The ordinary method of frame analysis is strictly applicable to frames the deflections of which are not restricted by the fuselage skin. Actually, the frame deflections may become quite pronounced and the outward deflections are resisted by double-curvature effects in the fuselage skin or by the support of adjacent frames. This action of the skin is equivalent to introduction of inward-acting loads resisting the frame bending and hence to a reduction of frame stresses to smaller values than those indicated by an analysis based upon shear distributions as described. The present development of the theory does not indicate quantitatively just what allowance can be made for this reduction of stresses. It is recommended, therefore, that the frame analysis be conducted by the methods similar to the ones indicated.

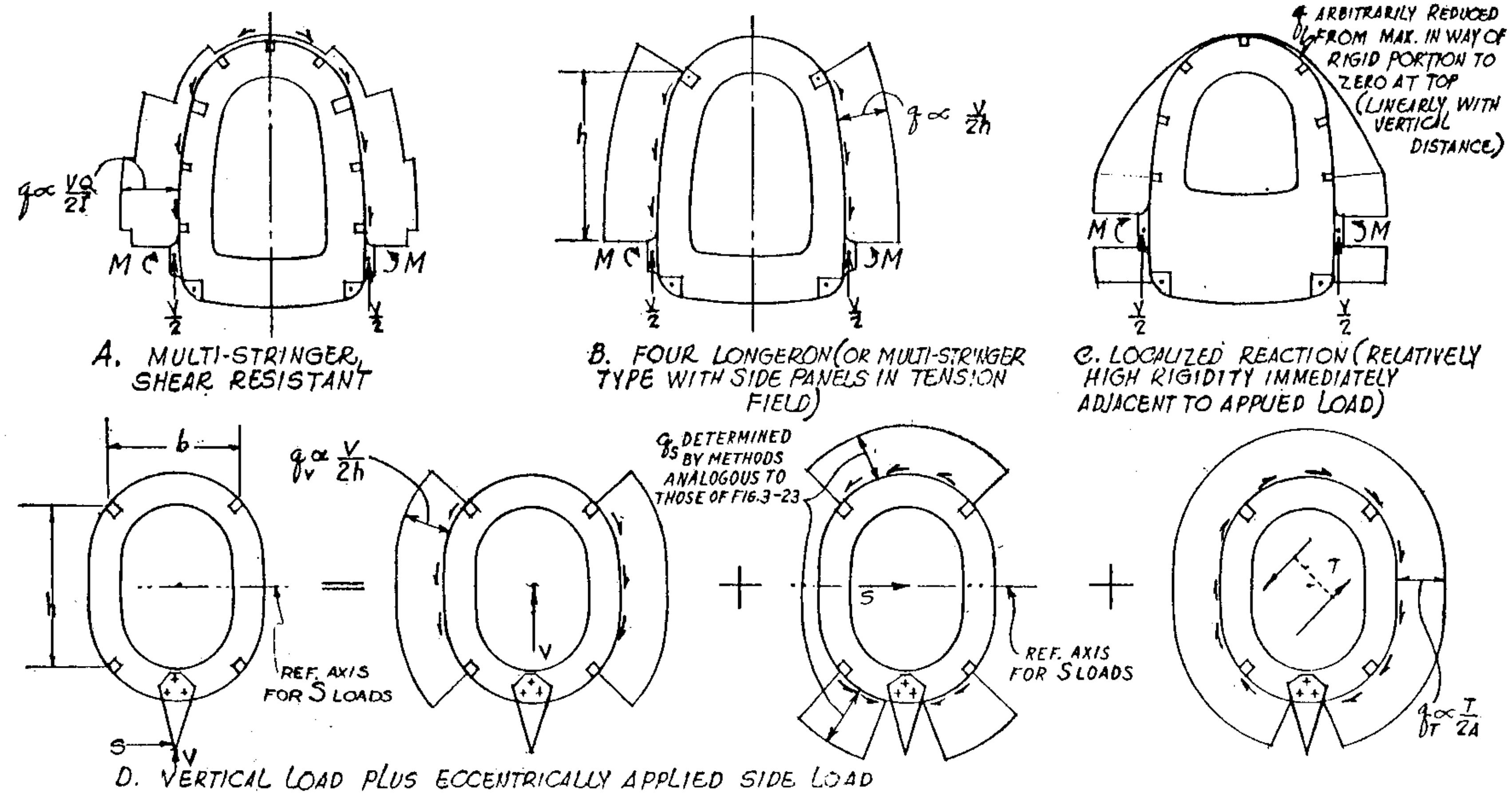
Where relatively deep frames are used, the moments induced by the wing deflections may become important and should, therefore, be analyzed.

3.4422. *Intermediate frames.* Intermediate frames are provided to preserve the shape of the fuselage structure, to reduce the column length of the stiffeners, and to prevent failure of the structure due to general instability. They are subjected to several types of loading; such as, those due to tension fields in the skin, to fuselage bending, to transfer of shear to the fuselage plating. Many of these loads are comparatively small and often tend to balance each other. For these reasons the design of intermediate frames is often based on the experience of the designer or on semi-empirical methods. In the case of large airplanes, however, it becomes of considerable importance to design frames of this type to provide suitable stiffness for the prevention of general instability.

3.443. *Effect of cut-outs.* Effects of cut-outs usually are allowed for by omitting the bending material affected by the cut-out from the computation of the section properties. For shearing stress computations in the location of regularly spaced cut-outs, such as windows, the shear stress in the skin between cut-outs may be taken as equal to that computed on the assumption that no cut-outs are present and then increasing this value by the ratio of distance between cut-out centerlines to the distance between the cut-outs. Such treatment, although quite arbitrary, has served satisfactorily with metal material. Because of the inherent lack of ductility in wood and its inability to deform plastically and redistribute stresses adjacent to local concentrations such as cut-outs, the incorporation of large calculated margins of safety is recommended in such locations.

In case of large openings, such as the cabin door cut-outs, allowance for bending stress redistribution usually is made by modifying the section properties by omitting the material affected by the cut-out. For computation of the shearing stresses, it may be assumed that the direct shear load is carried through that side of the fuselage not containing the cut-out. The couple resulting from this unsymmetrical reaction in way of the cut-out can be assumed to be resisted by a shear couple consisting of equal and oppositely directed transverse reactions in the top and the bottom of the fuselage. The redistribution of the shear stress, as assumed, can be achieved best if bulkheads are provided on both sides of the door. Where only one main bulkhead is provided (at only one end of the cut-out) shear redistribution on the other side of the cut-out must be accomplished by the frame under the flooring and by the intermediate frames. Reference 3–19 describes the basic theory and recommended methods of determining the shear distribution in the plating about cut-outs, and also the corresponding effect of the cut-outs upon the loads in the stringers and frames.

3.444. *Secondary structures within the fuselage.* Often the designer is faced with the problem of existence of a secondary beam structure inside the main fuselage or hull structure. This secondary structure may consist of keels or keelsons in a flying-boat hull, or of the floor supporting structure or nose-wheel retracting tunnel in a fuselage. If this type of structure is analyzed separately under the specified local loads alone, the stress distribution may not correspond to the distribution that will be obtained with it acting in conjunction with



1. q = SHEAR INTENSITY OF REACTIONS PROVIDED BY PLATING PLOTTED AS AN ORDINATE NORMAL TO THE PLATING.
2. A, B, AND C REPRESENT A FRAME UNDER SYMMETRICAL FLIGHT (OR LANDING) LOADS.
3. D REPRESENTS A FRAME UNDER UNSYMMETRICAL LOAD, (SUCH AS AT TAIL WHEEL)
4. LOADS ON FRAME RESULTING FROM LOCALLY SUPPORTED MASS ITEMS ARE NOT SHOWN — BUT SHALL BE CONSIDERED AND PROPERLY BALANCED.
5. PLATING IN WAY OF FITTINGS HAS BEEN SHOWN AS INEFFECTIVE. ALL SHEAR INTENSITIES DETERMINED BY ASSUMING PLATING FULLY EFFECTIVE ARE INCREASED PROPORTIONATELY TO OBTAIN EQUILIBRIUM.

M 41718 F

Figure 3–27. Shear distributions applicable to frame design.

the rest of the fuselage structure. The designer should make certain that the combined effects of the two structures are in agreement and that the action of the structure as a whole is consistent with expected deformations.

3.45. Strength Determination. The strength of the structure is determined by comparison of the calculated internal loads and stresses with the allowables obtained either from tests or from the information given in chapter 2. The determination of the strength of shell structures, including reinforced shells, is presented in detail in section 3.138.

3.5 Hulls and Floats

The analysis of hulls and floats may be treated in a manner similar to that used with fuselage structures, the chief difference being in the manner in which the major external loads are applied, that is, by direct contact with the water in the form of normal pressures. Fundamentally, hull and float structures consist of:

(1) *Bottom plating*—that, in contact with the water, is loaded by the normal pressures developed in landing, take-off, or buoyancy, and transfers such loads to the—

(2) *Bottom stringers*—that support the plating and transfer the plating loading to the supporting—

(3) *Frames*—that in turn carry the water loads through to the—

(4) *Main longitudinal girder* — or general structure. Consideration is given to the fact that water causes concentrated local loads on float and hull bottoms that may reach intensities considerably above the average loading and may be applied at different times and for different durations to different portions of the bottom structure. For these reasons the strength requirements for design of the bottom plating are specified as more severe than those for stringer design. The bottom stringer strength requirements are, in turn, more severe than those for complete frame design. The specified loads as applicable to the design of the general structure are in general of lesser local intensities but are consistent with the design airplane accelerations and total reactions.

3.51. Main Longitudinal Girder. This structure may consist of a centerline truss or bulkhead girder to which the frames, deck, side and bottom plating are attached. Or, the deck, side, and bottom plating and stringers plus other longitudinal material connecting to, and capable of acting with, the skin plating and stringers may be considered as a reinforced shell which comprises the longitudinal girder. In such a structure the frames not only serve to transmit the water loads to the general structure, but provide the transverse and circumferential stiffening for the shell. The effective longitudinal members ordinarily considered to take the bending loads consist of: keel, bottom stringers, keelson, chine, deck, and stiffeners. The effective shear material consists of side, deck, and bottom plating. The analysis assumptions, calculation of section properties, and determination of normal and shearing stresses applicable to the longitudinal girders are in general as described under section 3.4 for fuselage analysis.

3.52. Bottom Plating. Thin plating, when subjected to sufficient normal pressures will either rupture or deflect excessively and take a permanent set. In hulls and floats this latter effect is known as "wash boarding," and in an acceptable structure should not be allowed to occur at loads below those corresponding to yield-point loads. For this reason the design criteria established by the procuring or certificating agency in general consists of specification of certain design-bottom-pressure loadings in conjunction with the permissible permanent deformations at the specified pressure loadings. Permanent deformation is measured at the center of the plating panel, between stringers and relative to the stringers, in a direction normal to the plane of the plating.

The analytical determination of bottom-plating stresses and deflections is exceedingly difficult of accurate attainment, and the problem of design calculation methods, including the basis for allowable stresses, hence lends itself most readily to treatment by testing procedures. Test panels representative of (1) the plating species, thickness and plywood type, (2) the stringer spacing, frame spacing, and panel aspect ratio, and (3) method of edge support and type of edge restraint should be tested under normal pressures, and the applicable strength criteria (ultimate strength, arbitrary or true yield, and permanent deformation) determined. Test data may be interpreted and converted in light of the calculation procedures described in chapter 2.

In such a treatment, two of the influential factors that determine the calculated stresses and deflections are (1) type of edge support, and (2) aspect ratio of panel. Clamped or fixed edges assume the plating to be restrained from any rotation at the edges, the neutral plane of the plywood maintaining zero slope. In simply supported edges, conversely, a possibility of rotation of the neutral plane of the plywood at the edge is implied. The plates actually encountered in the design of floats and hulls lie somewhere between fixed and supported edges and may be considered as elastically restrained. The maximum stress in a plate with fixed edges occurs at the long edges, whereas it occurs in the middle of a plate with simply supported edges. It follows from this that a slight deflection or twist of the fixed edges of a plate will decrease the stress close to the edges where it is a maximum and increase it near the middle where it was, however, originally much less. Bottom stringers are not ordinarily very stiff torsionally and constitute a type of support bordering upon the simply supported edge. On the other hand, keel, keelson, and chine members are necessarily quite stiff torsionally, as well as laterally, in that they must be well gusseted to adjacent frames and, forming the edge of the plate panels, must be stiff enough to prevent lateral deflections. Hence, the analytical treatment under both limiting conditions of edge support should give considerable guidance in design.

The ratio of frame spacing to stringer spacing ordinarly exceeds 3.0 and hence, the aspect ratio of the plating panels for use in design can usually be taken as infinite.

3.53. Bottom Stringers. As previously mentioned, the bottom stringers serve to transfer the bottom plating normal loads to the transverse frames. They may be considered in general as continuous beams supported at the frames with a running load per unit of length equal to the stringer spacing times the intensity of bottom pressure. Under the ordinary conditions of uniform pressure, frame and stringer spacings, the symmetry of loading would permit the consideration of the stringer as a uniformly loaded continuous beam over fixed supports. This would lead to a design bending moment in the stringer:

$$M=\frac{wL^2}{12} \qquad (3:61)$$

where W=stringer transverse loading, in pounds per inch

and L=support spacing, in inches.

The extreme probability of loadings other than symmetrical and the finitely elastic nature of the support restraint leads to the use of the more conservative specification of the design bending moment as:

$$M=\frac{wL^2}{10} \qquad (3:62)$$

When the conditions of loading are definitely different from these assumptions (that is, when the pressure varies, when the stringer is not continuous, or when the support has unusual restraint characteristics) the stringer should of course, be designed to the local conditions specifically applicable.

It is rational to consider a portion of the plating adjacent to a stringer as effectively contributing to the section properties of the stringer. It is important that the same assumptions as to plating effectiveness be used in converting test data into allowable stresses as is used in the analysis of the flight article under the specified loads.

As well as being analyzed for the specified design bottom-pressure loading, the plating and stringer combination should be checked for the conditions in which it is both subjected to direct water loads and also forms a part of the effective flange material of the general longitudinal girder structure. In such conditions, the stresses resulting from the bottom pressures consistent with the loading on the general structure are superimposed upon the stresses incurred as a portion of the flanges of the general structure.

3.54. Frames. Hull and float frame design differs from ordinary fuselage frame design principally in the nature of the applied loads which result from direct water pressures. Each frame is considered to take the bottom loadings applied to the plating and stringer combination structure in the area adjacent to the frame. Such loaded area extends approximately one-half of the frame spacing to both sides. The bottom loads are usually transmitted from the stringers directly to the frame in the area between the chines. The assumptions as to the nature and magnitude of the balancing reactions in the form of shear in the side and deck plating may be patterned after those used in fuselage frame design.

In almost all instances, frame analysis involves the problem of the application of the fundamental methods of least work and in this respect may be treated in a manner similar to that employed with analogous fuselage frames. The probability of unsymmetrical loading applications on V-bottom hulls and floats in take-off and landing is quite high. For this reason the procuring or certifying agency specifies in all instances certain unsymmetrical design-loading conditions. Such loading conditions are often critical for the design of frames, and hence the analysis of frames loaded in this manner should be given the utmost care and consideration.

3.55. STRENGTH DETERMINATION. The strength of the structure is determined by comparison of the calculated internal loads and stresses with the allowables obtained either from tests or from the information given in chapter 2. The determination of the strength of shell structures, including reinforced shells, is presented in detail in section 3.138.

3.6. Miscellaneous

Treatment of the wing, fuselage, hull, tail, and control surfaces does not complete the stress analysis of the airplane structure. In airplanes of wood construction, however, it is considered that these same structural components constitute nearly all of those in which the use of wood is significant and in the analysis of which the physical properties of wood will enter as an important factor. Hence, for such reasons and as explained in section 3.00, treatment of the detailed analysis problems related to the remaining important airplane structural components will not be included herein. Such components would include, for example, landing gear, engine mount, control systems, fittings, and joints. The determination of the design load applied to each individual wood structural element of a joint (mechanical joint or glue joint), or fitting attachment, may be determined by basic principles of mechanics and machine design. Where it would significantly affect the distribution of the design load, the nonisotropic nature of wood, which results in the strength and elastic characteristics being dependent upon the relation between the directions of the load and of the grain, should be taken into account by a rational treatment or provided for by conservative arbitrary assumptions. The design load thus determined for such an element should be compared with the allowable load defined by the applicable portions of chapter 2 (principally section 2.9). The designer is referred, in general, to the many existing tests, technical papers, and publications which adequately handle such miscellaneous analysis problems.

REFERENCES FOR CHAPTER 3

(3–1) AKERMAN, J. D. AND STEPHENS, B. C.
1938. *Polar Diagrams for Solution of Axially Loaded Beams.* Jour. Aero. Sci. July, 1938.

(3–2) CROSS, HARDY
1930. *The Column Analogy.* Univ. of Illinois Eng. Exp. Sta. Bulletin 215.

(3–3) ———
1930. *Analysis of Continuous Frames by Distributing Fixed-End Moments.* Proc. A.S.C.E. May, 1930.

(3–4) ERLANDSEN, O. AND MEAD, L.
1942. *A Method of Shear-Lag Analysis of Box Beams for Axial Stresses, Shear Stresses, and Shear Center.* N.A.C.A. Advance Restricted Report.

(3–5) HATCHER, ROBERT S.
1937. *Rational Shear Analysis of Box Girders.* Jour. Aero. Sci. April, 1937.

(3–6) EBNER, HANS
1934. *Torsional Stresses in Box Beams With Cross Sections Partially Restrained Against Warping.* N.A.C.A. Tech. Memo. 744.

(3–7) KUHN, PAUL
1935. *Analysis of Two-Spar Cantilever Wings With Special Reference to Torsion and Load Transference.* N.A.C.A. Tech. Rept. 508.

(3–8) ———
1935. *Bending Stresses Due to Torsion in Cantilever Box Beams.* N.A.C.A. Tech. Note 530.

(3–9) ———
1938. *Approximate Stress Analysis of Multi-Stringer Beams With Shear Deformation of the Flanges.* N.A.C.A. Tech. Rept. 636.

(3–10) KUHN, PAUL
1939. *Loads Imposed on Intermediate Frames of Stiffened Shells.* N.A.C.A. Tech. Note 687.

(3–11) ———
1939. *Some Elementary Principles of Shell Stress Analysis With Notes on the Use of the Shear Center.* N.A.C.A. Tech. Note 691.

(3–12) ———
1942. *A Method of Calculating Bending Stresses Due to Torsion.* N.A.C.A. Advanced Technical Report. (Restricted)

(3–13) KUHN, P. AND CHAIRITO, P.
1941. *Lag in Box Beams, Methods of Analysis and Experimental Investigations.* N.A.C.A. Tech. Note 739. (Restricted)

(3–14) LUNDQUIST, E. AND SCHWARTZ, E. B.
1942. *A Study of General Instability of Box Beams With Truss Type Ribs.* N.A.C.A. Tech Note 866. (Restricted)

(3–15) NILES, A. S. AND NEWELL, J. S.
1938. *Airplane Structures.* Second edition John Wiley and Sons, Inc.

(3–16) ROWE, C. J.
1924. *Application of the Method of Least Work to Redundant Structures.* A.C.I.C. 495.

(3–17) SCHWARTZ, A. M. AND BOGERT, R.
1935. *Analysis of a Strut With a Single Elastic Support in the Span, With Applications to the Design of Airplane Jury-Strut Systems.* N.A.C.A. Tech. Note 529.

(3–18) SHANLEY, F. R. AND COZZONE, F. P.
1941. *Unit Method of Beam Analysis.* Jour. Aero. Sci. April, 1941.

(3–19) WAGNER, H.
1937. *The Stress Distribution in Shell Bodies and Wings as an Equilibrium Problem.* N.A.C.A. Tech. Memo. 817.

CHAPTER 4

DETAIL STRUCTURAL DESIGN

4.0. General

4.00. Introduction. Detail design practice is constantly changing and current good practice may at any time be obsoleted by some new treatment of a particular design problem. Therefore, the examples presented on the following pages represent only the current methods used in handling problems of design details. It should be remembered, however, that many of these methods have withstood the test of time, having been used since the first introduction of wood aircraft.

4.01. *Definitions.* The following definitions explain a few general terms which are sometimes confused by the wood aircraft designer. Other terms requiring definition are explained as they appear in the text.

4.010. *Solid wood.* Solid wood or the adjective "solid" used with such nouns as beam or spar refers to a member consisting of *one piece* of wood.

4.011. *Laminated wood.* Laminated wood is an assembly of two or more layers of wood which have been glued together with the grain of all layers or laminations approximately parallel.

4.012. *Plywood.* Plywood is an assembled product of wood and glue that is usually made of an odd number of thin plies (veneers) with the grain of each layer at an angle of 90° with the adjacent ply or plies.

4.013. *High-density material.* The term "high density material" as used throughout this chapter includes compreg or similar commercial products, heat stabilized wood, or any of the hardwood plywoods commonly used as bearing or reinforcement plates.

4.1 Plywood Covering

4.10. General. Nearly all wood aircraft structures are covered with stressed plywood skin. The notable exceptions are control surfaces and the rear portion of lightly loaded wings. Shear stresses are almost always resisted by plywood skin, and in many cases, a portion of the bending and normal loads is also resisted by the plywood.

4.11. Joints in the Covering. Lap, butt, and scarf joints are used for plywood skin.

When plywood joints are made over relatively large wood members, such as beam flanges, it is desirable to use splice plates, often called aprons or apron strips, regardless of the type of joint. It is desirable to extend the splice plates beyond the edges of the flange so that the stress in the skin will be lowered gradually, thus reducing the effect of the stress concentration at this point. Splice plates (fig. 4–1) can be made to do double duty if they are scalloped corresponding to rib locations so that they may act as gussets for the attachment of the ribs.

Scarf joints are the most satisfactory type and should be used whenever possible. Scarf splices in plywood sheets should be made with a scarf slope not steeper than 1 in 12 (fig. 4–2). Some manufacturers prefer to make scarf joints in such a way that the external feather edge of the scarf faces aft in order to avoid any possibility of the airflow opening the joint.

If butt joints (fig. 4–3) are made directly over solid or laminated wood members, as over a spar or spar flange, experience has indicated that there is a tendency to cause splitting of the spar or spar flange at the butt joint under relatively low stresses. A similar tendency toward cleavage exists where a plywood skin terminates over the middle of a wood member instead of at its far edge.

Lap joints (fig. 4–4) are not recommended because of the eccentric load placed upon the glue line. If this type is used it should be made parallel to the direction of airflow, only, for obvious aerodynamic reasons.

4.12. Taper in Thickness of the Covering. Loads in the plywood covering usually vary from section to section. When this is so, structural

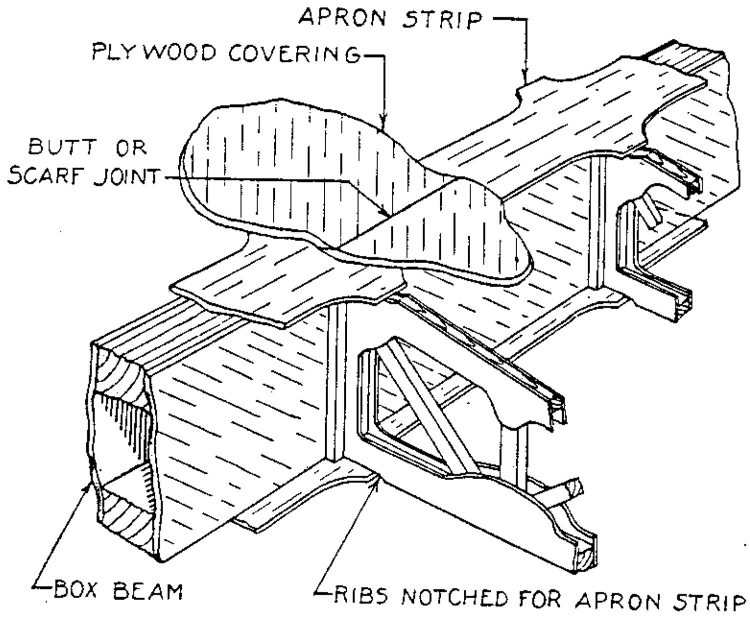


Figure 4–1. Use of splice plate or apron strip.

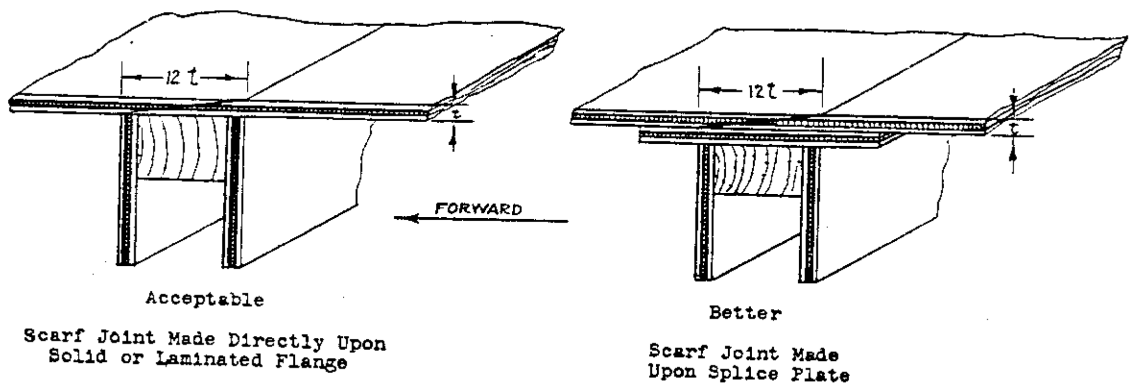


Figure 4–2. Scarf splices.

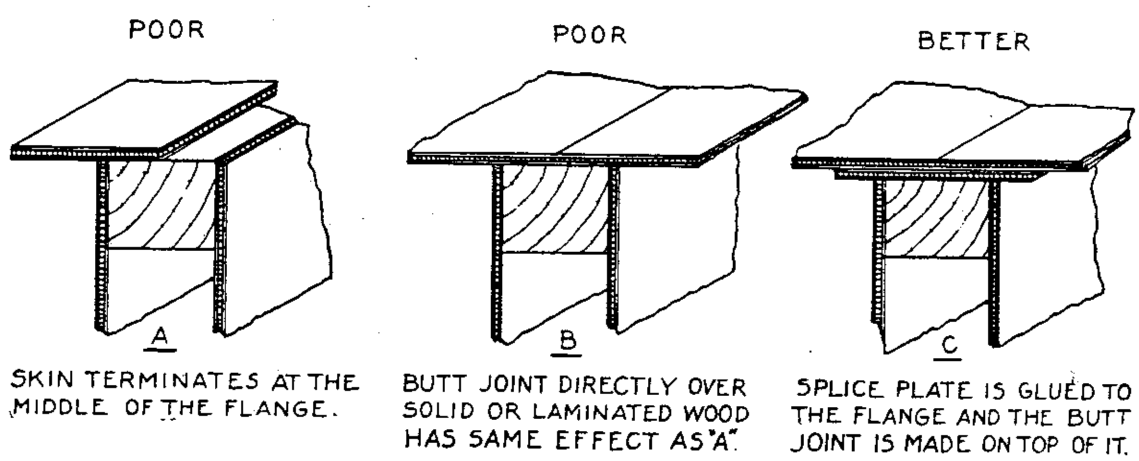


Figure 4–3. Butt splices.

efficiency may be increased by tapering the plywood skin in thickness so that the strength varies with the load as closely as possible (fig. 4–5). To taper plywood in thickness plies should be added as dictated by increasing loads. In doing so, the plywood should always remain symmetrical. For example, plywood constructed of an odd number of plies of equal thickness can be tapered, and at the same time maintain its symmetry, by adding two plies at a time. This method is suitable for bag molding construction. Stress concentrations should be avoided by making the change in thickness gradual, either by feathering or by scalloping. In bag molding construction, the additional plies are often added internally so that the face and back are continuous.

When flat plywood is used, the usual method of tapering skin thickness is to splice two standard plywood sheets of different thicknesses at an appropriate rib station with a slope of scarf not steeper than 1 in 12 as shown in figure 4–6.

4.13. Behavior Under Tension Loads. Because the proportional limit in tension and the ultimate tensile strength of wood are reached at approximately the same time, plywood skin loaded in tension must be designed very carefully. Observation of various static test articles has indicated that square-laid plywood (plywood laid so that face grain is parallel or perpendicular to the direction of the principal bending stresses) has a tendency to rupture in tension before the ultimate strength of the structure has been reached (fig. 4–7). Diagonal plywood, however, seldom ruptures before some other structural member fails. The reason for this behavior is probably due partly to the fact that none of the fibers of the diagonal plywood are in pure tension. The failure under tension load at 45° to the grain is almost entirely a shear failure, and the fibers, which have a definite yield beyond the proportional limit in shear, may undergo enough internal adjustment to permit the plywood to deflect with the structure until some other member becomes critically loaded. Square-laid plywood does not yield because some of its plies will fail in tension almost immediately after the proportional limit has been reached. This drawback of square-laid plywood becomes less important when the skin is designed to carry a greater proportion of the bending loads. For the limiting case of a shell structure without flanges, square-laid plywood is preferable.

Rupture of the skin is also influenced by its relative distance from the neutral axis. If the beam or beams are located so that part of the skin is appreciably farther from the neutral axis than the beam flanges, the skin is more likely to have a premature failure than if the flanges are located at the greatest outer fiber distance. Such a con-

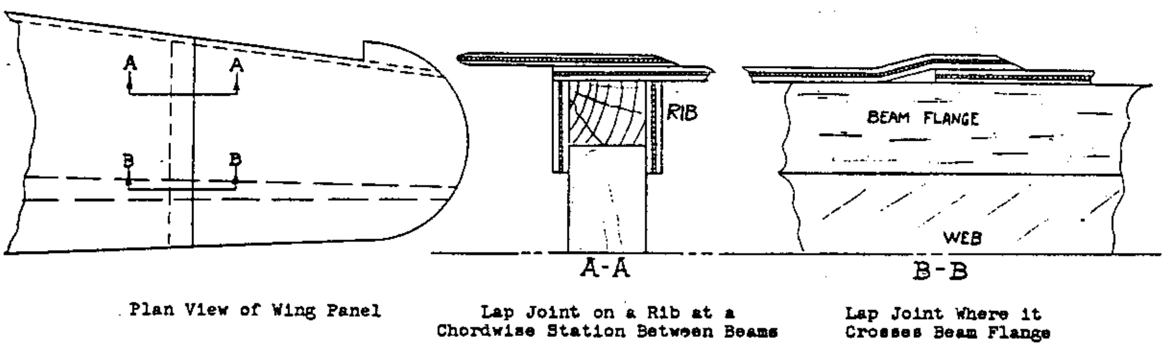


Figure 4–4. Lap splices.

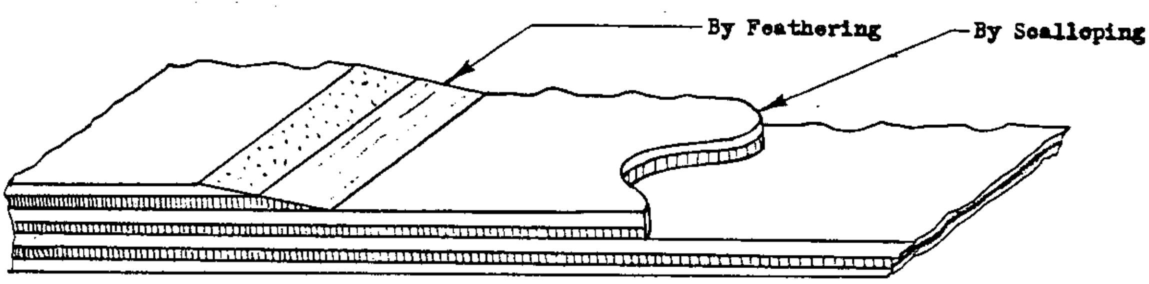


Figure 4–5. Tapering plywood in thickness

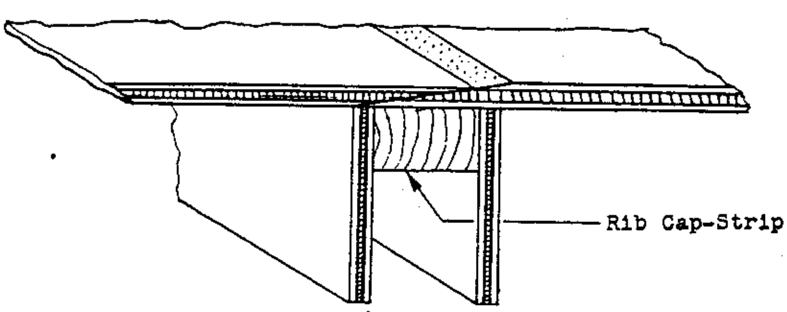


Figure 4–6. Scarfing plywood of different thicknesses.

dition is illustrated by wing spars placed at the 15 and 65 percent chordwise stations of a normal airfoil.

Where the spanwise plies of plywood covering are of a wood species different from the beam flanges, it is, of course, desirable that such plies have a ratio of ultimate tensile stress to modulus of elasticity equal to or greater than that of the beam flanges.

4.14. BEHAVIOR UNDER SHEAR LOADS. Diagonal plywood (face grain at 45° angle to the edge of the panel) is approximately five times stiffer in shear than square-laid plywood and somewhat stronger. When shear strength or stiffness is the controlling design consideration, diagonal plywood should be used (sec. 4.22).

4.15. PLYWOOD PANEL SIZE. In certain cases the size of plywood panels is dictated by the magnitude of directly computable stresses. These occur, for example, in spar webs, D-tube nose skin, and fuselage side panels subjected to high shear. In many other cases, however, the design loads are insignificant. It then becomes necessary to choose combinations of skin thickness and panel size which will stand up under expected handling loads, have acceptable appearance, and aerodynamic smoothness. The typical values given in table 4–1 have been employed by experienced manufacturers.

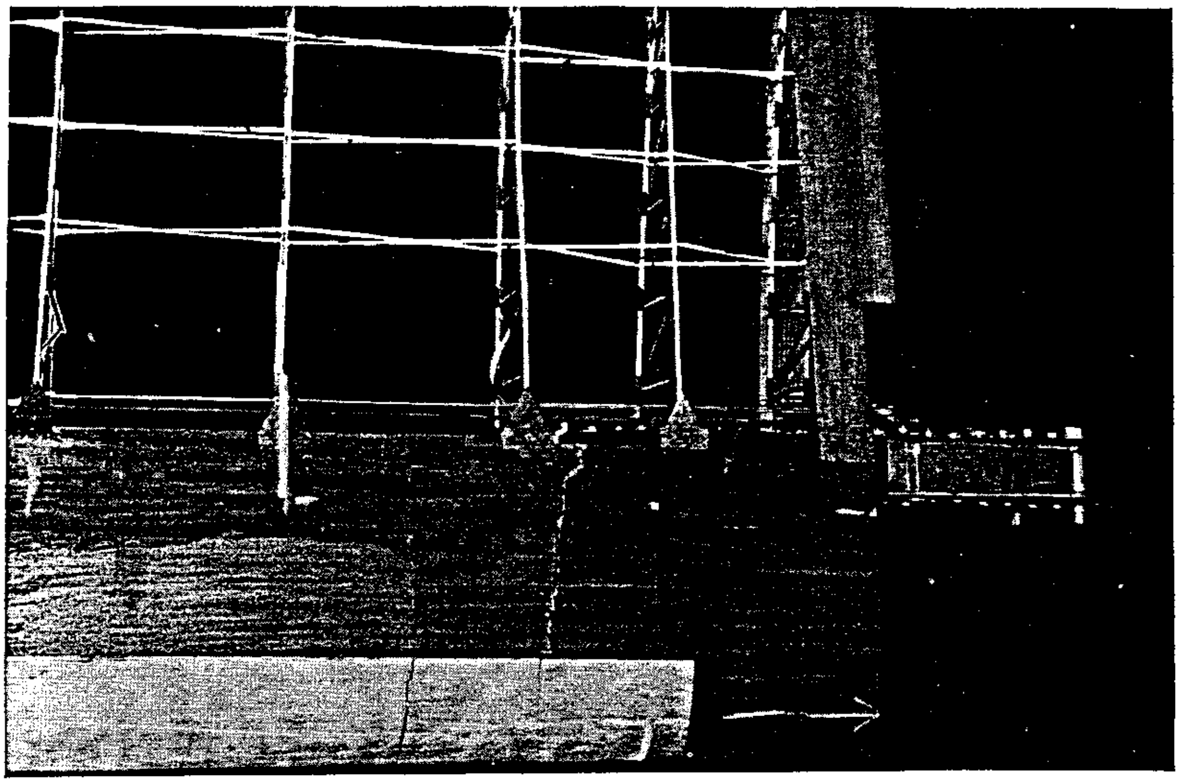


Figure 4–7. Static test wing showing tension failure of plywood covering.

Table 4–1. Typical panel sizes

Material	Thickness	Panel size	Location	Remarks
	Inch	*Inch*		
Mahogany, yellow poplar core	1/16–3/32	12 by 24 maximum	Wing skin.	
Do	1/16	9½ by 10½	do	Spawnwise face grain.
Do	1/16	10 by 12	do	
Do	1/16	5 by 9	Leading edge skin.	
Do	1/16	11 by 20	Vertical fin.	
Do	1/16	10 by 11	Stabilizer.	
Do	1/8	24 or 36 square	Fuselage	Some curvature required.
Mahogany	1/16	7 by 14	Leading edge skin	Spanwise face grain.
Do	1/16	18 by 24	Fuselage	Just aft of cabin.
Yellowpoplar	3/32	14 by 36	Wing aft of 50 percent chord.	

4.16. CUT-OUTS. When cut-outs are made in plywood skin for windows, inspection holes, doors, or other purposes, sharp corners should be avoided, and for all but small holes in low-stressed skin, a doubler should be glued to the skin around the cut-out. For some types of cut-outs a framework can be installed to carry the shear load and doublers need not be used (figs. 4–8, 4–9, and 4–10).

4.2. Beams

4.20. TYPES OF BEAMS. The types of beams shown in figure 4–11 have been used frequently as wing spars, control surface spars, floor beams and wing ribs. The terms "beam" and "spar" are often used interchangeably and both are used in this chapter.

The wood-plywood beams (box-, I-, double I-, and C-) are generally more efficient load-carrying members than the plain wood types (plain rectangular and routed). A discussion of the relative merits of these various beam types is presented in succeeding paragraphs.

The box beam is often preferred because of its flush faces which allow easy attachment of ribs (sec. 4.32). The interior of box beams must be finished, drained, and ventilated. Inspection of interiors is usually difficult. The shear load in a box beam is carried by two plywood webs. By checking shear web allowables by the method given in section 2.73, it will be seen that for the same panel size a plywood shear panel half the thickness of another will carry *less* than half the shear load which can be carried by the thicker panel.

The preceding statement points to an outstanding advantage of the I-beam since its shear strength is furnished by a single shear web rather than the two webs required of a box or double I-beams. Also, the I-beam produces a more efficient connection between the web and flange material than the box beam in cases where the web becomes buckled before the ultimate load is

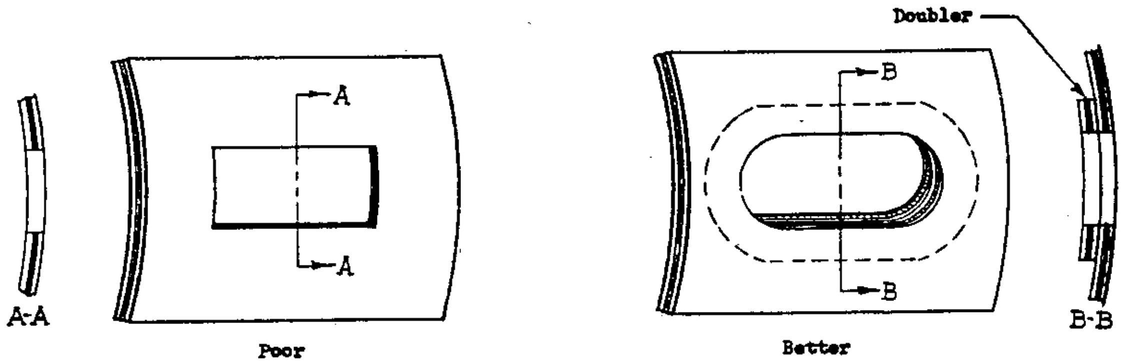


Figure 4–8. Plywood cut-outs.

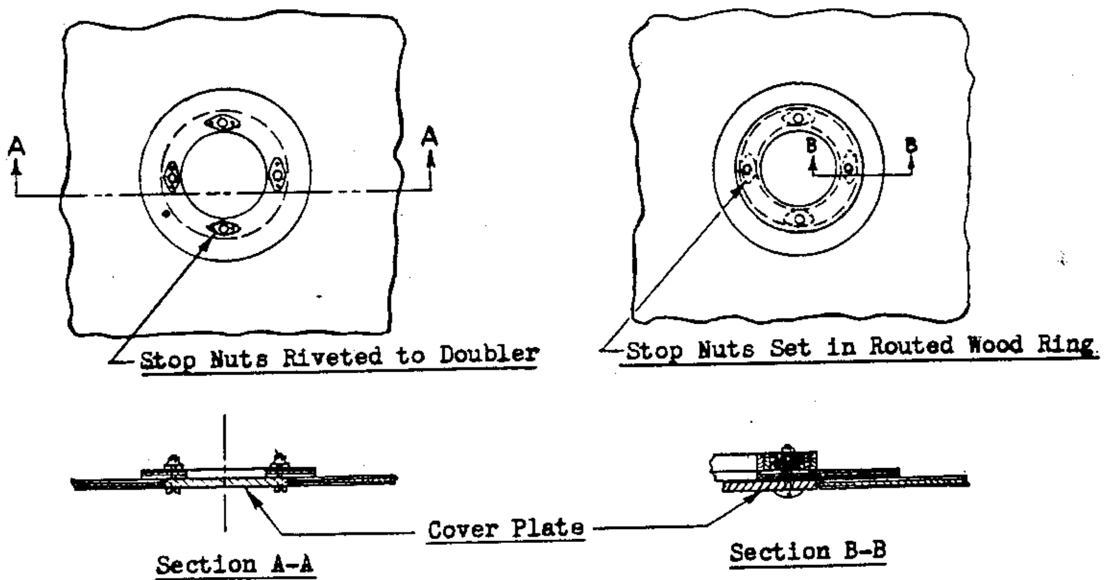


Figure 4–9. Two methods of attaching inspection hole covers.

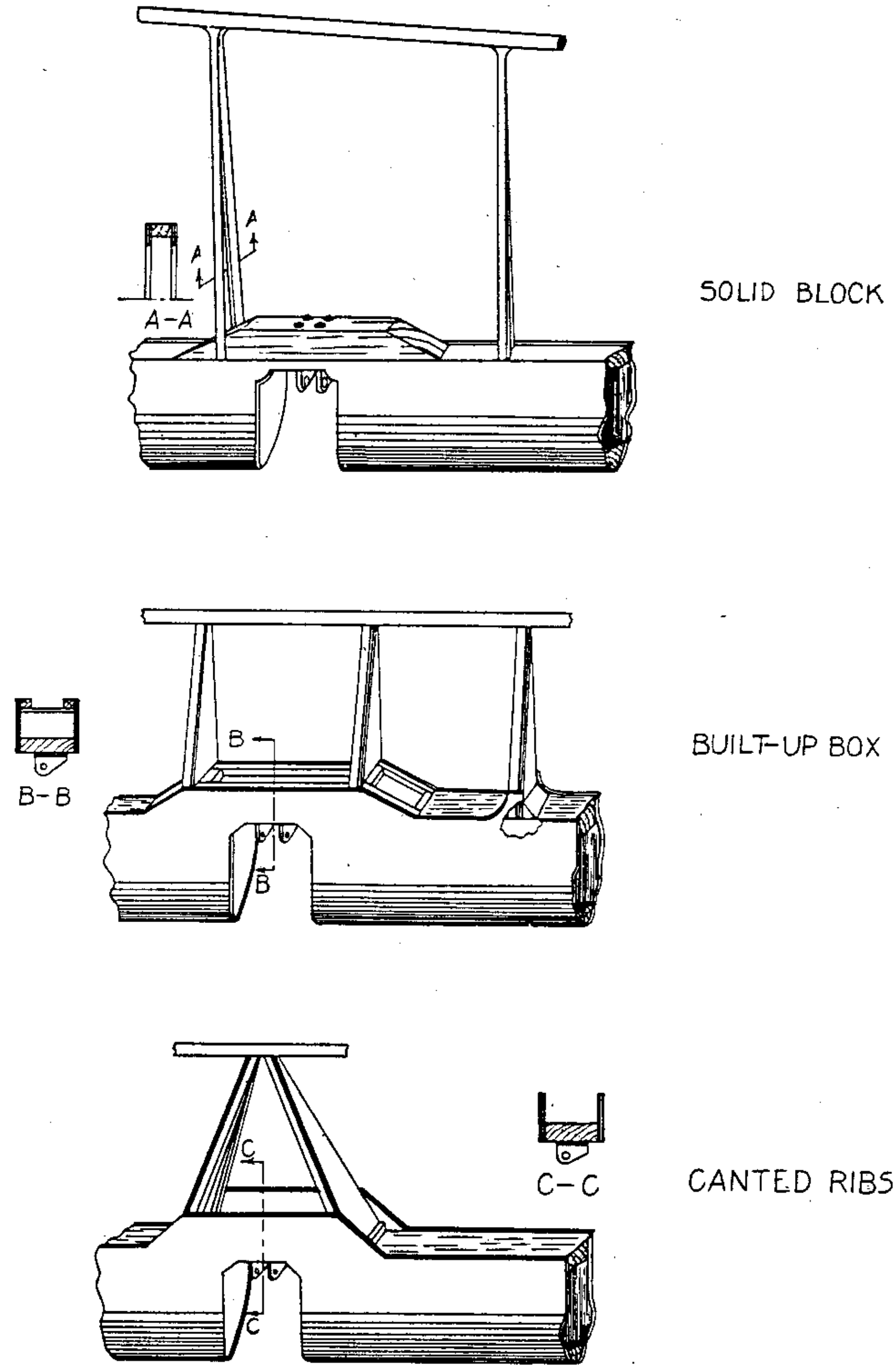


Figure 4–10. Methods of carrying torsion loads around hinge cut-outs in control surfaces.

reached. This is because the clamping action on the webs tends to reduce the possibility of the tension component of the buckled web cleaving it away from the flange.

The double I-beam is usually a box beam with external flanges added along that portion where the bending moments are greatest. This type allows a given flange area to be concentrated farther from the neutral axis than other types.

The C-beam affords one flush face for the flush type of rib attachment but it is unstable under shear loading. C-beams are generally used only as auxiliary wing spars or control surface spars.

Plain rectangular beams are generally used where the saving in weight of the wood-plywood types is not great enough to justify the accompanying increase in manufacturing trouble and cost. This is usually the case for small wing beams, control-surface beams, and beams that would require a great deal of blocking.

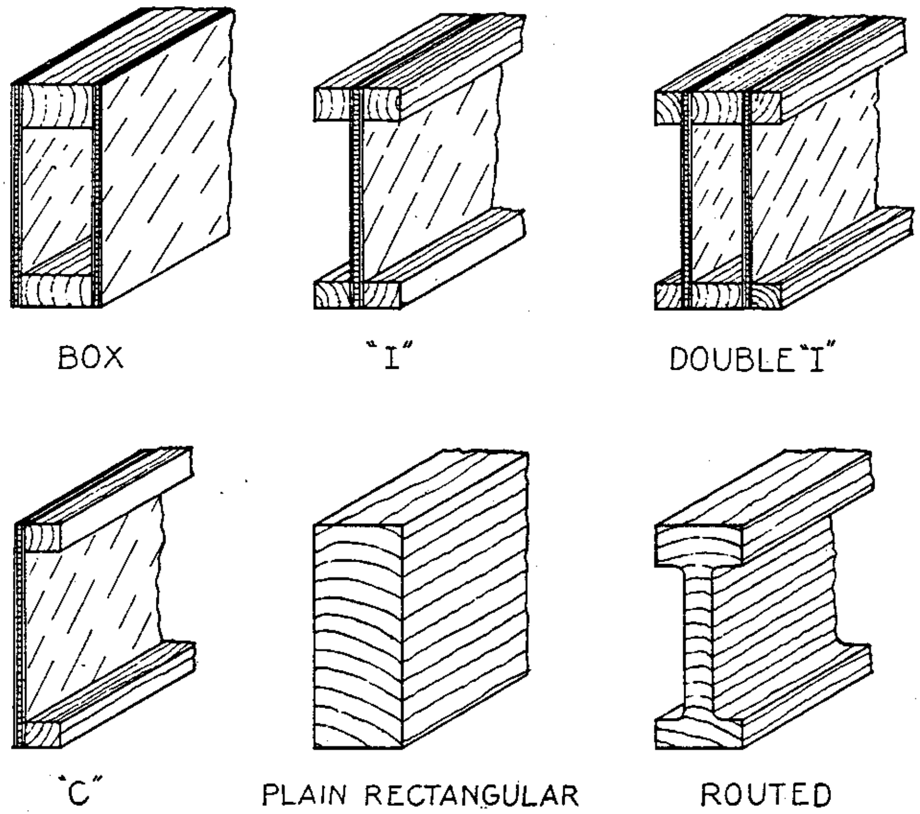


Figure 4-11. Types of beams.

Routed beams are somewhat lighter than the plain rectangular type for the same strength but not so light as wood-plywood types. Usually this small weight saving does not justify the increase in fabrication effort and cost.

In determining the relative efficiency of any beam type, reduction in allowable modulus of rupture due to form factors must be considered.

4.21. LAMINATING OF BEAMS AND BEAM FLANGES. Beam flanges and plain rectangular and routed beams can be either solid or laminated. A detailed discussion of methods of laminating beams and beam flanges is presented in section 2.4 of ANC Bulletin 19, Wood Aircraft Inspection and Fabrication (ref. 2–24).

Since the tension strength of a wood member is more adversely affected by any type of defect than is any other strength property, it is recommended that all tension flanges be laminated in order to minimize the effect of small defects and to avoid the possibility of objectionable defects remaining hidden within a solid member of large cross section.

4.22. SHEAR WEBS. Although square-laid plywood has been used extensively as shear webs in the past, the present trend is to use diagonal plywood (fig. 4–12) because it is the more efficient shear carrying material (sec. 4.14).

It is desirable to lay all diagonal plywood of an odd number of plies so that the face grain is at right angles to the direction of possible shear buckles. In this way the shear web will carry

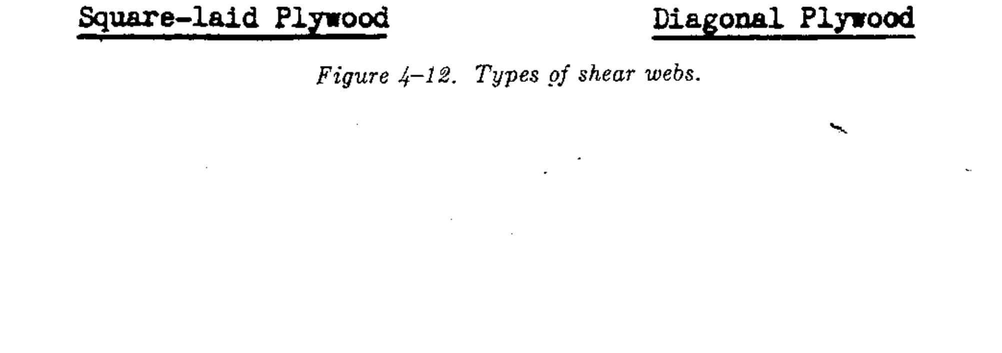


Figure 4-12. Types of shear webs.

appreciably higher buckling and ultimate loads because plywood is much stiffer in bending in the direction of the face grain and offers greater resistance to buckling if laid with the face grain across the buckles (fig. 4–13). This effect is greatest for 3-ply material.

Figure 4–14 illustrates various methods of splicing shear webs. If the splices are not made

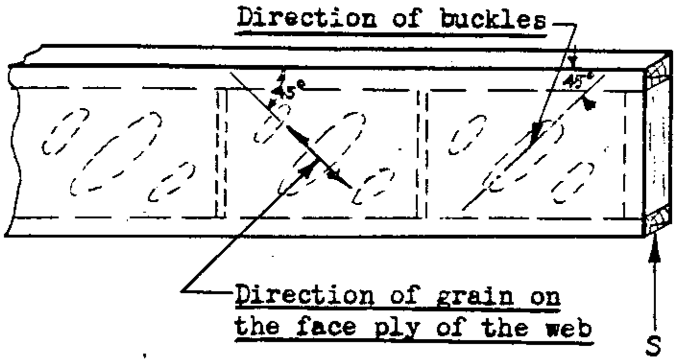


Figure 4–13. Orientation of face grain direction of diagonal plywood shear webs.

prior to the assembly of the web to the beam, blocking must be inserted at the splice locations to provide adequate backing for the pressure required to obtain a satisfactory glue joint.

4.23. Beam Stiffeners. Shear webs should be reinforced by stiffeners at frequent intervals as the shear strength of the web depends partly upon stiffener spacing (fig. 4–15). In addition to their function of stiffening the shear webs, the ability of beam stiffeners to act as flange spreaders is very important and care must be exercised to provide a snug fit between the ends of the stiffeners and the beam flanges. External stiffeners for box beams are inefficient because of their inability to act as flange spreaders.

Stiffeners are usually placed at every rib location so that the web will be stiffened sufficiently to resist rib-assembly pressures.

4.24. Blocking. Any blocking, introduced for the purpose of carrying fitting loads (fig. 4–16),

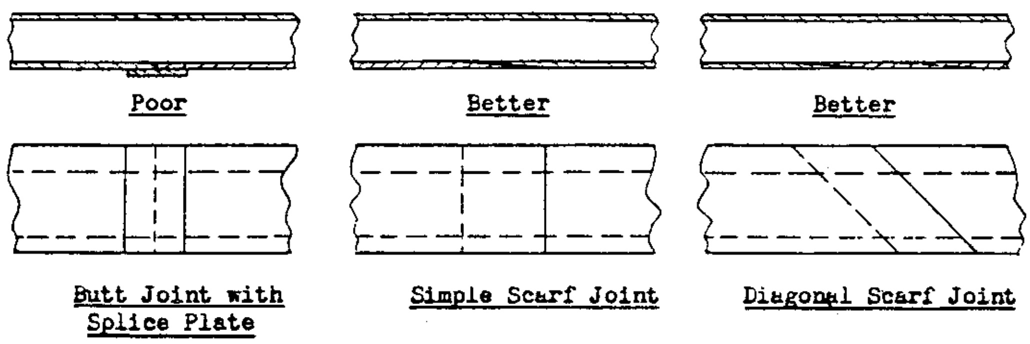


Figure 4–14. Methods of splicing shear webs.

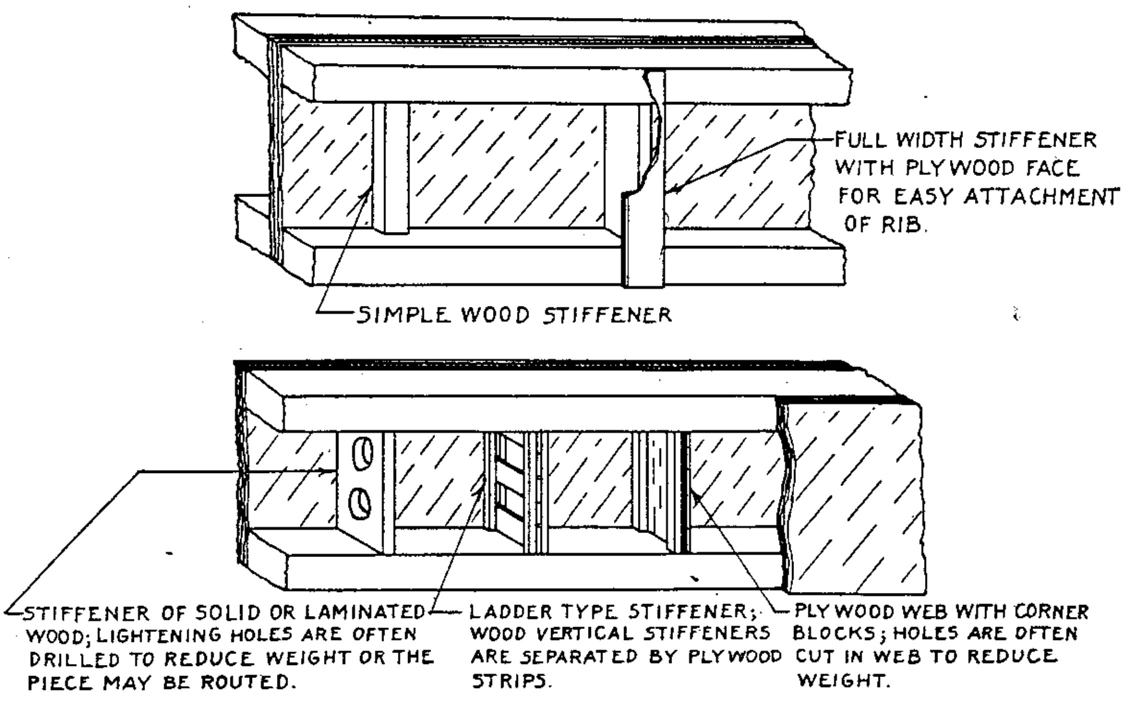


Figure 4–15. Typical stiffeners for I- and box-beams.

should be tapered as much as possible to avoid stress concentrations. It is desirable to include a few cross-banded laminations in all blocking in order to reduce the possibility of checking.

4.25. Scarf-Joints in Beams. The following requirements should be observed in specifying scarf joints in solid or laminated beams and beam flanges:

1. The slope of all scarfs should be not steeper than 1 in 15. The proportion of end grain appearing on a scarfed surface is undesirably increased if the material to be spliced is somewhat cross-grained, and the scarf is made "across" rather than in the general direction of the grain (fig. 4–17). For this reason it is very desirable that the following note be added to all beam drawings showing scarf joints:

   Note. Where cross grain within the specified acceptable limits is present, all scarf cuts should be made in the geneal direction of the grain slope.

2. In laminated members the longitudinal distance between the nearest scarf tips in adjacent laminations shall be not less than 10 times the thickness of the thicker lamination (fig. 4–18).

In addition to the previously mentioned specific requirements, it is recommended that the number of scarf joints be limited as much as possible; the location be limited to the particular portion of a member where margins of safety are most adequate and stress concentrations are not serious; and special care be exercised to employ good technique in all the preparatory gluing, and pressing operations.

4.26. Reinforcement of Sloping Grain. Where necessary tapering produces an angle between the grain and edge of the piece greater than the allowable slope for the particular species, the piece should be reinforced to prevent splitting by gluing plywood reinforcing plates to the faces (fig. 4–19).

4.3. Ribs

4.30. Types of Ribs. Rib design has changed very little for several years. See N. A. C. A. Technical Report 345 (ref. 2–64). The more common types are the plywood web, the lightened plywood web, and the truss. The truss type is undoubtedly the most efficient, but lacks the simplicity of the other types.

For fabric-covered wings the ribs are usually one piece with the cap strips continuous across the spars. When plywood covering is used the ribs are usually constructed in separate sections (fig. 4–20).

Continuous gusset stiffen cap strips in the plane of the rib. This aids in preventing buckling and helps obtain better rib-skin glue joints where nail gluing is used because such a rib can resist the driving force of nails better than other types. Continuous gussets (fig. 4–21) are more easily handled than the many small separate gussets otherwise required.

Any type of rib may be canted to increase the

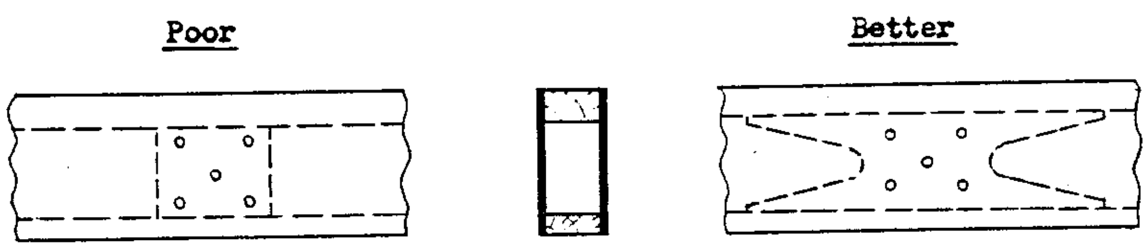


Figure 4–16. Bearing blocks in box spar.

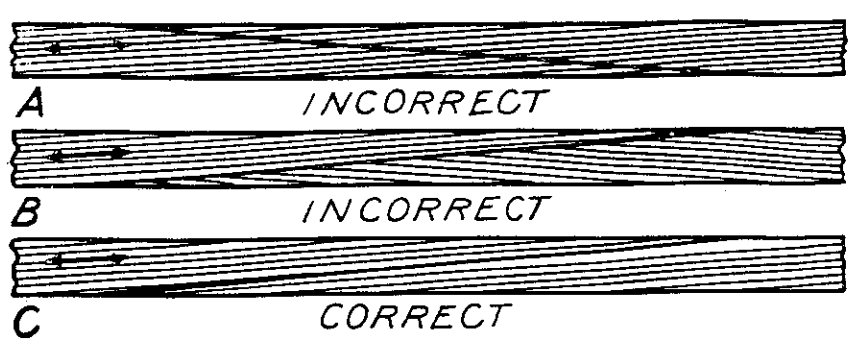


Figure 4–17. Relationship between grain slope and scarf slope.

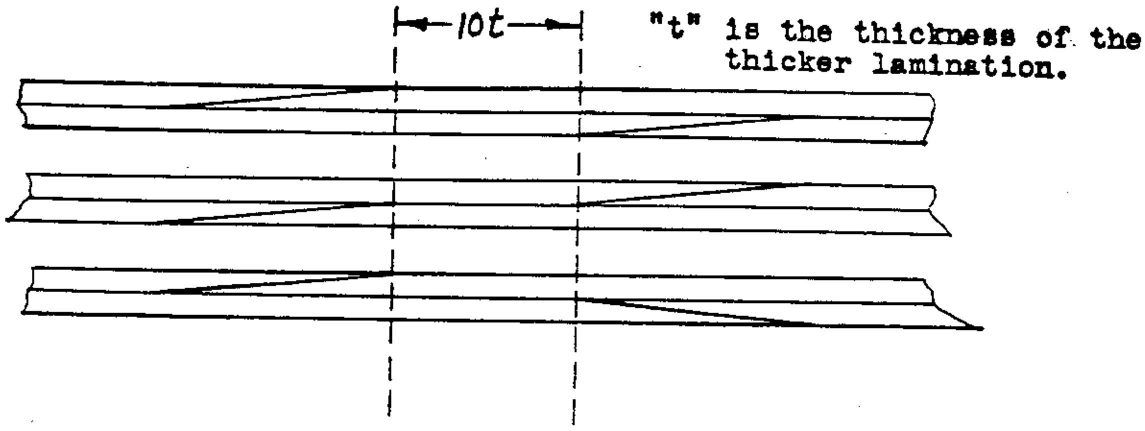


Figure 4–18. Minimum permissible longitudinal separation of scarf joints in adjacent laminations.

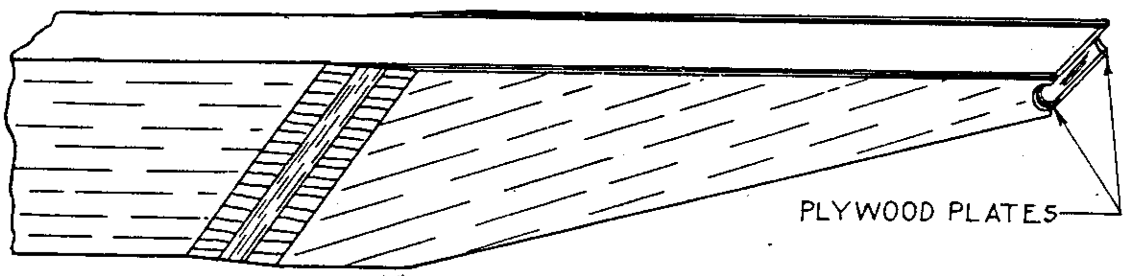


Figure 4–19. Solid wing spar at tip.

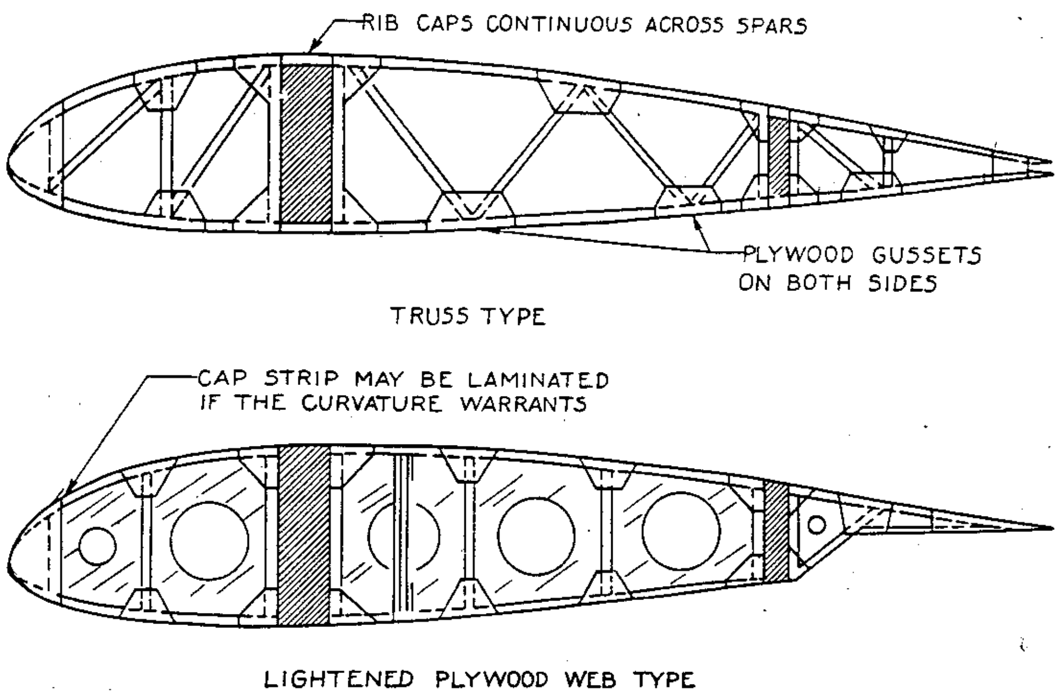


Figure 4–20. Typical wing ribs.

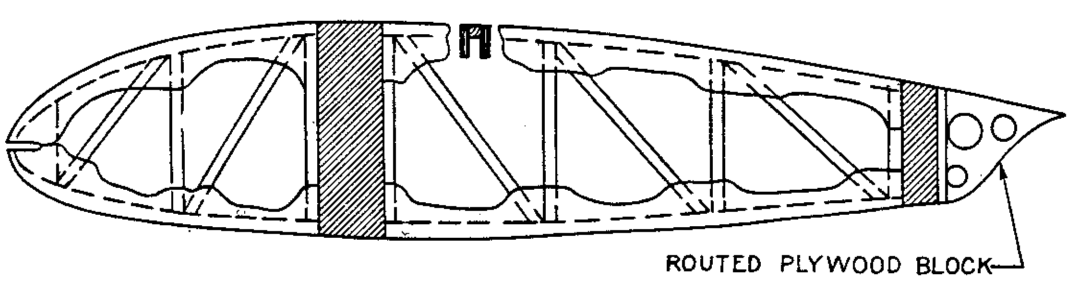


Figure 4–21. Rib employing continuous gussets.

torsional rigidity of a structure such as a wood-framework, fabric-covered control surface (fig. 4–22).

Diagonals loaded in compression are more satisfactory than diagonals loaded in tension since tension diagonals are more difficult to hold at the joints.

4.31. Special Purpose Ribs. Where concentrated loads are introduced, as at landing gear or nacelle attachments, bulkhead-type ribs can be used to advantage. When this is the case, the rib acts as a chordwise beam, and the principles presented in section 4.2 will apply (fig. 4–23).

4.32. Attachment of Ribs to the Structure. In general, ribs are glued to the adjacent structure by means of corner blocks, plywood angles or gussets, or in special cases, by some mechanical means. These are all shown in detail in figures 4–24, 4–25, 4–26, 4–27, 4–34, and 4–39.

Although the attachment of ribs to I-beams may complicate the rib design, many engineers believe that the mechanical shear connection obtained by notching the ribs so that the end may be inserted between the I-beam flanges is an advantage since the shear connection is not dependent upon quality of the glue joint between the rib and the beam shear web. This type of connection is shown in figure 4–25. The web vertical also acts as a stiffener for the beam shear web and as a flange spreader.

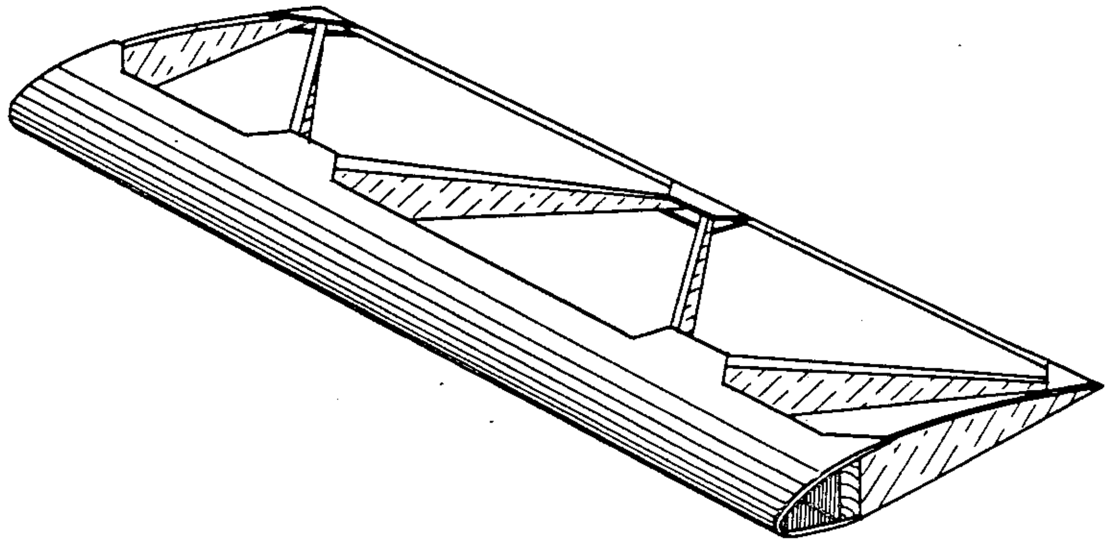


Figure 4–22. Control surface employing canted ribs.

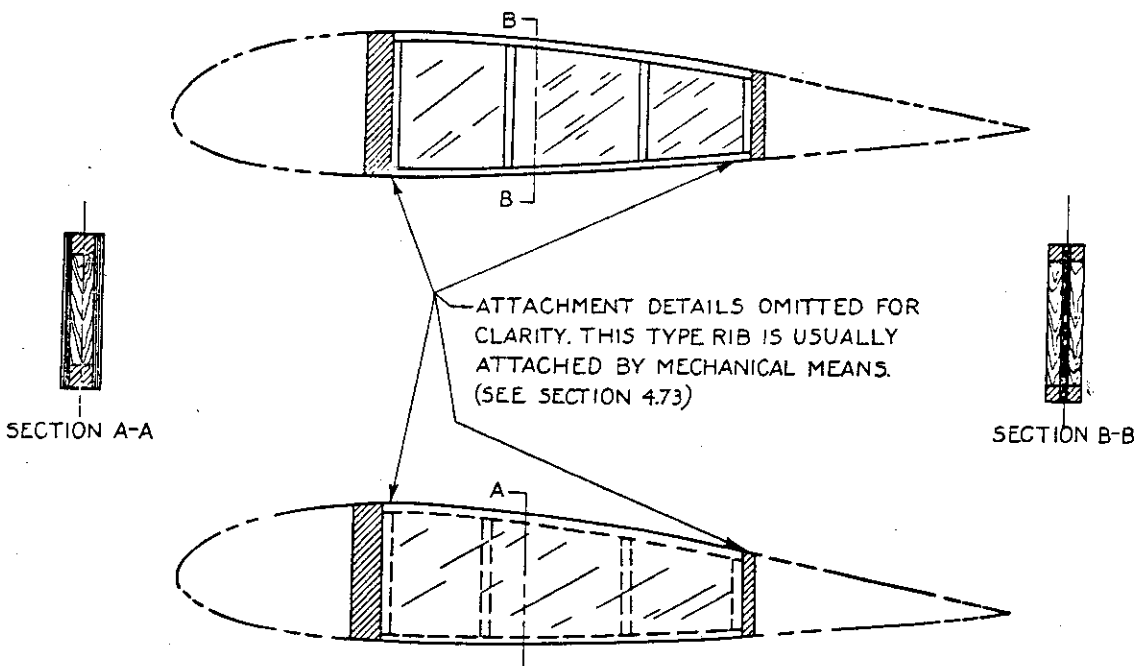


Figure 4–23. Special purpose ribs.

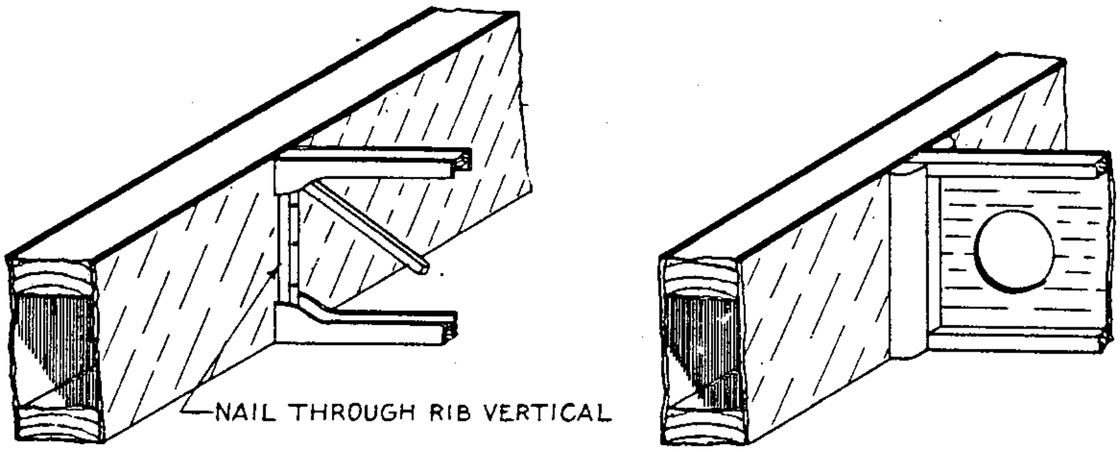


Figure 4–24. Typical rib attachments to flush surface beams.

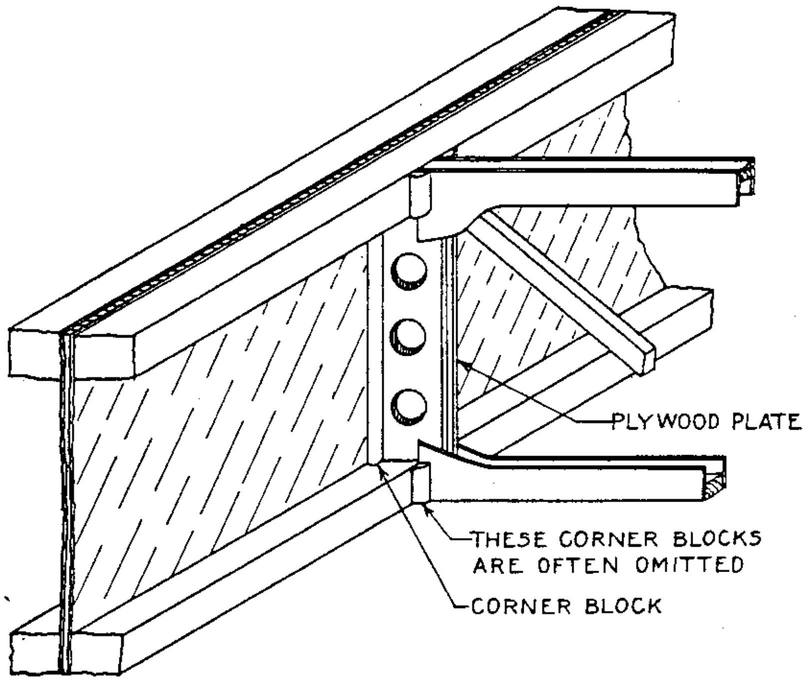


Figure 4–25. Typical rib attachment to I-beam.

The end rib verticals of plywood web type ribs are sometimes preassembled to plain rectangular spars to act as locating members for rib-to-spar assembly. This is shown in figure 4–26. Preassembled locating corner blocks might also be used to advantage in other types of rib-to-spar attachments if care is taken to provide sufficient backing for plywood webs to which corner blocks are being glued so that sufficient gluing pressure can be obtained.

Canted ribs may be attached to beam members by beveling the ends of the ribs or by using corner blocks as shown in figure 4–27.

4.4. Frames and Bulkheads

4.40. Types of Frames and Bulkheads. No one type of frame or bulkhead seems to be the best for all types of loading, but the laminated ring is probably the best type for use as an intermediate stiffening frame. Frames or bulkheads are usually made of formed laminated wood, cut or routed from plywood, or are a combination of the two (fig. 4–28).

4.41. Glue Area for Attachment of Plywood Covering. Care must be taken when using the routed plywood type of bulkhead that the plywood edge provides sufficient gluing area for the skin. It is often necessary to glue solid wood to the face of the ring near its edge to provide additional gluing surface. This is illustrated in figure 4–29.

4.42. Reinforcements for Concentrated Loads. When concentrated loads are carried into a frame it may be desirable to scarf in some

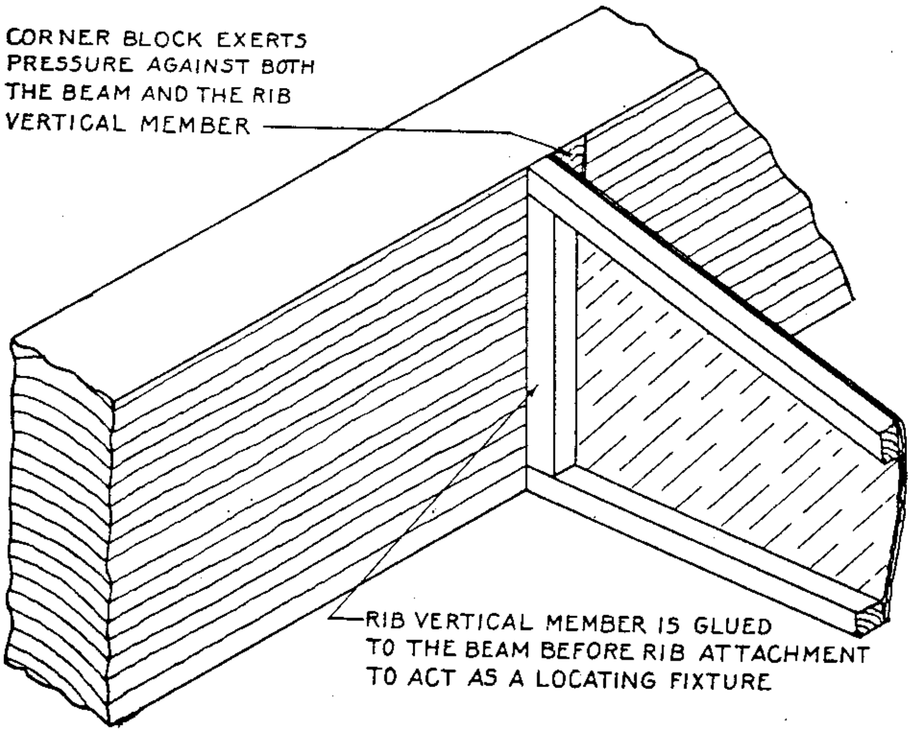


Figure 4–26. Use of rib vertical as locating fixture.

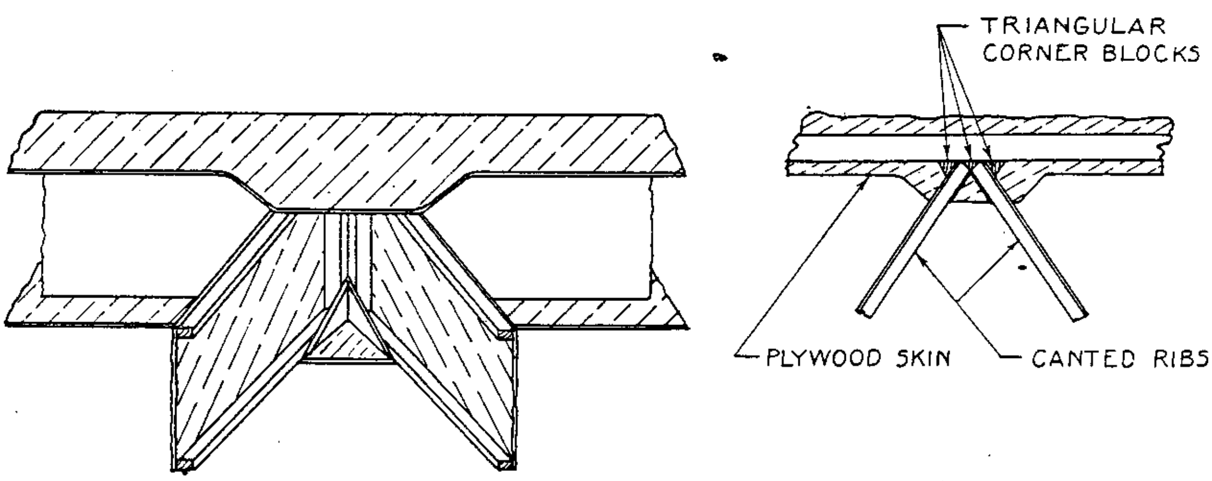


Figure 4–27. Typical canted rib to spar attachment.

high-density material and brace the frame with a plywood web or solid truss members.

4.5. Stiffeners

4.50. General. The terms "stringer," "stiffener," and "intercostal" are often used interchangeably. In the following discussion, "stringer" will refer to members continuous through ribs and frames and "intercostal" will refer to members terminating at each rib or frame. The term "stiffener" will not be used, since both stringers and intercostals act as stiffeners.

4.5.1. Attachment of Stringers. Ribs or frames must be notched if stringers are used. A method of reinforcing these notches and fastening the stringers to the rib or frame is illustrated in figure 4–30. Attachments may also be made by one of the methods shown in figure 4–34.

4.52. Attachment of Intercostals. All intercostals should be firmly attached to ribs or frames. Figure 4–31 illustrates the undesirable practice of terminating intercostals some distance from the rib or frame. This usually results in cleavage along the glue line starting at the free end of the intercostal. It is better to butt the stiffeners to the rib or frame and fasten them with saddle gussets as illustrated in figure 4–32 or by one of the attachments shown in figure 4–34.

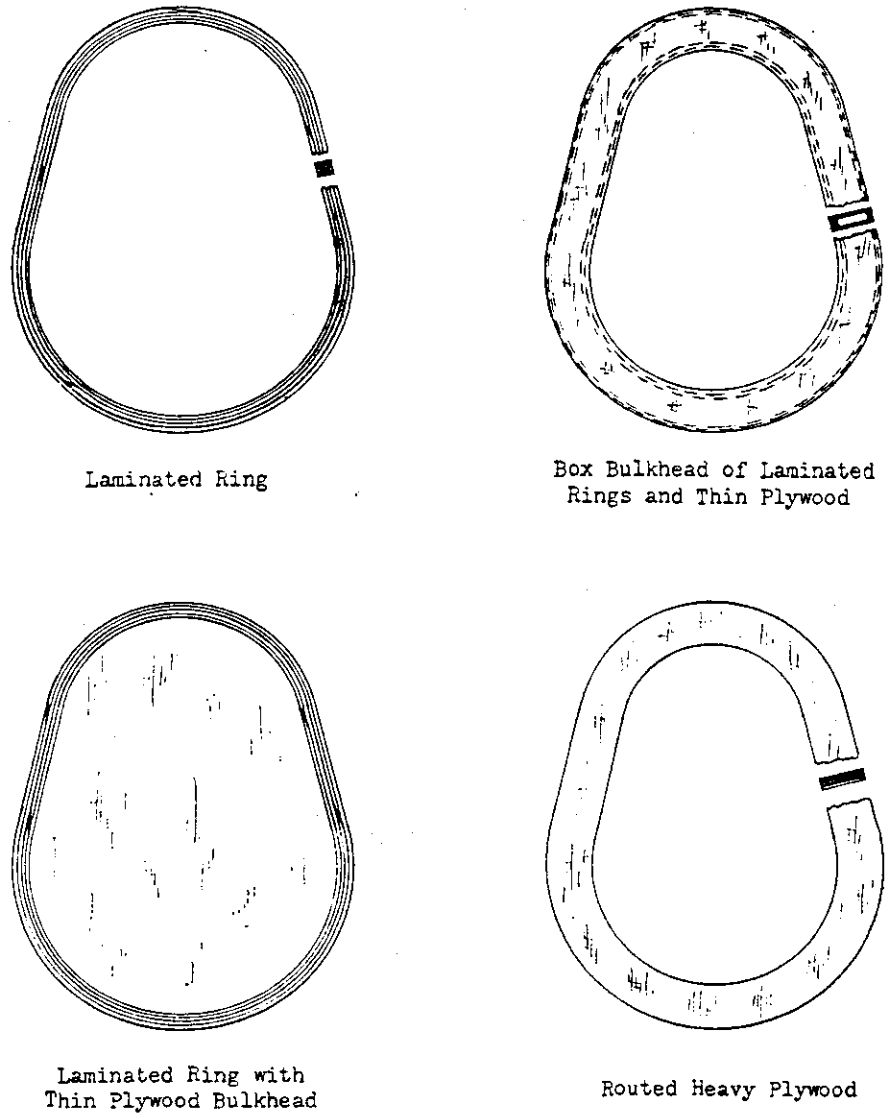


Figure 4–28. Typical frames.

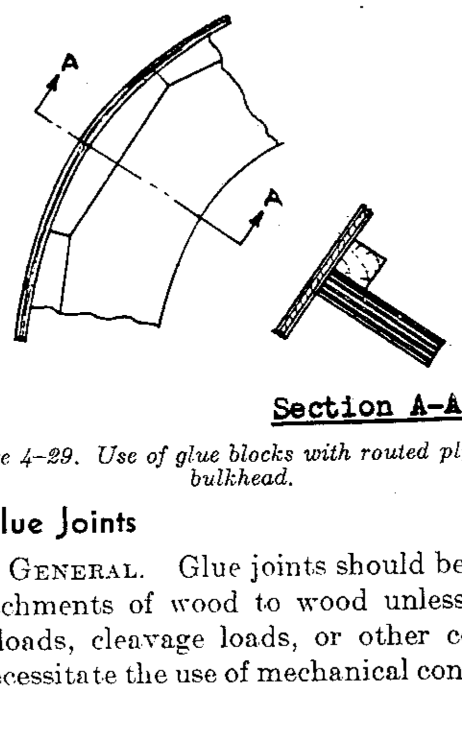


Figure 4–29. Use of glue blocks with routed plywood bulkhead.

4.6. Glue Joints

4.60. GENERAL. Glue joints should be used for all attachments of wood to wood unless concentrated loads, cleavage loads, or other considerations necessitate the use of mechanical connections.

4.61. ECCENTRICITIES. Eccentricities and tension components should be avoided in glue joints by means of careful design. Figure 4-33 illustrates an example of an eccentricity and a method of avoiding it.

4.62. AVOIDANCE OF END GRAIN JOINTS. End grain glue joints will carry no appreciable load. Strength is given to such a joint by using corner blocks or gussets as shown in figure 4–34. These sketches are typical of joints encountered in joining rib members, in attaching ribs to beams or intercostals to frames, or any other similar application.

4.63. GLUING OF PLYWOOD OVER WOOD-PLYWOOD COMBINATIONS. Many secondary glue joints must be made between plywood covering and wood-plywood structural members having plywood edges appearing on the surface to be glued. Wood-plywood beams or wing ribs em-

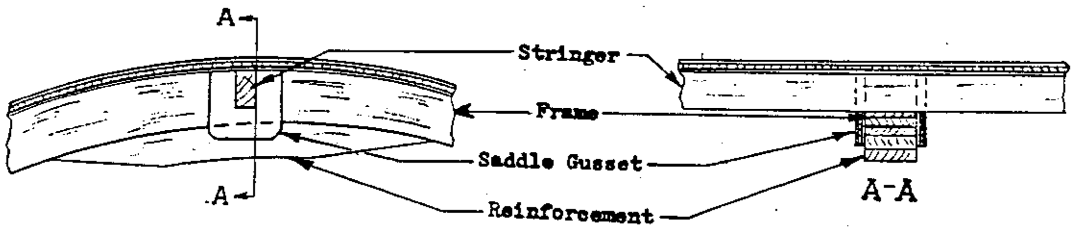


Figure 4–30. Stringer through frame joint.

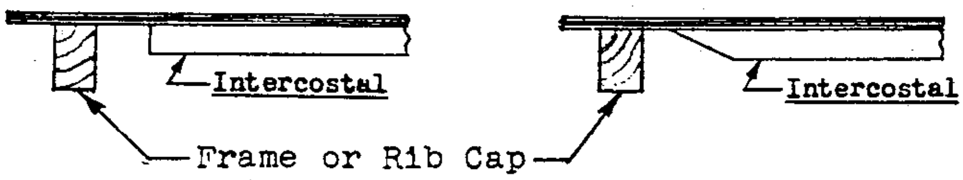


Figure 4–31. Poor method of intercostal attachment.

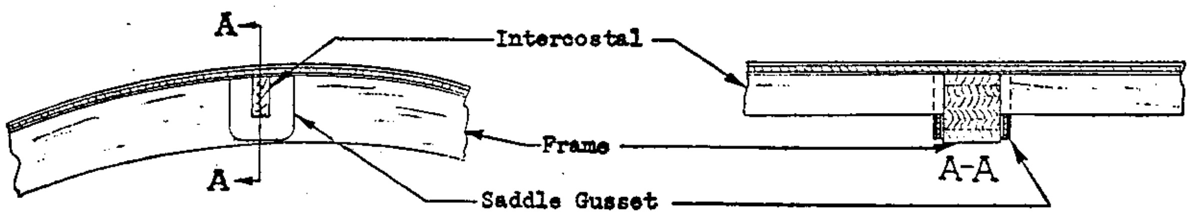


Figure 4–32. Acceptable method of intercostal attachment.

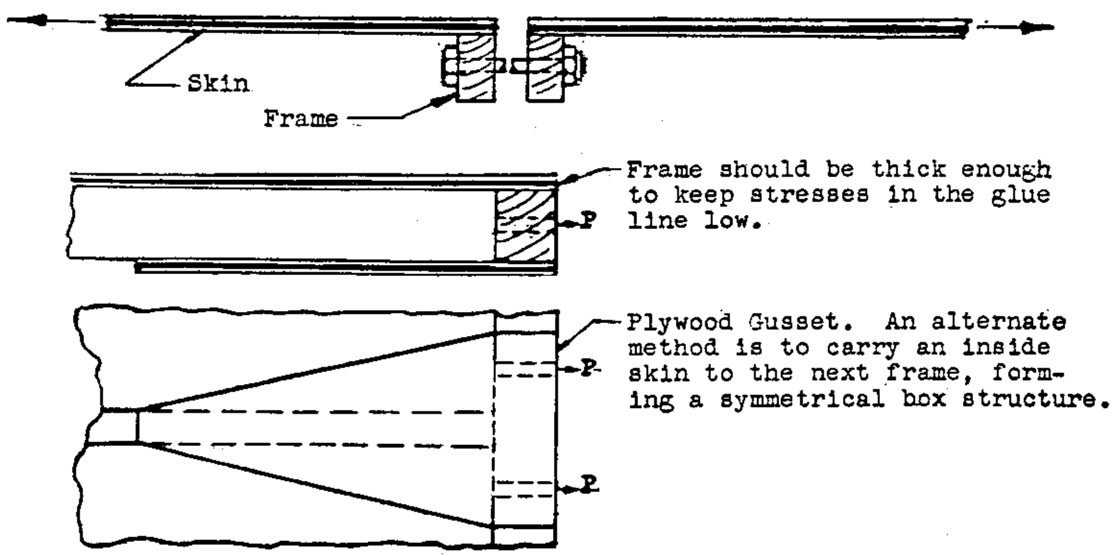


Figure 4–33. Joint in a shell structure.

ploying continuous gussets are examples of such members. The plywood edge has a tendency to project above the surface thereby preventing contact between the plywood covering and the wood portion of the wood-plywood surface. This condition can be the result of differential shrinkage between the wood and plywood or may be caused by the surfacing machine having a different effect cutting across the grain of the plywood from cutting parallel to the grain of the wood. Figure 4–35 shows this condition and shows how it can be eliminated by beveling the edges of the plywood.

4.64. Gluing of High-Density Material. Better glue joints can be obtained between a high-density material and a relatively soft wood if the surface of the high-density material is sanded before gluing. The purpose of sanding is to remove the glazed surface present on high-density material and present on some plywoods. Satisfactory compreg-to-compreg joints can be made if

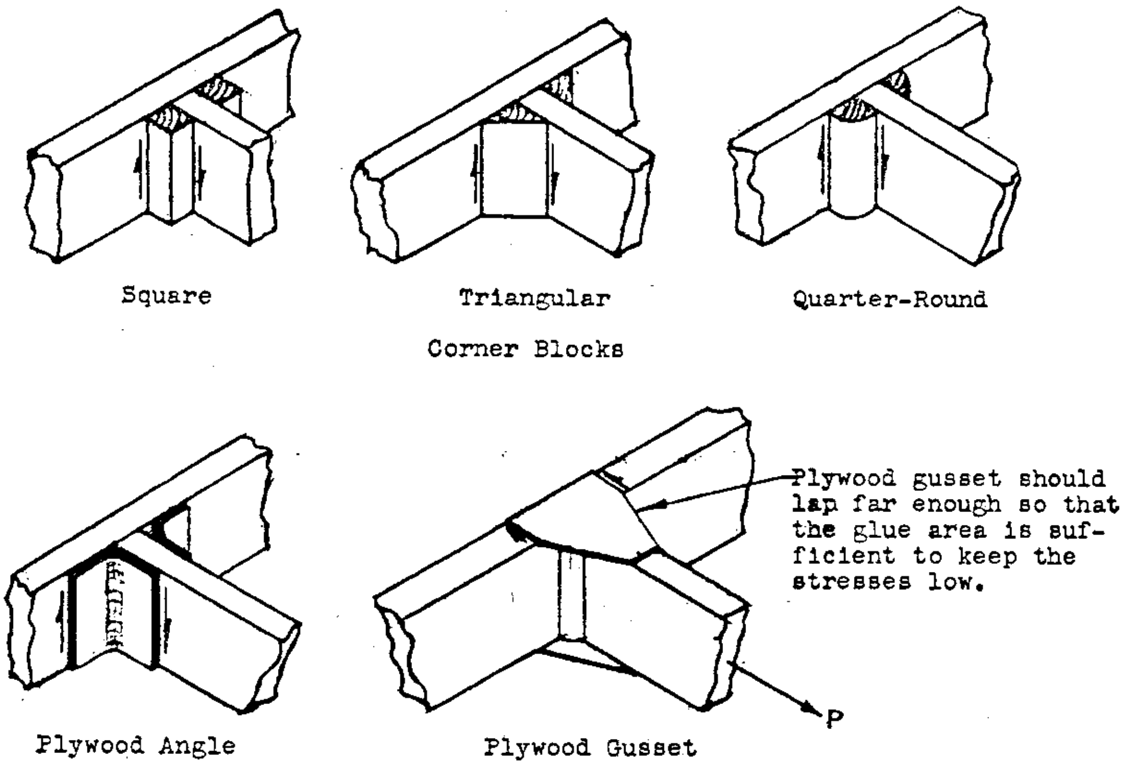


Figure 4–34. Typical reinforcement of end grain joints.

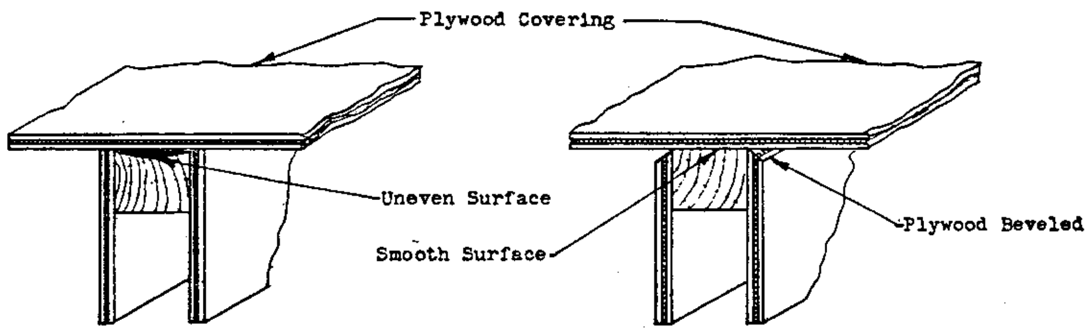


Figure 4–35. Beveling of plywood webs and gussets.

both surfaces are machined perfectly flat immediately prior to gluing.

4.7. Mechanical Joints

4.70. General. Mechanical joints in wood are usually limited to types employing aircraft bolts. Since bolts in wood can carry a much higher load parallel to the grain of the wood than across the grain, it is generally advantageous to design a fitting and its mating wood parts so that the loads on the bolts are parallel to the grain. The use of a pair of bolts on the same grain line, carrying loads perpendicular to the grain and oppositely directed, is likely to increase the tendency to split. When a long row of bolts is used to join two parts of a structure, consideration should be given to the relative deformation of the parts, as explained in section 4.82.

4.71. Use of Bushings. Bushings are often used in wood to provide additional bearing area and to prevent crushing of the wood when bolts are tightened (fig. 4–36). When bolts of large L/D (length/diameter) ratio are used, or when bolts are used through a member having high-density plates on the faces, plug bushings may be used to advantage.

4.72. Use of High-Density Material. Wherever highly concentrated loads are introduced, greater bearing strength can be obtained by scarfing-in high-density material (sec. 4.63). Some high density materials are quite sensitive to stress concentrations and the possibility of the serious effects of such stress concentrations should be considered when large loads must be carried through the high-density material.

Wherever metal fittings are attached to wood

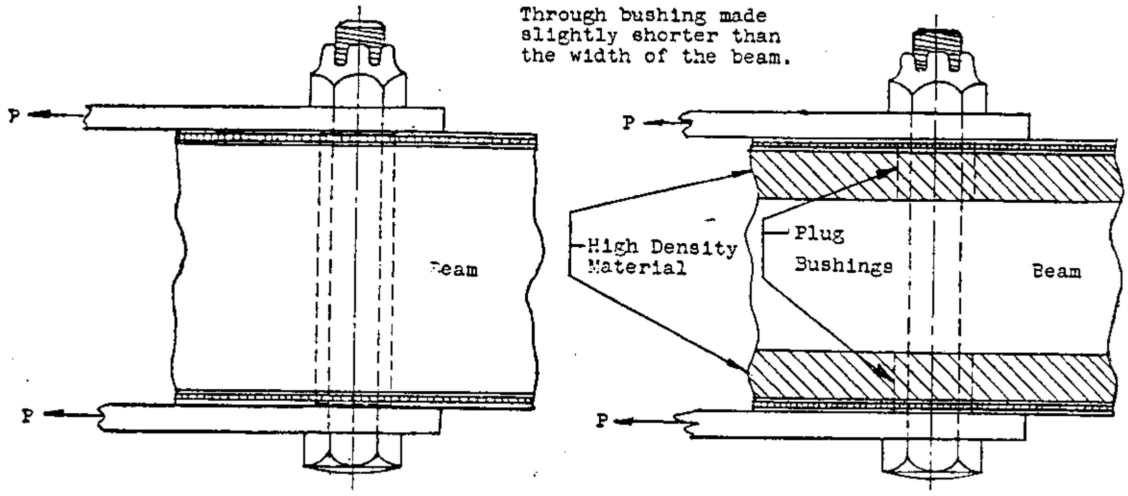


Figure 4–36. Types of bushings.

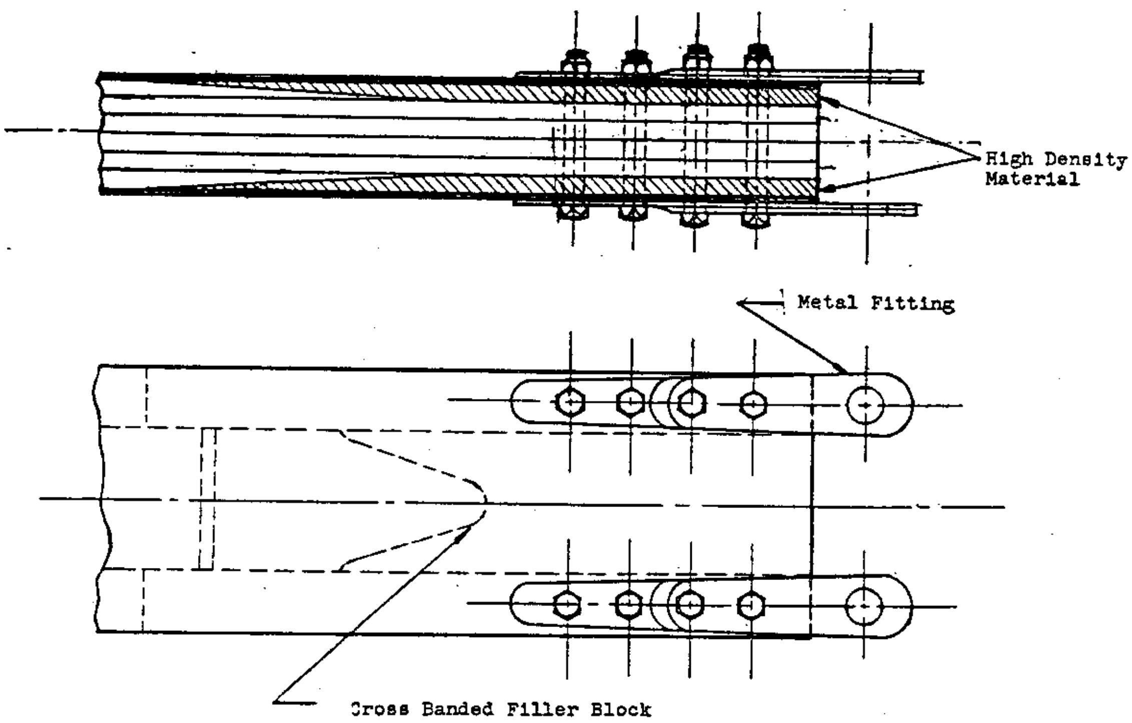


Figure 4–37. Typical wing beam attachment.

members, it is generally advisable to reinforce the wood against crushing by the use of high-density bearing plates (fig. 4–37), and to use a coat of bitumastic or similar material between the wood and metal to guard against corrosion. Cross banding of these plates will help to prevent splitting of the solid wood member.

4.73. Mechanical Attachment of Ribs. When ribs carry heavy or concentrated loads it is sometimes desirable to insure their attachment by use of mechanical fastenings (fig. 4–39).

4.74. Attachment of Various Types of Fittings. Fittings should have wide base plates to prevent crushing at edges. Wood washers have a tendency to cone under tightening loads. Where possible, it is desirable to use washer plates for bolt groups, as illustrated in figure 4–40, but if washers are used, a special type for wood, AN–970 or equivalent, are necessary to provide sufficient bearing area.

Clamps around wood members should be constructed so that they can be tightened symmetrically (fig. 4–41).

4.75. Use of Wood Screws, Rivets, Nails, and Self-Locking Nuts. Wood screws and rivets are sometimes used for the attachment of secondary structure but should not be used in connecting primary members. Wood screws have

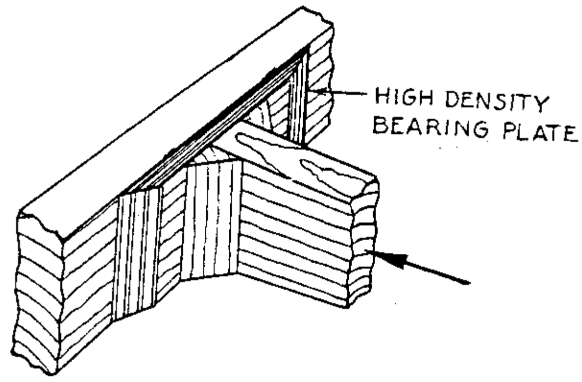


Figure 4–38. Distribution of crushing loads.

been successfully used to prevent cleavage of plywood skin from stringers in some skin-stringer applications. Nails should never be used in aircraft to carry structural loads.

Self-locking nuts of approved types designed for use with wood and plywood structures are preferable to plate or anchor nuts. When the latter type is used, however, attachment may be made to the structure with wood screws or rivets provided that care is taken not to reduce the strength of load-carrying members. Riveting through wood is always questionable because of the danger of crushing the wood under the rivet heads and the possibility of bending the shank while bucking the rivet. Also, there is no way of tightening the joint when dimensional changes from shrinkage occur.

4.8. Miscellaneous Design Details

4.80. Metal to Wood Connections. Metal to wood connections are complicated by an inherent weakness of all untreated wood—low shear and bearing strength. Sections 4.6 and 4.7 present various methods of minimizing this drawback.

Another way of improving the efficiency of wood structures is to keep the number of joints to a minimum. For example, when other design considerations will permit, a one-piece wood wing is desirable; when this is not permissible, the wing joint should be placed as far outboard as possible so that the fitting loads will be low.

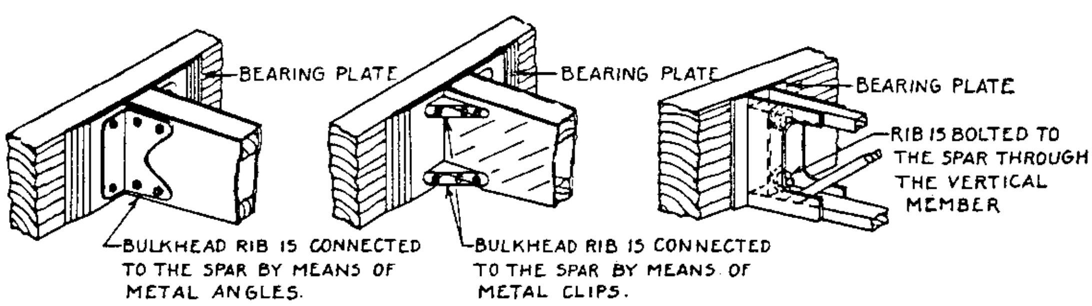


Figure 4–39. Mechanical attachment of ribs.

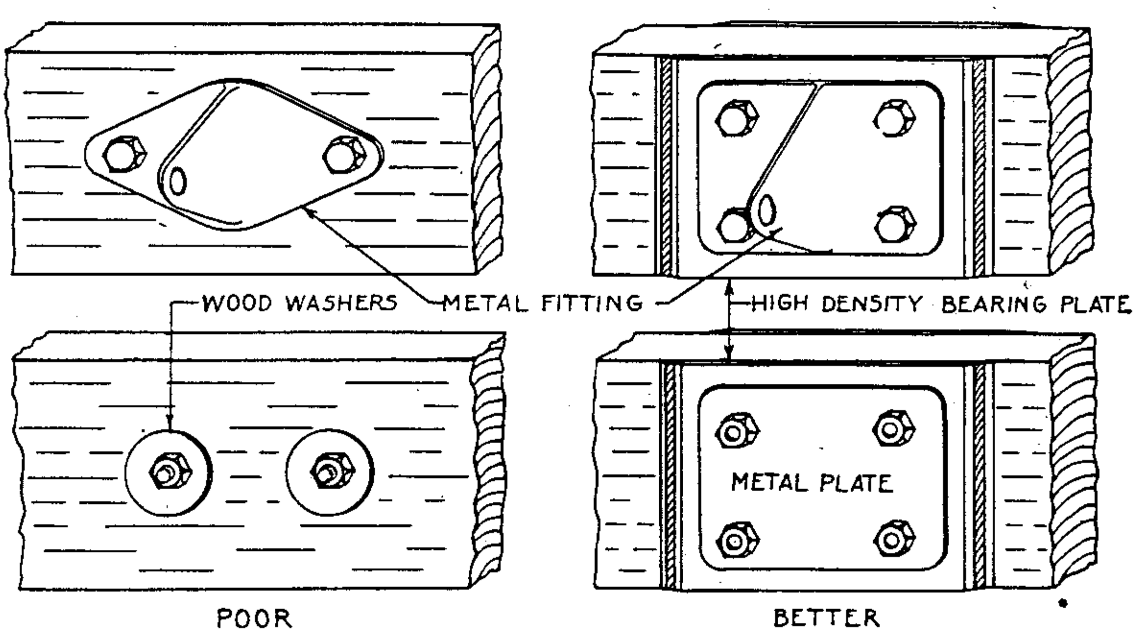


Figure 4–40. Example of control surface hinge fitting attachment.

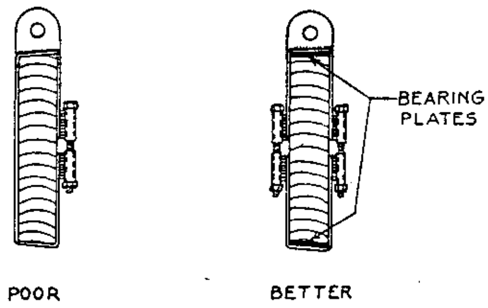


Figure 4–41. Installation of clamp fittings.

4.81. Stress Concentrations. Since wood in tension has practically no elongation between the proportional limit and the ultimate strength, there is little of the "internal adjustment" common to metal structures. Stress concentrations, therefore, become more critical and, for efficient design, must be held to a minimum. The fact that compreg and similar materials are very sensitive to stress concentrations should be carefully considered when these materials are used.

4.82. Behavior of Dissimilar Materials Working Together. When materials of differing rigidities, such as normal wood, compreg, or metal fittings, are fastened together for a considerable distance and are under high stress, consideration should be given to the fact that the fastening causes the total deformation of all materials to be the same. A typical example is a long metal strap bolted to a wood spar flange for the purpose of taking the load out of the wood at a wing joint. In order that the load be uniformly distributed among the bolts, the ratio of the stress to the modulus of elasticity should be the same for both materials at every point. This may be approximated in practical structures by tapering the straps and the wood in such a manner that the average stress in each (over the length of the fastening) divided by its modulus of elasticity gives the same ratio.

When splicing high-density materials to wood, or in dropping off bearing plates, the slope of the scarf should be less steep than the slope allowed for normal wood.

4.83. Effects of Shrinkage. When the moisture content of a piece of wood is lowered its dimensions decrease. The dimensional change is greatest in a tangential direction (across the fibers and parallel to the growth rings), somewhat less in a radial direction (across the fibers and perpendicular to the growth rings), and is negligible in a longitudinal direction (parallel to the fibers). For this reason a flat-grained board will have a greater change in width for a given moisture content change than an edge-grained board. Flat-grained boards also have a greater tendency to warp than do edge-grained boards.

These dimensional changes can have several deleterious effects upon a wood structure and the designer must study each case to determine which effects are most harmful, and which are the most satisfactory methods of minimizing them. Loosening of fittings and wire bracing are common results of shrinkage. Checking or splitting of wood members frequently occurs when shrinkage takes place in members that are restrained against dimensional change. Restraint is sometimes given by metal fittings and quite often by plywood reinforcements since plywood shrinkage is roughly only 1/20 of cross grain shrinkage of solid wood.

A few of the methods of minimizing these shrinkage effects are:

1. Use bushings that are slightly short so that when the wood member shrinks the bushings do not protrude and the fittings may be tightened firmly against the member (fig. 4–36).
2. Place the wood so that the more important face, in regard to maintaining dimension, is edge-grained. For example, solid spars are required to be edge-grained on their vertical face so that the change in depth is a minimum.
3. Wood members can be reinforced against checking or splitting by means of plywood inserts or cross bolts (fig. 4–42). Care should be taken to avoid constructions that introduce cleavage (cross-grain) loads when shrinkage occurs.
4. Plywood face plates should be dropped off gradually either by feathering or by shaping so that the cleavage loads at the edge of the plywood are minimized when shrinkage occurs (fig. 4–43).

4.84. Drainage and Ventilation. Wood structures must be adequately drained to insure a normal length of service life. This applies to box spar sections as well as all low portions of wings and fuselages. The usual method is to drain each compartment separately as illustrated in figure 4–44. Another acceptable method is to drain from one compartment to another until the lowest compartment is reached, or structural requirements prohibit further internal drainage, before drainage holes to the exterior are bored. This method is illustrated in figure 4–45.

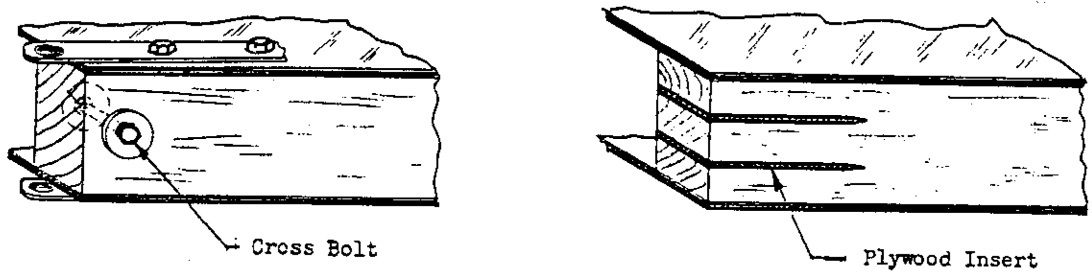


Figure 4–42. Protection against splitting.

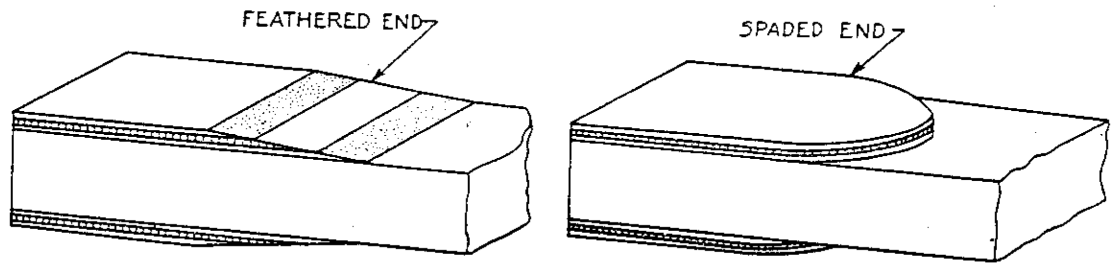


Figure 4–43. Tapering of face plates.

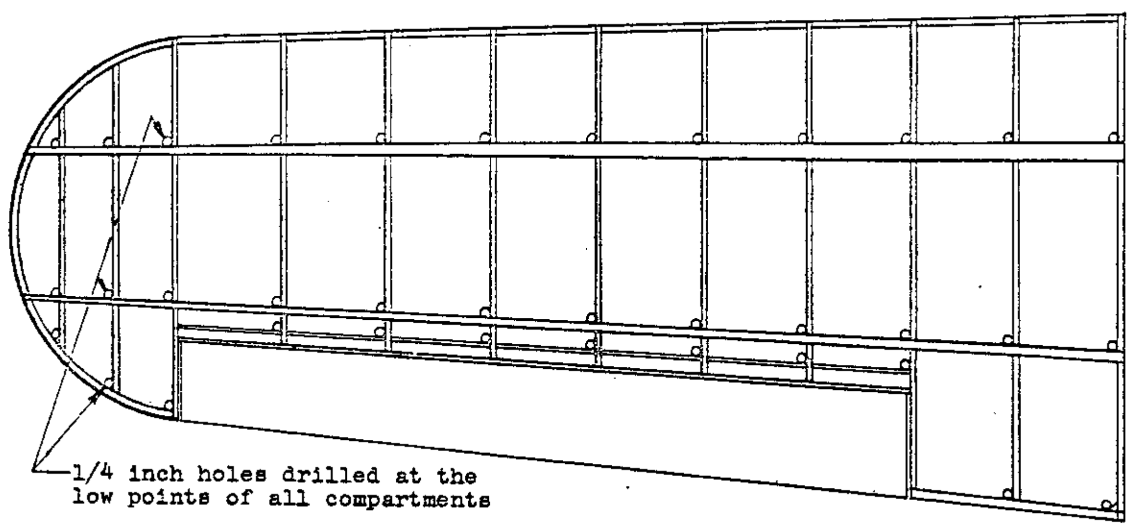


Figure 4–44. Drainage diagram of wing, direct method.

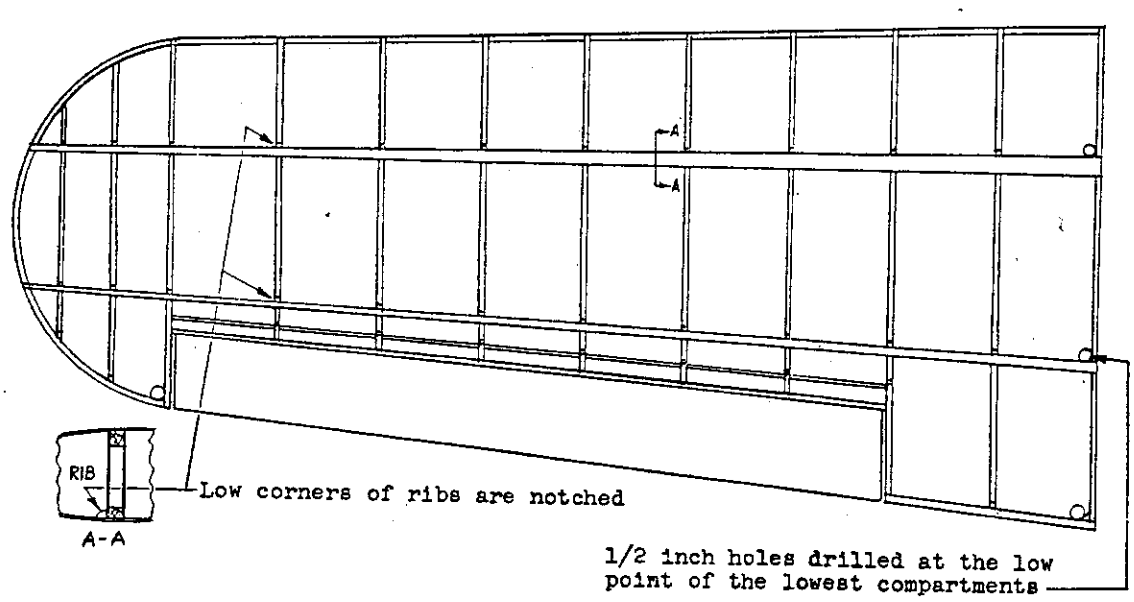


Figure 4–45. Drainage diagram of wing, internal method.

Service experience indicates that drainage holes for individual compartments should be not less than one-quarter inch in diameter, with three-eighths inch being preferable. Drainage holes to the exterior used with the internal drainage system should probably be somewhat larger. If the internal drainage system is used it is suggested that the inter-compartment drainage holes be inspected after the internal finish has been applied to make sure that the finish has not clogged the internal drain holes. This will necessitate attaching the top skin last.

Drain holes are usually drilled from the external surface so that the splintering does not mar the external finish. After drilling drain holes, all splinters should be carefully removed from the inner surface, and the edges of the holes should be sanded lightly and protected by the application of several coats of spar varnish. It is common practice, in order to avoid damage to structural members by the drill, to drill drainage holes an appreciable distance from the low corner of a compartment. This practice must be avoided and some method of insuring proper location of drain holes at the *actual* low points must be developed by the aircraft manufacturer that will not only prevent damage to the framework but will also provide complete drainage of the structure.

It is, therefore, recommended that proof of the adequacy of the drainage system chosen be demonstrated by setting up the structure, with the top cover removed, in a position corresponding to its attitude when the airplane is resting on the ground. Water is then poured into the structure and the actual performance of the drainage system observed.

Careful design to prevent entry of water into the structure is equally important. Careful location of all openings and use of boots and gaskets should be considered. If interiors do happen to get wet, good ventilation will accelerate the drying. Marine grommets have been suggested for use with external drain holes in wing, tail, and control surfaces. This type of grommet produces a suction or scavenging action in flight and also protects the holes themselves from direct splash during taxiing on wet or muddy fields. Periodic inspection and cleaning of drainage holes covered with marine grommets, however, may be difficult.

4.85. Internal Finishing. It is recognized that applying finish to the inner surfaces of the closing panels of plywood-covered structures is a difficult problem. The usual method, other than dipping, is to mask off the locations of secondary glue areas prior to the application of finish to the surface, for wood coated with a protective finish cannot be glued. This is a time-consuming operation, and after the plywood covering is finally fitted into place, the film of finish usually stops short of the intersection lines between the plywood covering and framework. These are the very places where the finish is needed most if water does accumulate in the interior.

Wood-rotting organisms can act only if the moisture content of the wood is above approximately 20 to 25 percent. Although finishes will not prevent moisture content changes in wood, they will retard such changes so that the wood moisture content will not follow the rapid changes in atmospheric conditions but only the more gradual changes. Therefore, if wood members are finished, dangerously high moisture contents will be reached in wood aircraft structures only when parts are in contact with standing water since atmospheric conditions that produce high moisture contents are generally of relatively short duration, except in extreme climates such as the tropics, and the retarding effect of the finish may be expected to prevent the wood from attaining a high moisture content within this short period.

In view of the foregoing discussion, it is suggested that consideration be given to the following method of finishing the inner surfaces of plywood-covered assemblies. Since any free water would be in contact with the lower skin almost entirely, the lower wing covering and control surface coverings should be attached to the framework prior to the upper covering. In this way, finish can be applied thoroughly to the lower covering and adjacent framework quite easily after the assembly gluing operation has been completed. Since gaps in the finish on the upper covering along framework members are not so harmful as they would be on the inner surfaces of the lower covering, wider masking strips may be used over secondary glue areas on the upper covering at the time of applying the internal finish, thereby reducing the chance of finished surfaces falling over framework members. Some method of accurately registering the covering should be used.

4.86. External Finishing. Two types of external finish for plywood covered aircraft have been used successfully, the direct-to-plywood finish and the fabric-covered plywood finish. There is little difference in weight between the two systems because the weight of the fabric is offset by the difference in weight between the finishes used in the two systems.

Direct-to-plywood finishes have a tendency to check wherever a glue joint appears on the surface. Checking of the finish is also apt to occur when the grain of the wood tends to raise, as in those softwoods having appreciable contrast between spring and summerwood, such as Douglas-fir. Fabric-covered finishes do not check from these causes.

Light airplane fabric of the type specified in AN-C-83 is the usual material used for the fabric-covered plywood finish system. The fabric provides a better protection from the abrasive action of stones, sand, and other objects kicked up while taxiing than does the direct-to-plywood finish.

Observation of wood airplanes in service has revealed that plywood or fiber plates glued over exposed end grain may act as a moisture trap rather than as a moisture barrier. Several coats of brushed-in aluminized spar varnish are believed to give a much more satisfactory protection to exposed end grain. Exposed end grain should be interpreted to include exposed feathered surfaces.

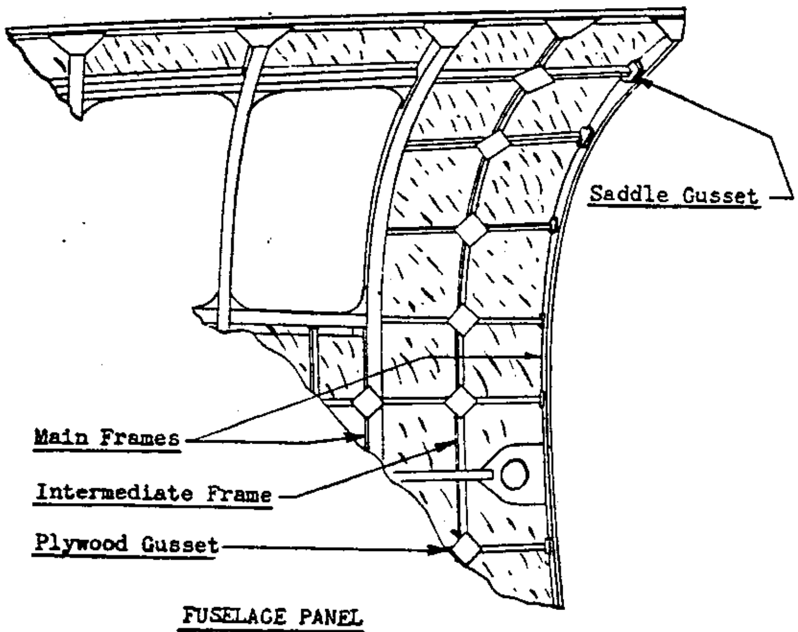
4.87. Selection of Species. Properties other than the usually listed strength and elastic properties should also be considered when selecting a wood for any specific purpose. For example, birch and maple are relatively difficult to glue; yellow-poplar has lower resistance to shock than spruce; Douglas-fir is low in cleavage strength.

4.88. Use of Standard Plywood. From a maintenance viewpoint is is desirable to use only standard plywoods for design so that too great a variety of types will not need to be carried in stock. Table 2-9 lists many of the more common constructions. If one of these is used, the formulas in chapter 2 can be used with greater ease because many of the basic parameters and strength values are given in this table. Two-ply diagonal plywood is considered a special construction by most plywood manufacturers and has the disadvantage of tending to warp because of its unsymmetrical construction.

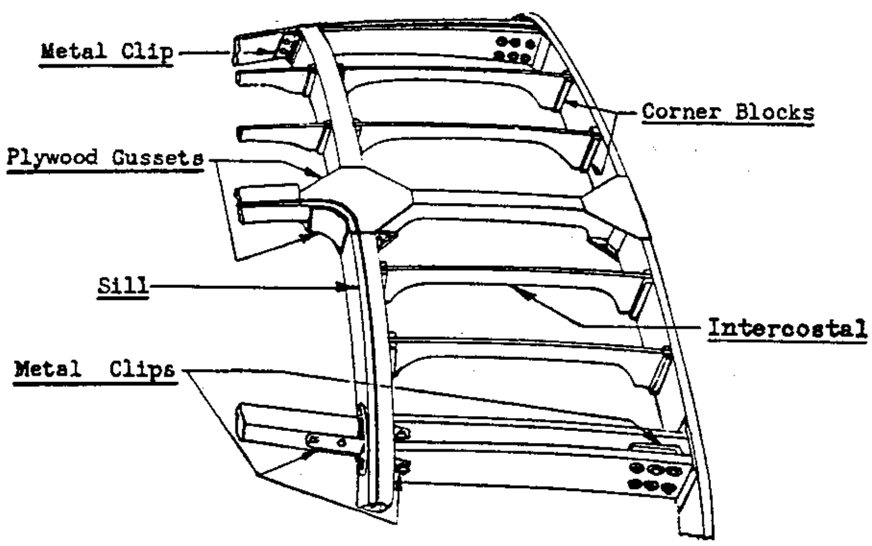
4.89. Tests. Quite often, time and effort may be saved by the use of simple tests in the early stages of the design of complex joints.

4.9. Examples of Actual Design Details

On the following pages several sketches and photographs are presented to show how various manufacturers have treated details encountered in the design of wood aircraft. No effort has been made to label these sketches as either good or poor practice. They are merely presented to show what the industry has done when confronted with specific problems (figs. 4-46 through 4-63).

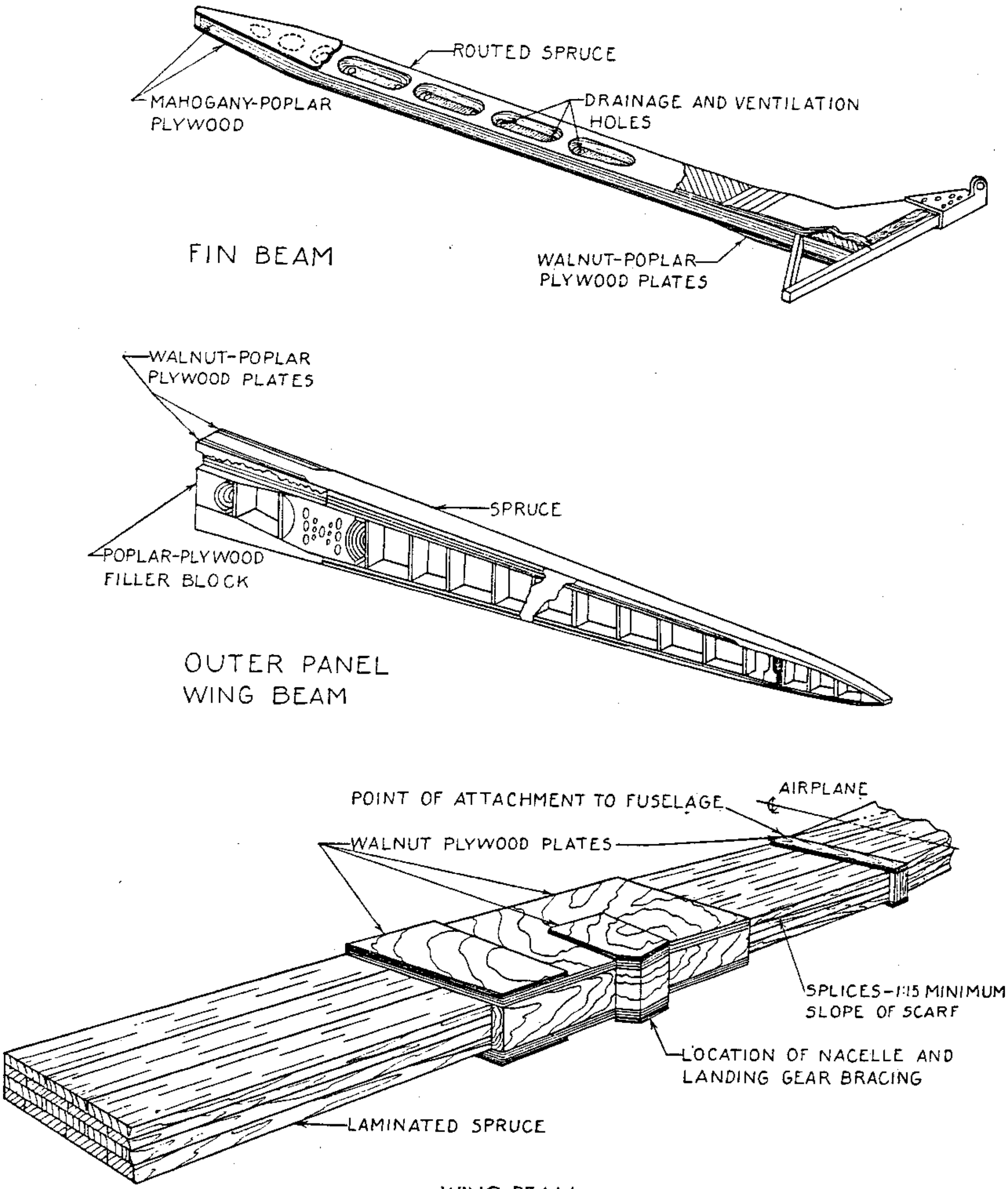


FUSELAGE PANEL



FUSELAGE NOSE SECTION

Figure 4–46. Fuselage framework.



WING BEAM

Showing Blocking and Plating at Location of Application of Concentrated Loans

Figure 4–47. Examples of beams.

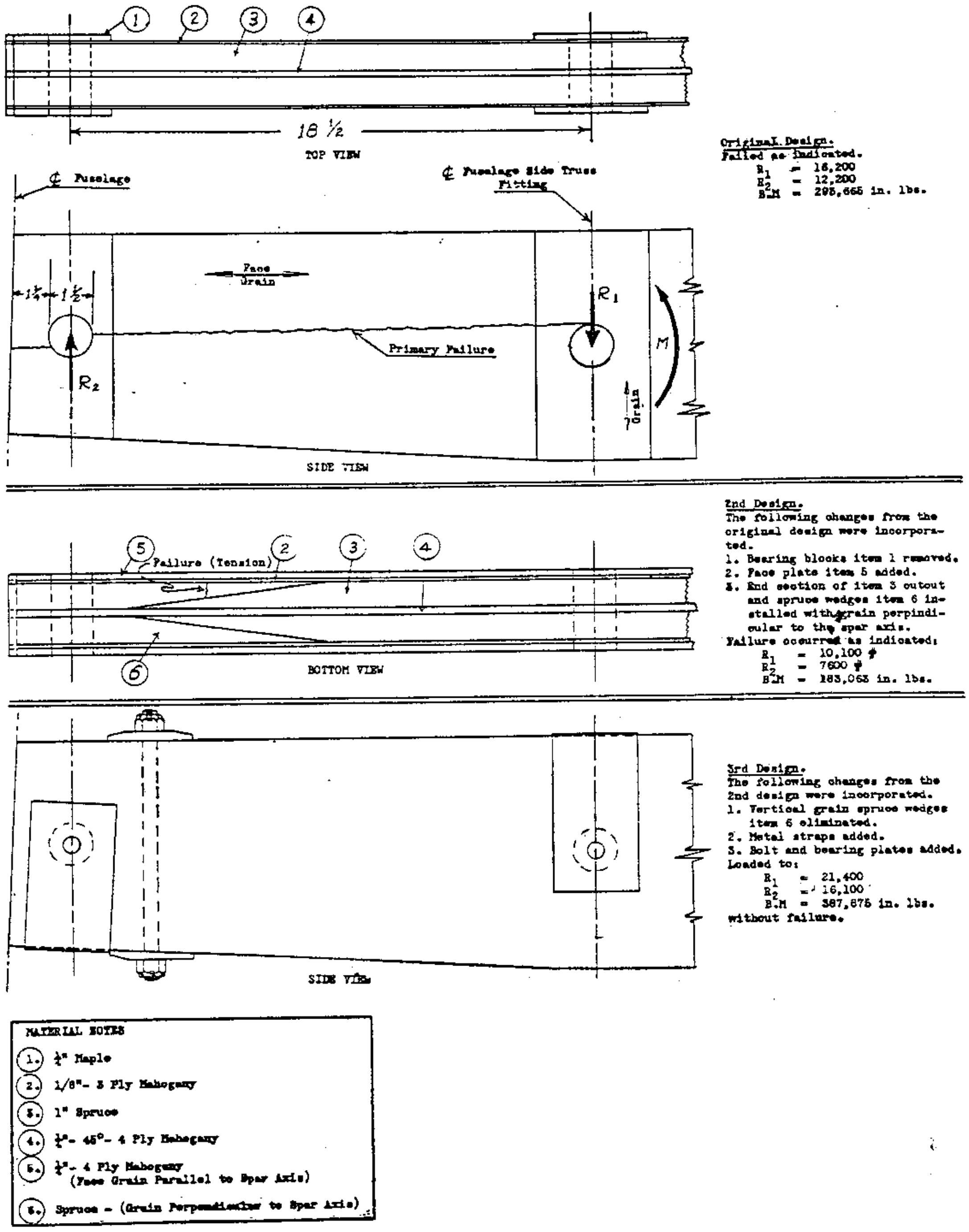
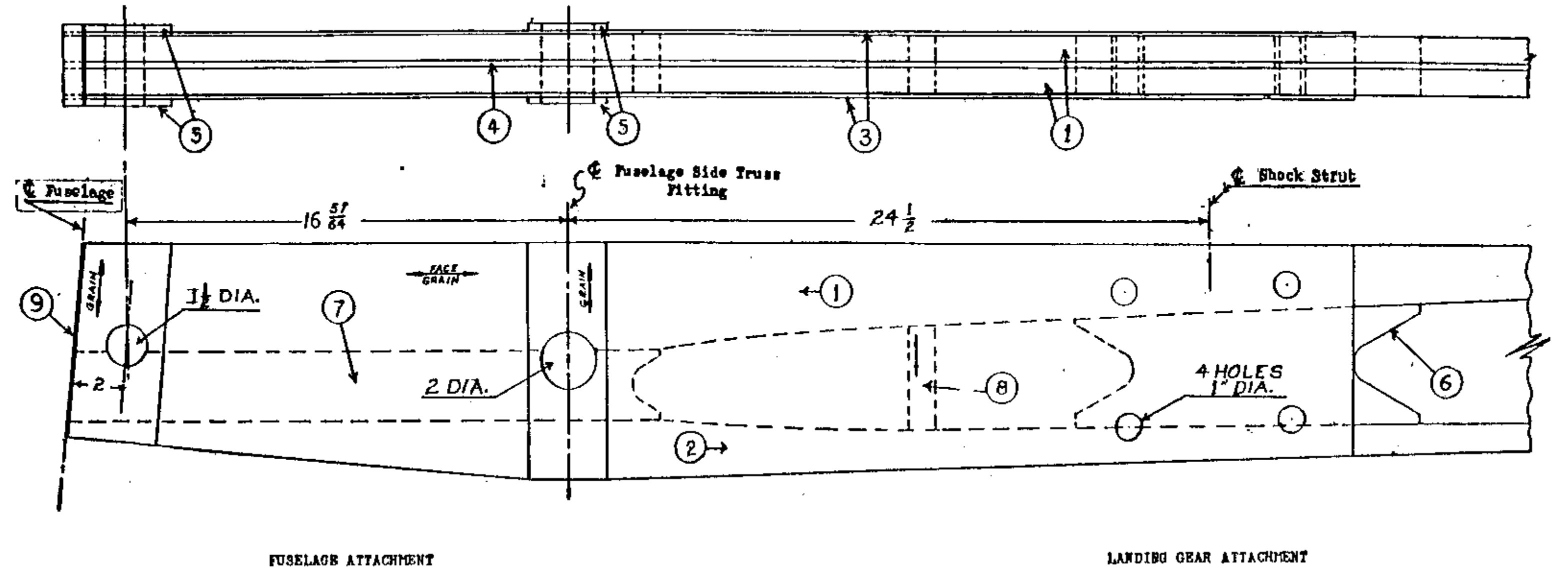


Figure 4–48. Cantilever wood spar at fuselage attachment.



MATERIAL NOTES

1 - 1" spruce
2 - 1" spruce
3 - 1/8"-45°-2 ply mahogany plywood
4 - 1/4"-45°-4 ply mahogany plywood
5 - 1/4"-45°-4 ply mahogany plywood
6 - 1" spruce
7 - 1" spruce
8 - 1" spruce
9 - 1/16"-3 ply mahogany plywood

In a test for a positive flight condition the spar failed at the fuselage attachment under the following loads:

Bending Moment	+ 227,000 in. lbs.
Shear	+ 3,750 lbs.
Axial load	1,380 lbs. (compression)

In a second test the spar was supported in such a manner as to insure failure at or outboard of the fuselage attachment. Failure occurred at the inner landing gear attachment bushings under the following loads:

Bending Moment	+ 174,138 in. lbs.
Shear	+ 4,480 lbs.

Figure 4–49. Spar details for low-wing monoplane, single spar construction. Gross weight, approximately 1,300 pounds.

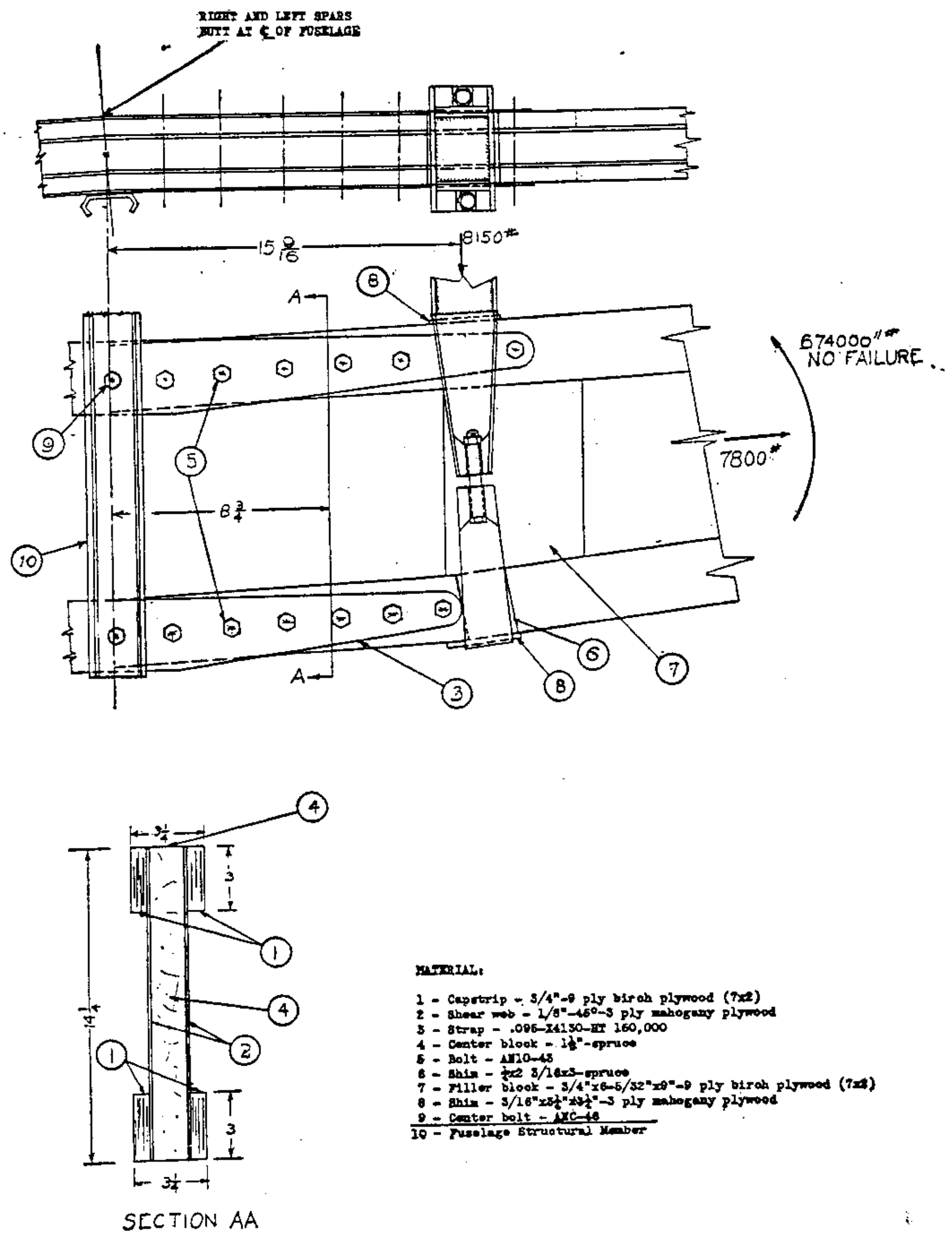


Figure 4–50. Spar details at root section and fuselage attachment.

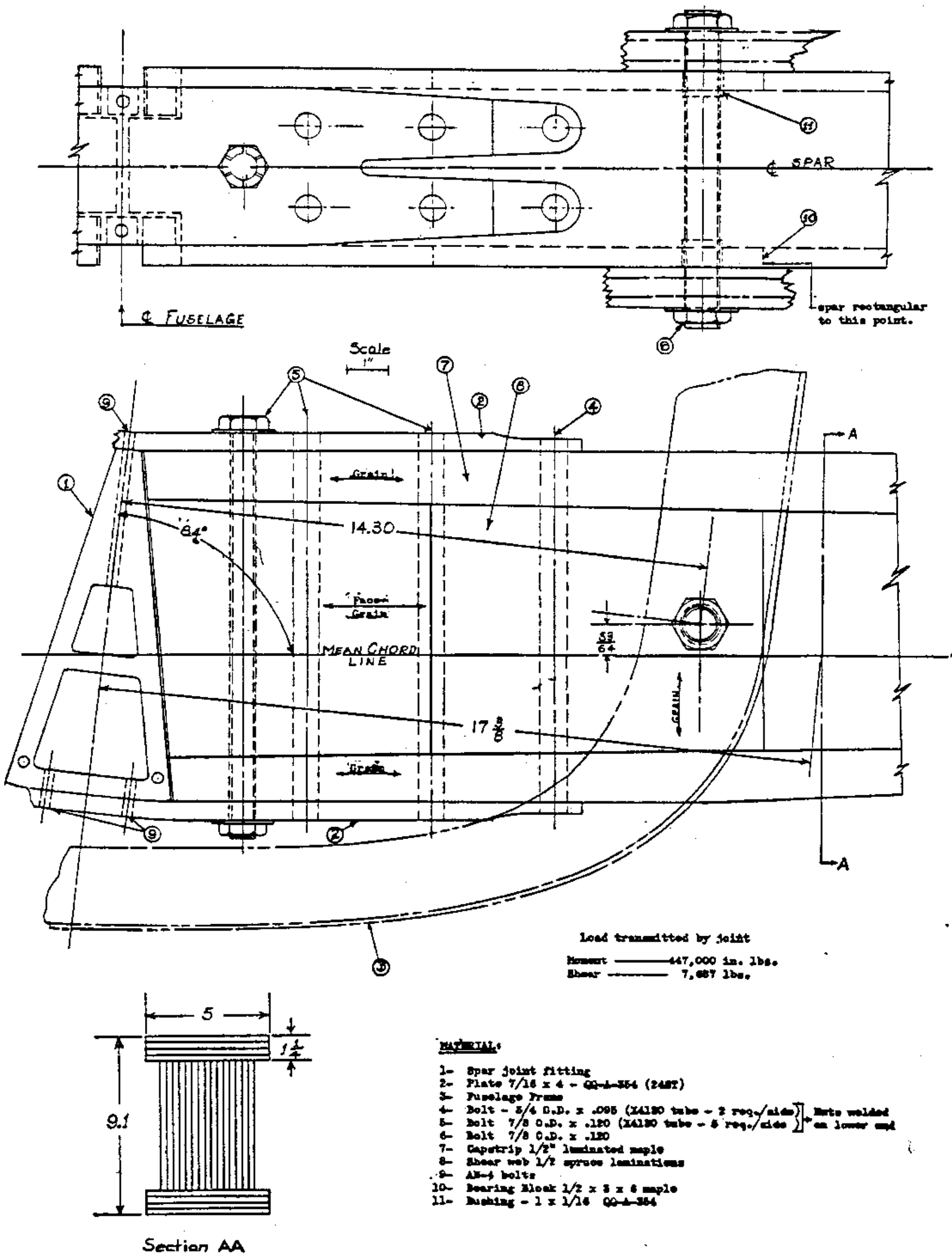


Figure 4–51. Further spar details at root section and fuselage attachment.

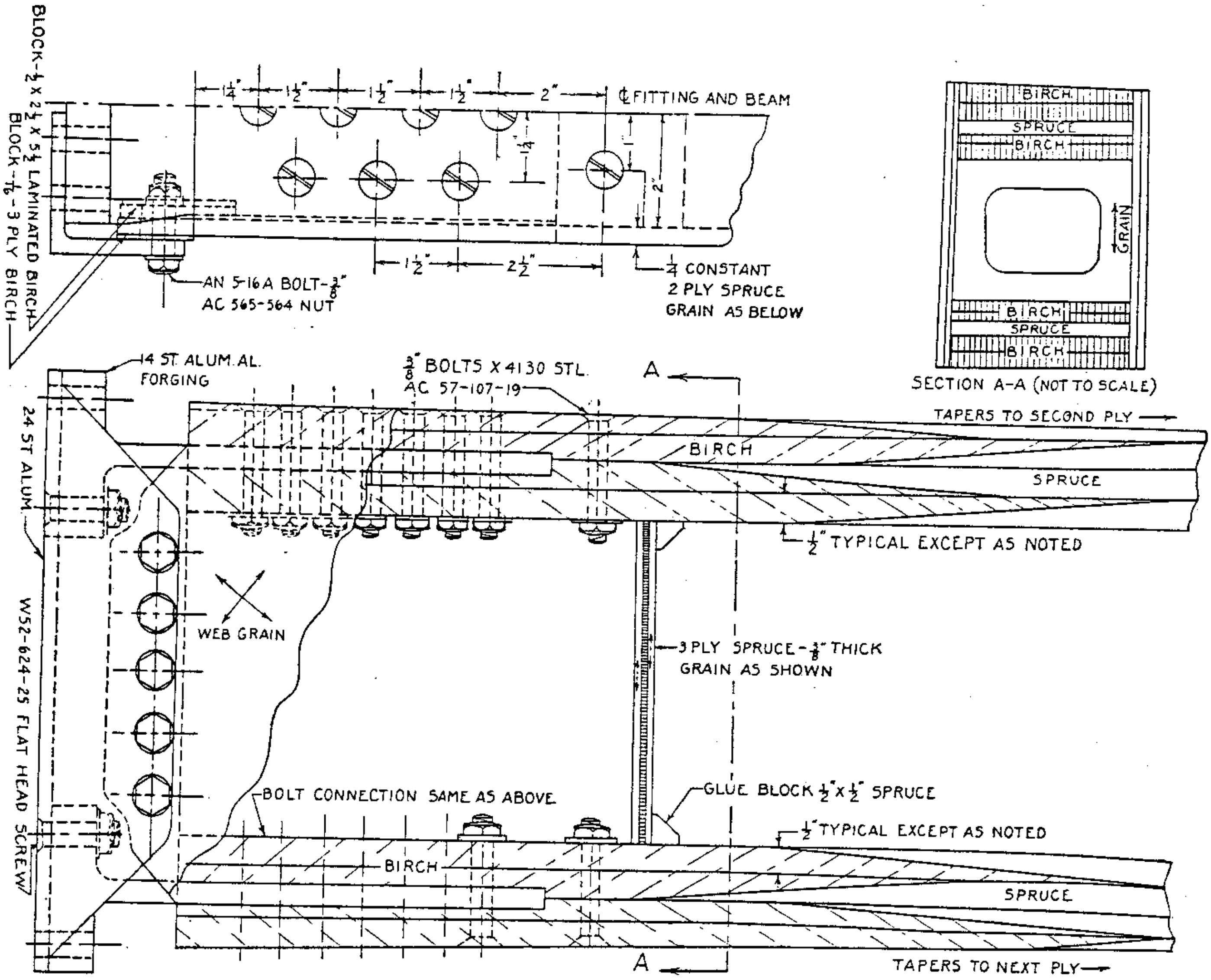


Figure 4–52. Wing beam attachment.

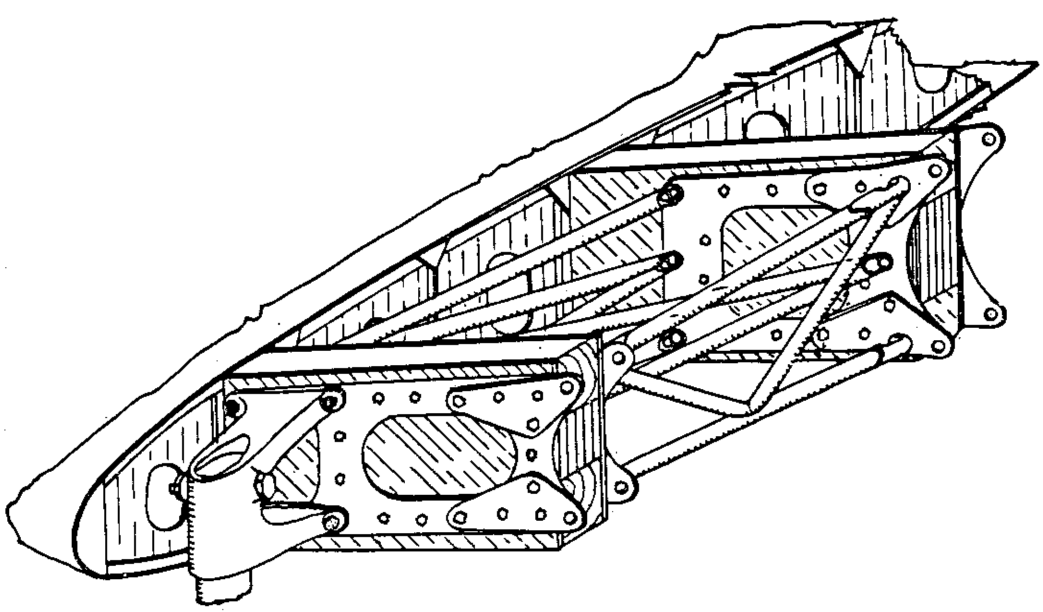


Figure 4–53. Details of landing gear attachment.

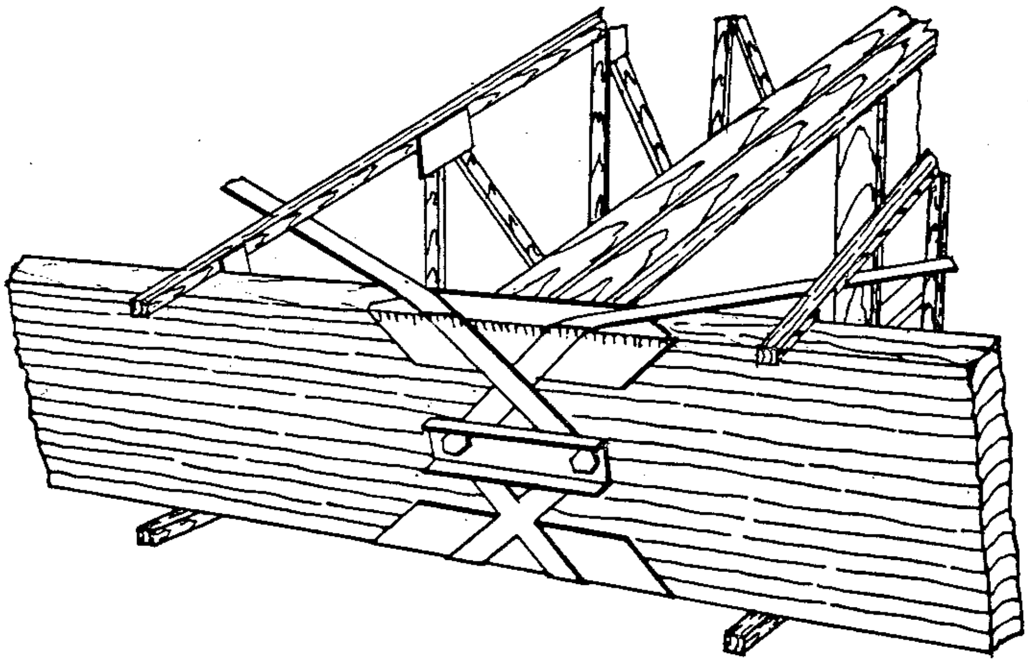


Figure 4–54. Method of double drag bracing.

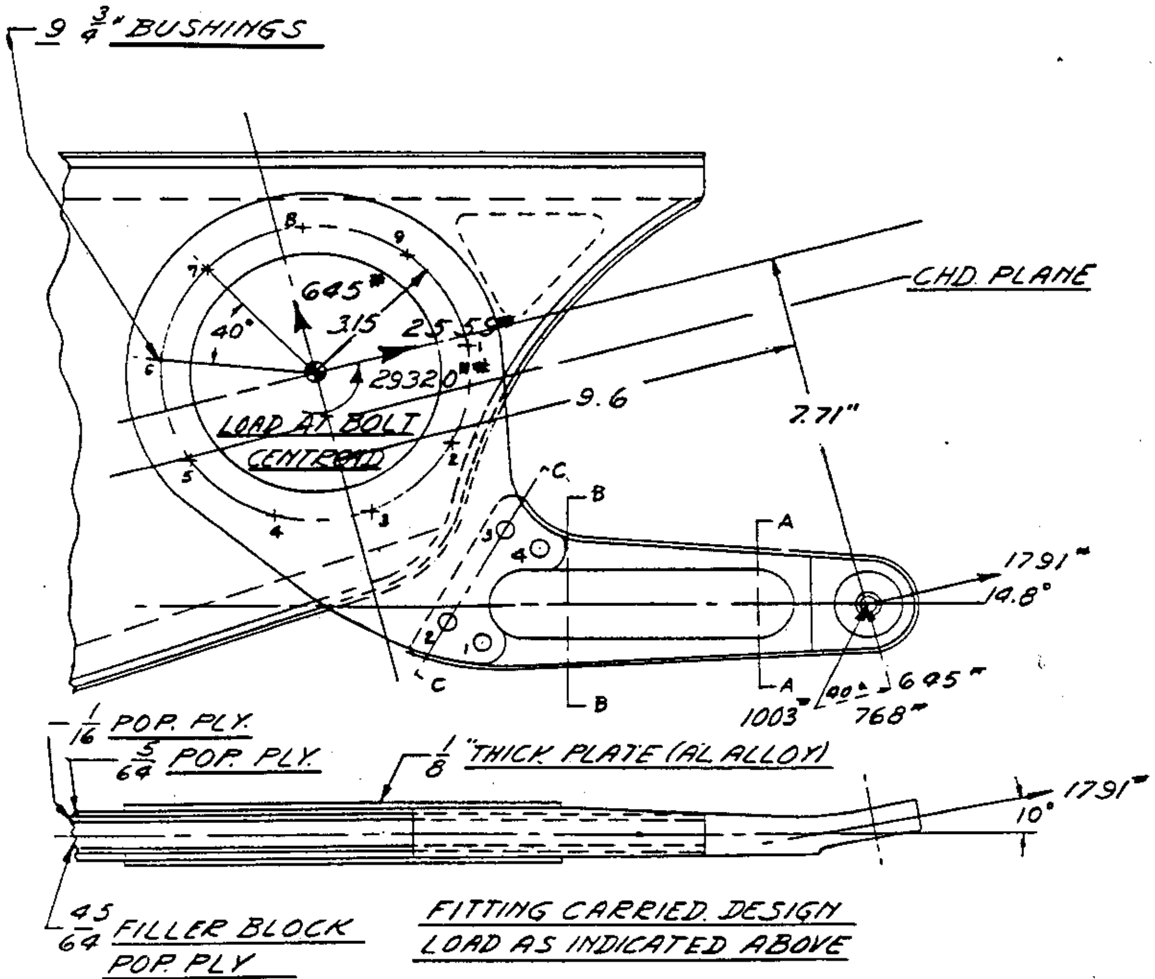


Figure 4–55. Attachment of flap hinge.

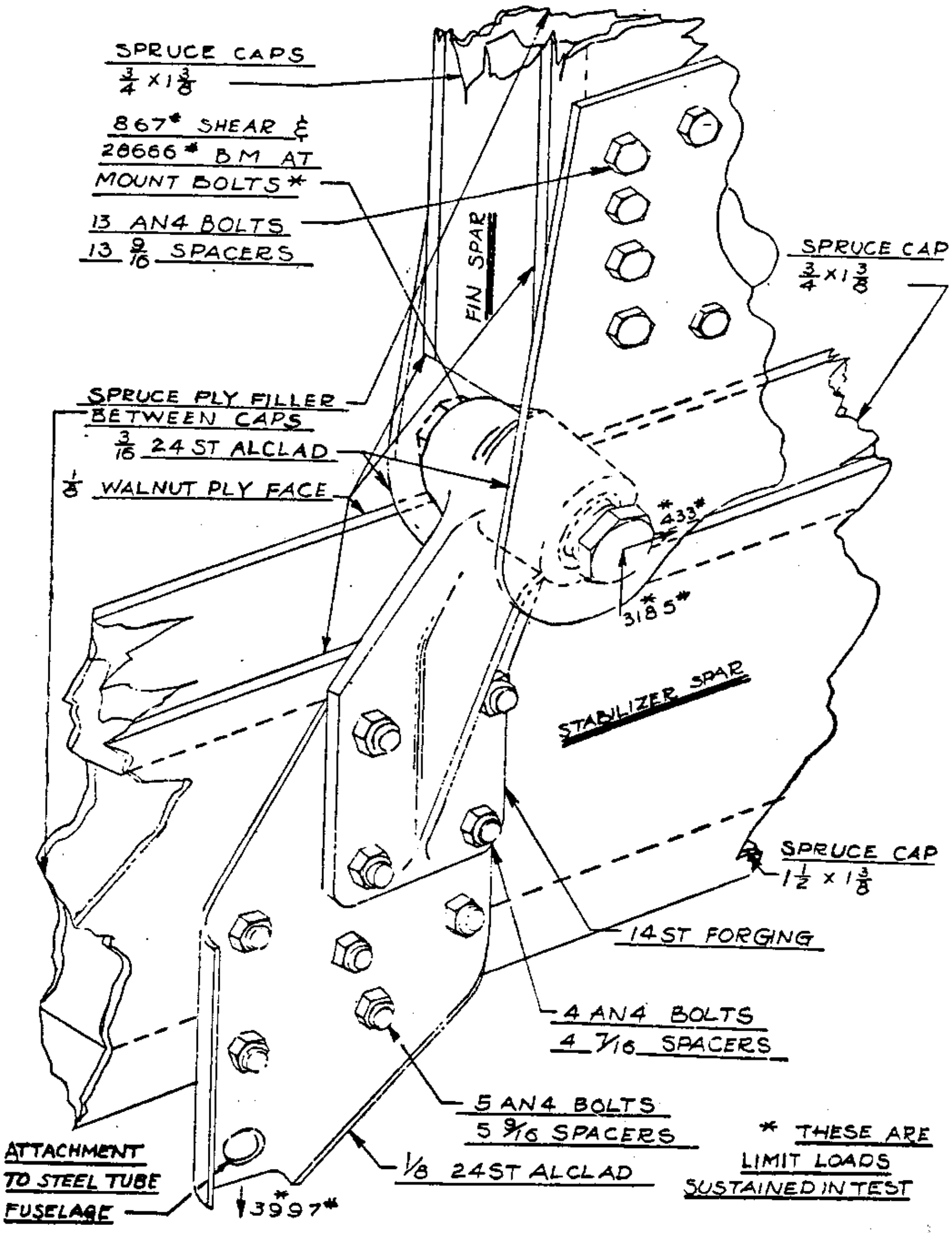


Figure 4–56. Attachment of empennage.

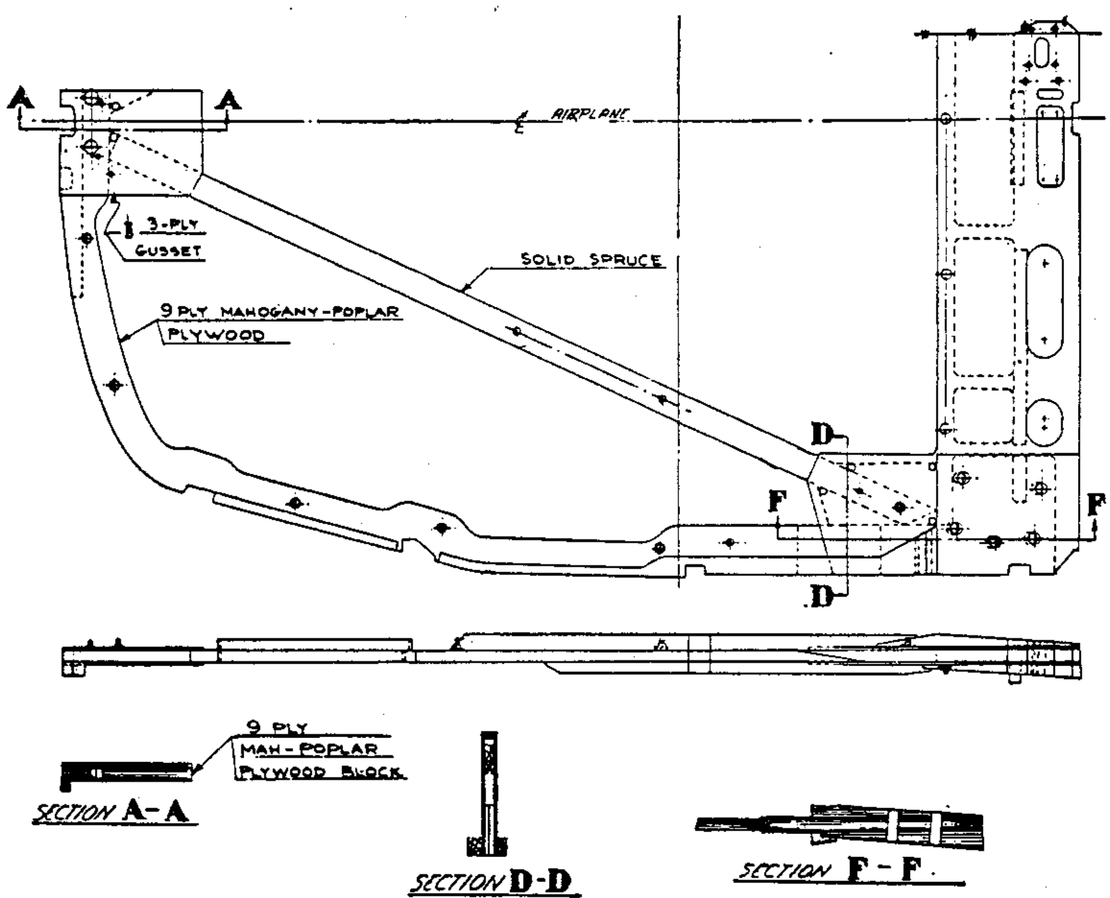


Figure 4–57. Reinforced fuselage frame.

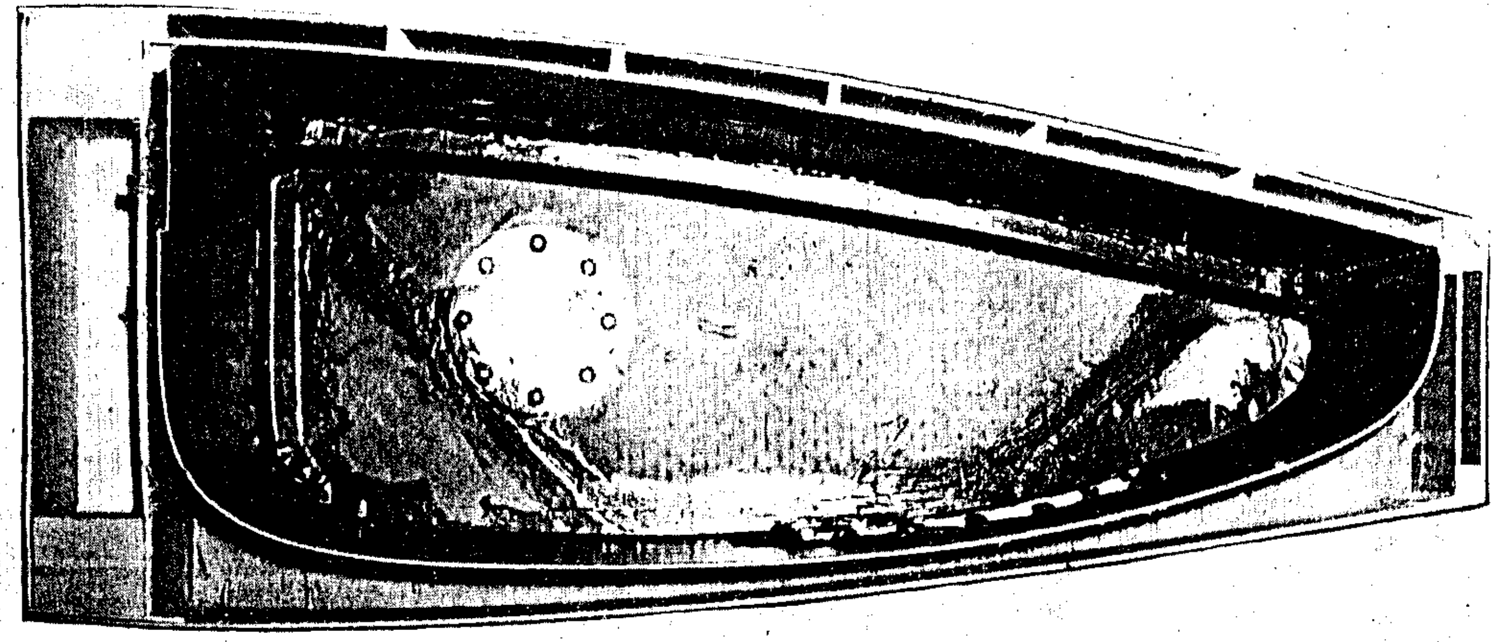


Figure 4-58. Cross section of wing showing integral fuel cell.

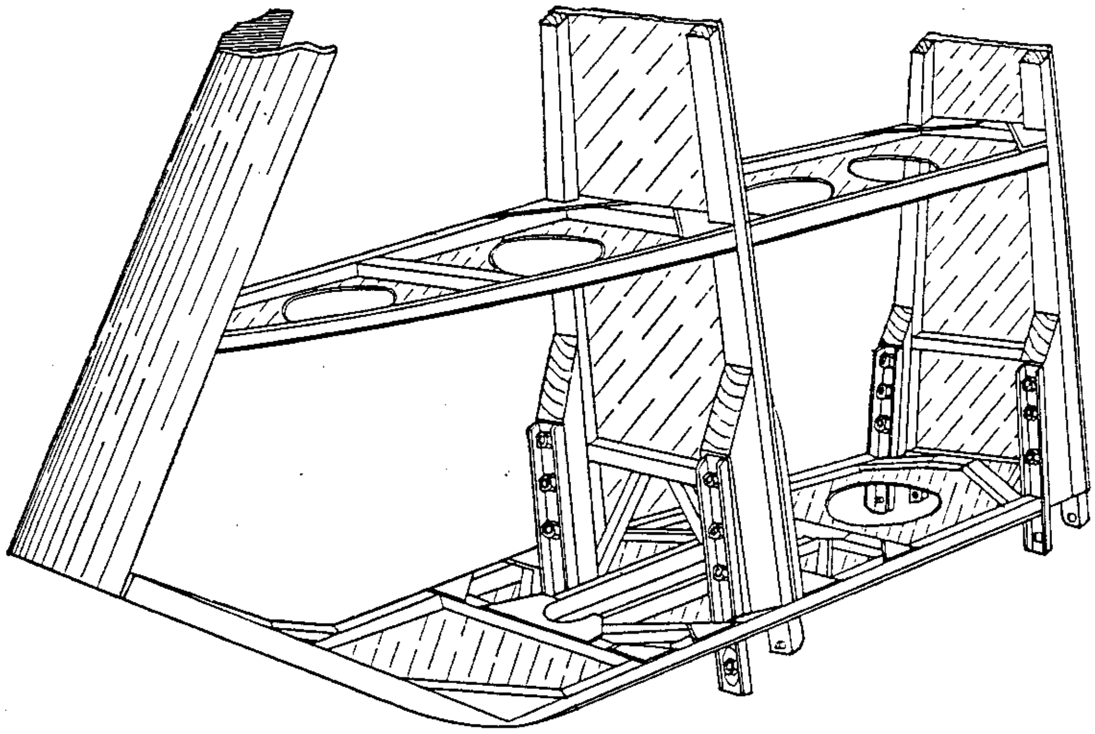


Figure 4–59. Example of wood control surface showing attachment details.

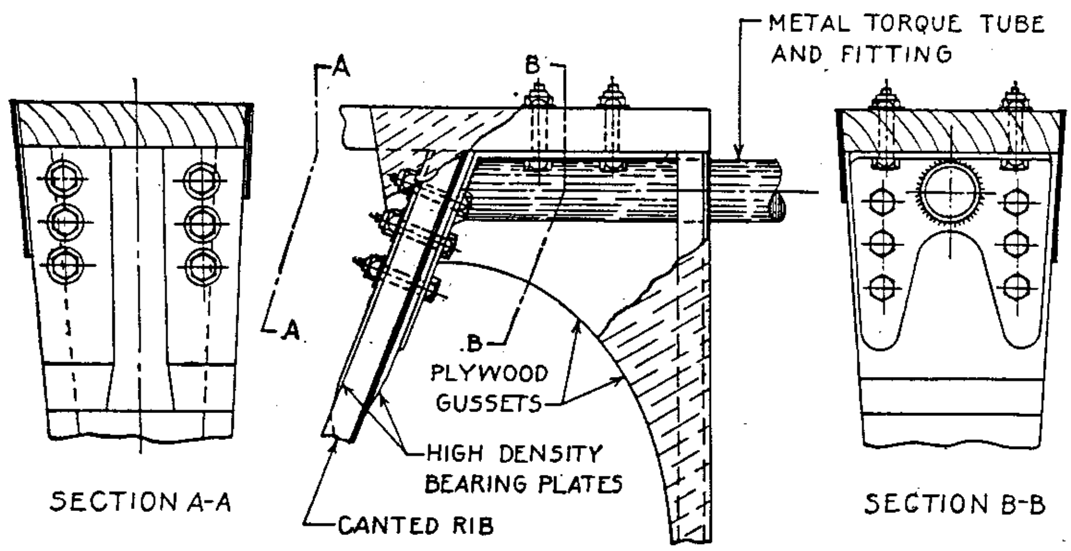


Figure 4–60. Example of elevator torque tube attachment to control surface.

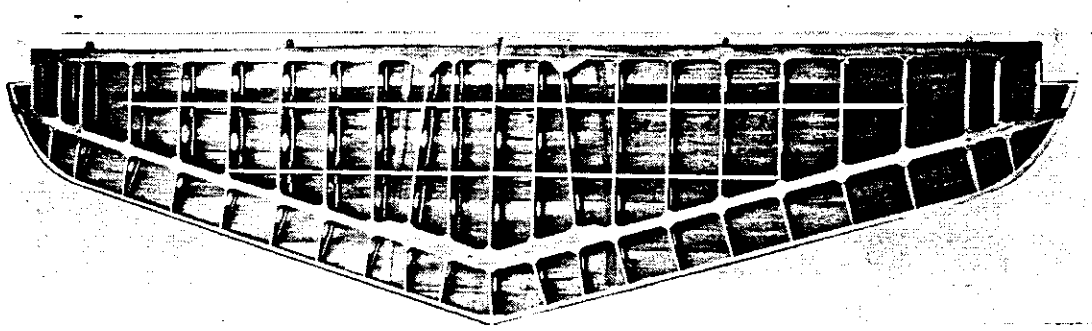


Figure 4–61. Wood stabilizer, one side of skin removed to show interior construction.

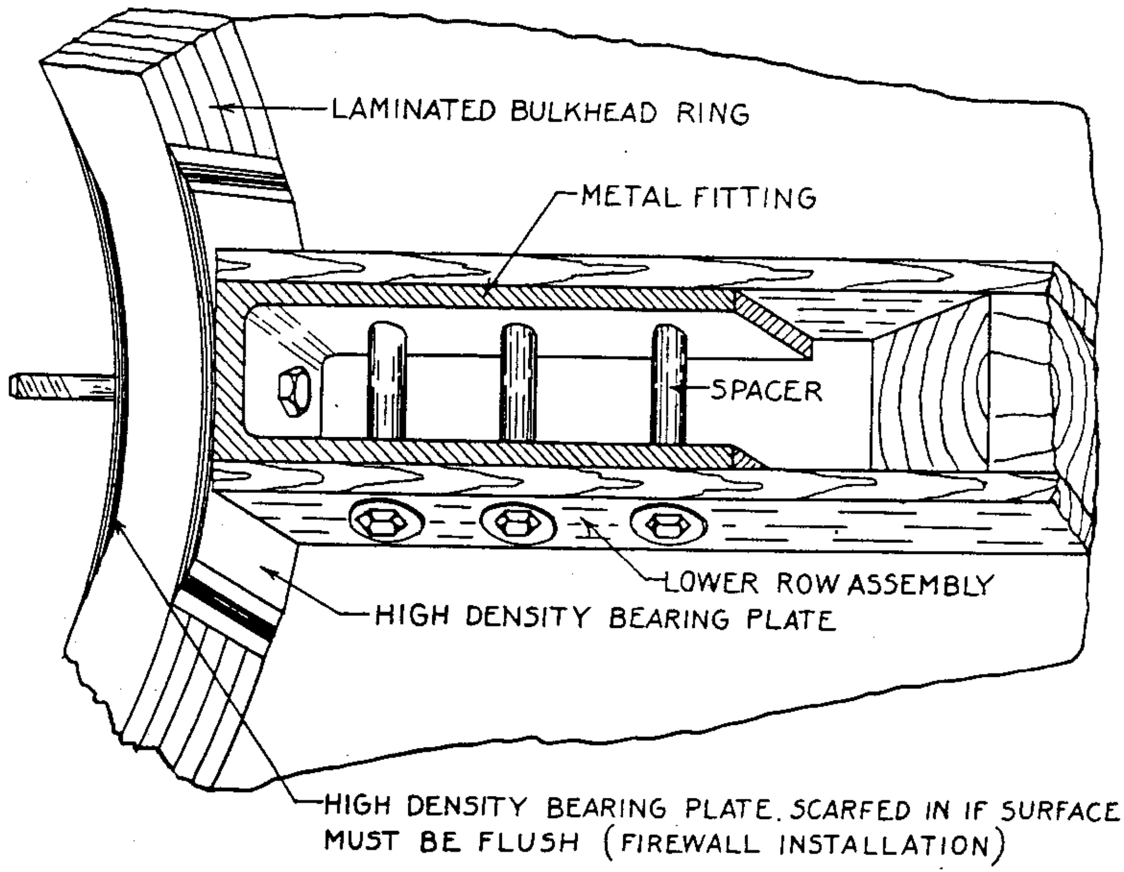
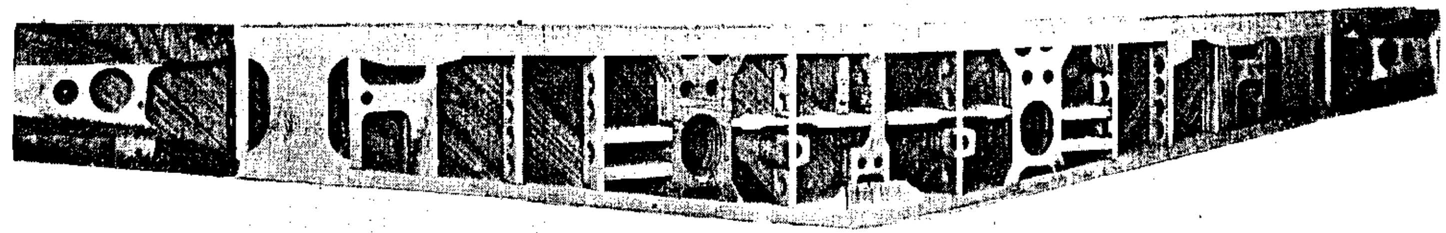
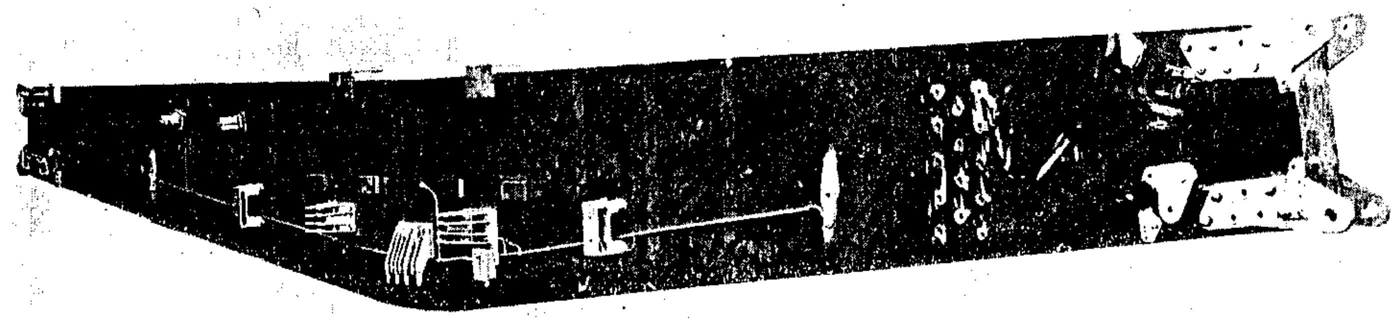


Figure 4–62. Typical fuselage joint or engine mount attachment.



CENTER SECTION BOX SPAR

FRONT WEB REMOVED TO SHOW BLOCKING



CENTER SECTION BOX SPAR SHOWING

FITTINGS

Figure 4–63. Typical main wing beam.

U. S. GOVERNMENT PRINTING OFFICE: 1951