CHAPTER IV.

PRINCIPLES OF CONSTRUCTION.

General.

91. The man who is concerned with the flying or maintenance of an aircraft is not usually in a position to work out the stresses in the various members, and he must therefore take these on trust, but it is very desirable that he should understand the more important underlying principles of construction of the aircraft with which he is in daily contact.

Basic considerations.

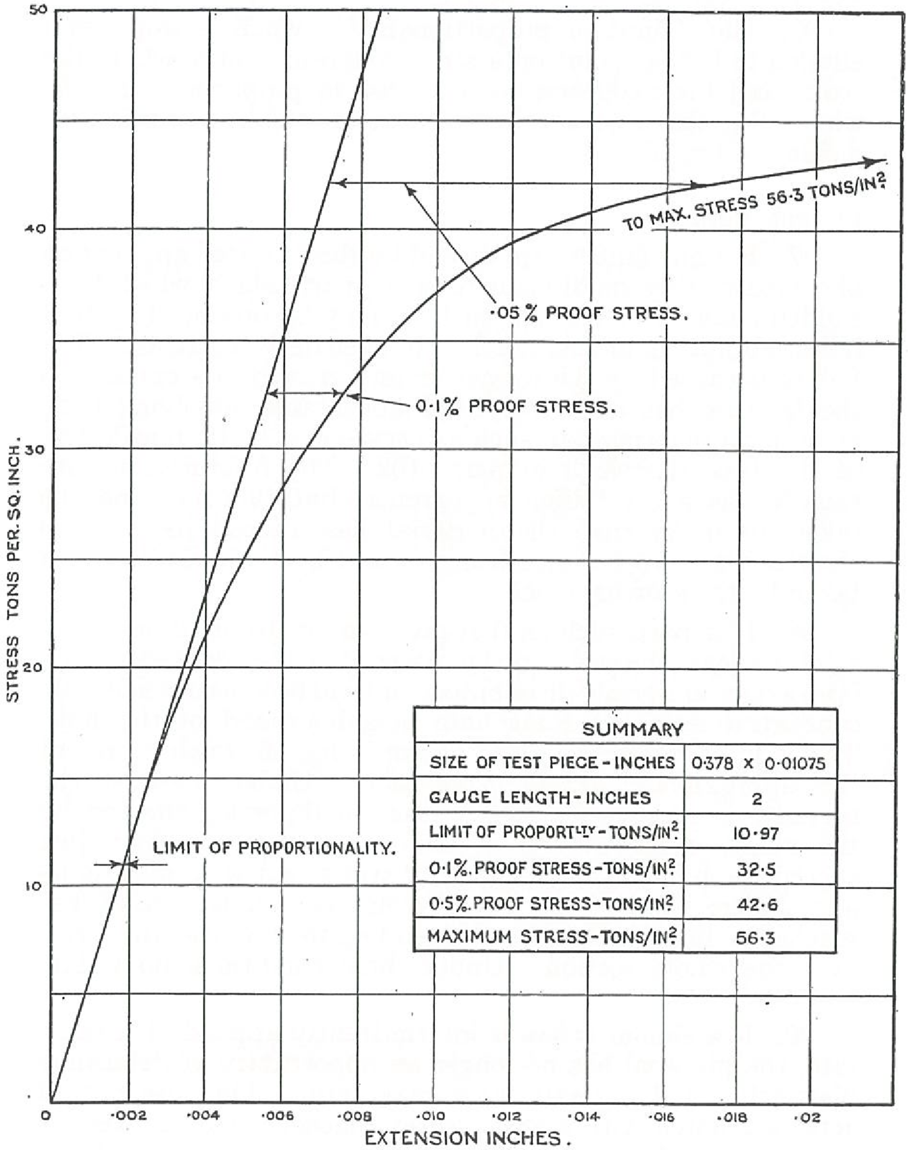
92. An aircraft is designed and built in accordance with a specification for the type. The specification contains all the requirements to be met by the designer, and on these he bases his calculations. The requirements laid down are, broadly, the duties and performance, load to be carried, and the horse-power available, and these are the main facts which determine the total weight and size of the aircraft, and which influence the designer in his selection of wing section, area, span, and general arrangement. There are many other considerations, both aerodynamic and structural, which have to be thoroughly thought out and understood before the design of an aircraft can take definite shape.

93. The basic principles of construction of an aircraft is that the maximum strength should be developed with the minimum weight of materials and resistance in flight. The attainment of this object is limited by such considerations as cost of materials and labour, and economy in manufacturing processes. Questions such as the durability and reliability of the materials and possible defects in workmanship affect the details of the design, and have also to be considered, but the principle of strength for weight must occupy the most prominent place.

94. The first problem of the designer is to ascertain, as accurately as possible, the maximum loads and stresses which may be encountered by the different parts of the aircraft. The second is to provide a structural arrangement capable of sustaining these loads and built in the lightest possible way. On the whole, aircraft structures follow accepted engineering principles, although the assumptions and the formulae used for heavy engineering are not always sufficiently accurate for aircraft strength calculations. In practice, the design and construction of aeroplanes present many problems, which are only solved by the closest investigation at the difficult points, making many tests, and paying great attention to accuracy of detail.

Strength of materials.

95. The strength of a material is generally quoted as that load in pounds or tons per square inch of cross-sectional area that it will withstand when placed in tension. The figure quoted is sometimes given as the "ultimate strength," or the load per square inch at which the material would break under test, and sometimes as the "proof stress" or "yield." The yield is not often quoted for aircraft use, because it is not



SUMMARY	
SIZE OF TEST PIECE - INCHES	0·378 x 0·01075
GAUGE LENGTH - INCHES	2
LIMIT OF PROPORT^LTY - TONS/IN².	10·97
0·1%. PROOF STRESS - TONS/IN²	32·5
0·5%. PROOF STRESS - TONS/IN²	42·6
MAXIMUM STRESS - TONS/IN²	56·3

FIG. 15.—Stress extension curve.

sufficiently definite; therefore the term proof stress is preferred. The proof stress may be taken to be that stress which causes a permanent extension in a test specimen, the extension being some agreed percentage of a gauged length, usually 0·1 per cent. to 0·5 per cent. This is made clearer by reference to fig. 15, which is a stress-extension curve plotted from the results of a tensile test of a steel specimen, with the stress as ordinates or vertical measurements, and the extension as abscissae or horizontal measurements.

96. The "limit of proportionality" which is sometimes alluded to is that point on a stress-extension curve where the stress and the extension cease to rise in proportion, that is, where the curve ceases to be straight. This point is also defined in fig. 15.

Fatigue failure.

97. Fatigue failures are caused by the repeated application of comparatively small loads to parts, particularly which have sudden changes of section, such as may be produced by bolt holes, abrupt shoulders, notches, or even deep scratches. The failure is caused by the formation and growth of a crack, and the fracture has a characteristic appearance involving little or no local deformation, such as occurs in a static tensile test of the type desciibed in para. 102. The fractured surface usually has a crystalline appearance, but this must not be taken to mean that the material has altered its internal structure, the appearance being caused by the irregular path taken by the growing crack.

98. If a part, such as that given in fig. 16, is subjected to a tensile load, the stress in the material at the reduced section is no longer uniformly distributed, and will become considerably concentrated in the immediate neighbourhood of the hole. This is indicated by the lines drawn in fig. 16 which represent the approximate stress distribution. Under load, if the material is reasonably ductile, the small portion affected by the stress concentration occasioned by the hole may just exceed the limit of proportionality stress and deform plastically, before the average stress over the whole section has reached a high value, thereby tending to equalise the stress over the whole section. Under these conditions no failure will result.

99. If a similar stress is intermittently applied at a rapid rate, the material has no longer an opportunity of deforming plastically, and a small crack may form. This crack itself acts as a notch with a still greater concentration of stress at its base, and the crack may therefore continue to spread and finally cause a fatigue failure.

Stress.

100. The direct stress in a member is the ratio $\frac{\text{load}}{\text{area}}$, or the load in pounds or tons, divided by the sectional area of the member in square inches. As an instance, if a specimen with a sectional area of 1 sq. in. is given a tensile load of 10 tons, then the stress is 10/1, or 10 tons per sq. in. If the specimen were only a half a square inch in area, then the stress would be $10/\frac{1}{2}$, or 20 tons per sq. in., and, conversely, if the area were 2 sq. in. the stress would be 5 tons per sq. in.

101. Up to the limit of proportionality, any member or specimen which is given a progressive tensile load will extend in direct proportion to the load applied, and, if the load is taken off, will return to practically its original length. This is the normal range of stress for any material under working conditions.

102. If the loading is taken beyond the limit of proportionality, the specimen will receive a permanent set, that is, if the load is released it will be found that the specimen is longer than it was originally. If the loading is continued until the specimen fails, considerable elongation will have taken place, the amount depending upon the ductility of the material. A good quality mild steel will extend at least 20 per cent. (about 1·6 in. when using a test specimen 8 in. long), and the area of any section of the specimens will have decreased proportionately, a very marked reduction in area being noticeable at the point of failure.

103. When a flat straight member, such as that shown on fig. 16, has a hole through the flat side, and is placed under a tensile load, the stress is greater across the section containing

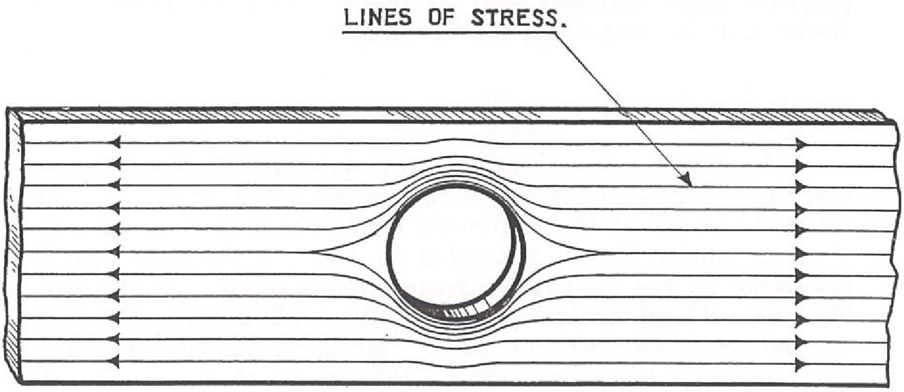


Fig. 16.—Lines of stress in a drilled bar under tension.

the hole than elsewhere, owing to the reduction of area at this point. That is, if the bar is 1 in. wide, and has in it a half inch diameter hole, then the stress would be doubled because the area is halved. The term " stress," as used above, is to be understood to imply " mean stress " or the average stress over any section taken through the test piece or member being considered. The stress in a member is not always evenly distributed over the section, but owing to irregularities in the material may be greater in one part than another. Referring again to fig. 16, the actual stress in this instance would be greater in the immediate vicinity of the hole than elsewhere, owing to the fact that the stress is diverted from the centre round the hole, as indicated by the lines of stress shown. The intensity of the mean stress of any part of a member under load can be fairly accurately estimated up to the point when the material commences to yield ; after this point the material reaches what is termed the plastic stage when the stress distribution is very problematical. Stress may be tensile, such as is found in bracing wires, or it may be compressive, as exemplified by struts, or shear, as encountered in pins. A tube or rod which is being twisted is also subject to shear.

104. The different ways in which the parts of an aircraft are stressed should at all times be studied, and it is a good exercise to examine the aircraft with this object in view. As an example, take the rather complicated stresses in the spars. The air pressure on the fabric produces a load on the bottom plane spars which is transferred by the struts and flying wires to the upper plane spars, and, by virtue of the lift wires, gives compressive stresses to these members. These stresses are again reduced or increased by the tension in the front, and compression on the back spars, caused by the drag of the wings. The ideal condition is that all parts should be proportionally stressed and employed as near their final strength as is compatible with safety.

Load factors.

105. The safety of an aircraft is dependent upon the maximum stress in the various members being lower than the ultimate stress of the materials from which they are made. The maximum stress allowable in the various members is regulated by the load factor requirements of the specification. The term " load factor " requires some explanation, as it must not be confused with the term " factor of safety " which is used in ordinary engineering. Presuming that a bridge is built with a factor of safety of six, then it would be strong enough to withstand six times the maximum load that could

come upon it. If an aircraft is to be given a load factor of 8, then the aircraft structure has to be made capable of sustaining eight times the unit load imposed by the particular condition of flight being considered, not eight times the maximum load that can come upon it. The value of the load factor to be given to the various parts of an aircraft is difficult to fix, as the conditions of loading in special circumstances are extremely difficult to anticipate, but usually the requirements are that the components should be capable of withstanding a certain load factor under certain conditions, the factors required varying with the type and duty of the aircraft.

106. Service aircraft fall into various categories, such as high-speed fighters, reconnaissance or bombing aircraft, etc. The load factors required under the various usual conditions are given in the specifications for the types, and detailed loading conditions are given in "The Handbook of Strength Calculations," Air Publication 970.

107. The usual conditions for which calculations are made to determine the strength of an aircraft are the wing structure with the centre of pressure forward (C.P.F.) and centre of pressure back (C.P.B.), nose diving, landing, the cut wire case and inverted flight. The load factors for the wing structure are in the neighbourhood of 8 and 6, with the centre of pressure forward and back respectively. The term "centre of pressure" is explained in a previous chapter (para. 17). In a terminal nose dive, an aircraft is presumed to be dived at its maximum speed with engine off, and for this case the load factor is about $1\frac{1}{2}$. This condition frequently stresses the aircraft to a greater extent than any other condition for which the structure is investigated.

108. All Service aircraft are liable to have a flying wire shot away, or otherwise cut or broken. In these circumstances the rest of the wires and structural members are made strong enough to enable the aircraft to remain in the air and reach its destination safely. For this condition, called the "cut wire case," load factors of half the normal flight load factors are required. For inverted flight, load factors are required equal to 2/3 the centre of pressure forward factor on the wings.

109. The alighting gear of an aircraft is arranged to withstand an impact load caused by a vertical velocity of from 10 to 12 ft. a second, and under this condition the load factors on the undercarriage are about $1\frac{1}{3}$, and about $1\frac{1}{2}$ on the remainder of the structure. When the aircraft is at rest on the ground, the corresponding factors are 4 and $4\frac{1}{2}$. In addition, presuming that an aircraft is landing with a

considerable amount of drift, there will be a side load on the undercarriage ; this condition is allowed for by making the undercarriage, with both or all the wheels touching, capable of withstanding a side load equal to the total weight of the aircraft.

Strength of members.

110. The most important of the aircraft parts are either beams, struts or ties, as instanced by the spars, interplane struts and the bracing wires respectively. The theoretical strength of a member in tension, such as a bracing wire, can be obtained fairly simply by multiplying its cross-sectional area by the ultimate strength of the material used, and if the normal direct stress, as described in para. 100, is known, then the load factors may be obtained by dividing the ultimate strength by the normal stress. On the other hand, the theories governing the strength of beams and struts are rather complicated, and cannot here be gone into in any detail. (Those interested are referred to Air Publication 970 and other publications where these matters are treated at length). It is, however, possible to describe briefly the considerations which govern their use.

Spars.

111. Aircraft spars are beams with, in many cases, a fairly heavy end load. The stresses in the spars, or in any member subjected to bending, are of a very complex nature. Firstly, there is the direct transverse shear due to the load applied, and secondly there are the stresses produced by pure bending. The transverse shear in a spar or beam is directly proportional to the load applied, but the stresses due to bending require some explanation. In fig. 17, A is an unloaded beam, and B the same beam with a heavy load at the centre,

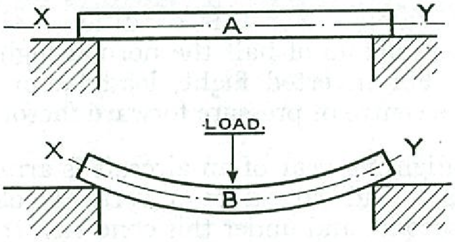


FIG. 17.—Loaded and unloaded beam.

the deflection being greatly exaggerated. Clearly, the upper surface of the beam B is shorter than the upper surface of beam A, and the lower surface of beam B correspondingly longer than that of beam A. It therefore follows that at some plane in beam B approximating to X–Y the length is the same as that of the length of the same plane of the unloaded beam, or alternatively the same length as it was before bending. This plane is called the neutral axis. As indicated in the stress distribution diagram in fig. 18, all the fibres composing the beam above the neutral axis are subjected to compression and longitudinal shear stresses, and all the fibres below to tension and longitudinal shear. The longitudinal shear stresses are present as the result of the upper and lower fibres having expanded or contracted to a greater extent than the centre fibres, and although the section of the beam is homogeneous, there is some tendency towards relative movement between each minute layer of material. The tendency of each layer of the material to slide over its neighbour is resisted by the cohesion of the material itself. Bending a pack of cards illustrates what would happen if each layer were free to slide. Theoretically no direct stress due to bending is present at the neutral axis, but there are usually shear stresses, as explained

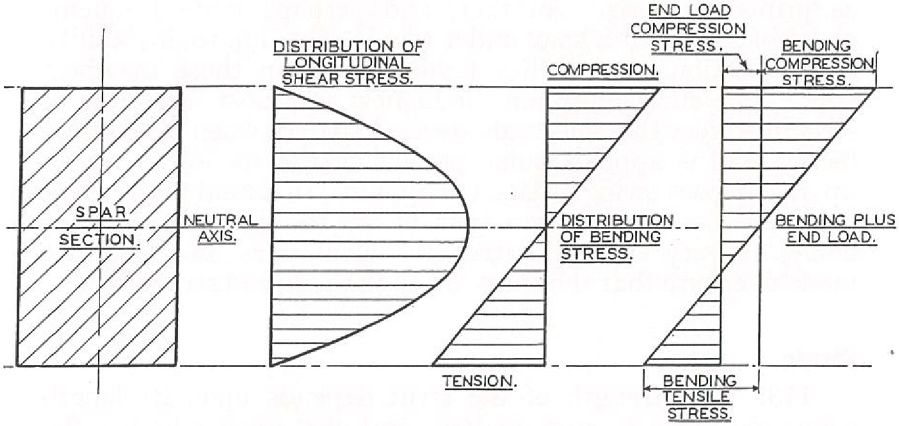


FIG. 18.—Diagram of stress distribution in a spar.

above, and these are at a maximum at this point. The stress distribution in a spar under bending, and bending plus end loads is indicated in the diagram given in fig. 18. A beam may obviously be of any cross-section, but the most efficient type is that in which most of the material is disposed as near as possible to the outer fibres, where the most work is done in resisting bending. The left-hand illustration of fig. 19 shows a section of the type of built up beam used in heavy engineering, the remainder show typical sections of beams or spars as used in aircraft.

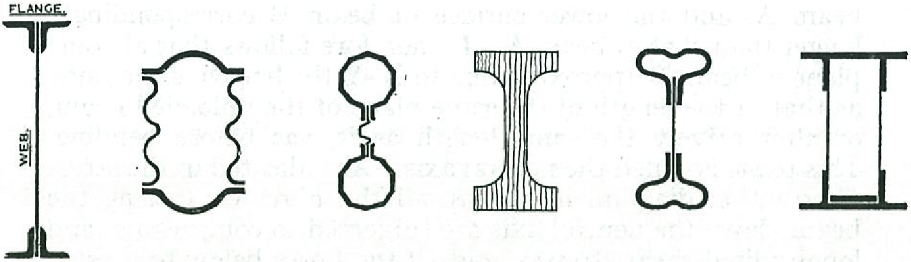


FIG. 19.—Typical beam sections.

112. In many cases an aircraft spar is subjected to end loads as well as bending loads, and is therefore acting partly as a strut and partly as a beam. When a solid member of regular cross-section, such as a wooden spar, is tested to destruction under these conditions, the failure can be usually attributed to primary causes, that is, owing to the externally applied load, some part of the outer fibres of the spar material reaches its limit of strength and failure occurs. On the other hand, when a hollow metal spar of built-up section is similarly tested, the failure may happen as the result of secondary as well as primary causes. Another, and perhaps more frequent, cause of failure of a spar under test is that due to instability of the section. Instability failures occur in those members where, in a effort to obtain the highest and most rigid form of structure, very thin materials are used, so that, when a relatively heavy load is applied, some portion of the section crumples up or collapses owing to lack of support. In actual flight it is a very rare occurrence for a spar to fail through the air loads alone, as very thorough strength calculations and tests are made to ensure that the spars are to the required strength.

Struts.

113. The strength of the strut depends upon its length compared with its cross-section, and also upon whether the ends are fixed or pinned. A fixed-ended strut is theoretically four times stronger than a corresponding pin-jointed strut, but full advantage cannot be taken of this for aircraft work, because parts to which the struts are connected are themselves comparatively flexible. In these conditions the benefit of using fixed-ended struts is doubtful, as it is possible for a strut of this nature to impart loads to the adjoining structure which would more or less nullify the benefit of the fixation of the strut ends. Take as an instance a fuselage strut which is connected to the longerons by rigid joints. Imagine that the strut has deflected under load, say from the undercarriage. The joint itself will not yield appreciably, so the longeron bends as well as the strut. If at the same time the longeron is

also highly stressed owing to, say, tail skid loads, then it is possible that the longeron would fail as the result of the combined loads, although quite strong enough in itself to withstand its own primary loads.

114. A strut, unless it is very short, cannot be subjected to the maximum compressive value of the material from which it is made, because when loaded at the ends a strut of parallel section will fail at the centre by bending as indicated on fig. 20. When a strut fails under end load, the stresses in the

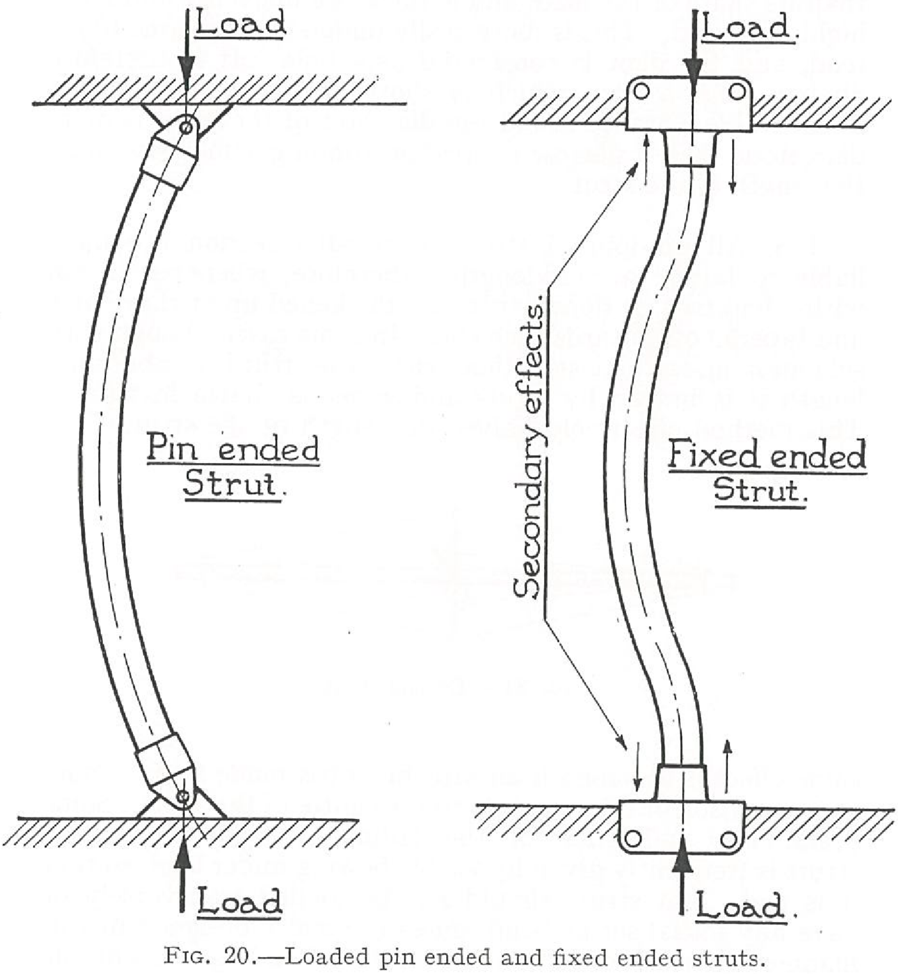


FIG. 20.—Loaded pin ended and fixed ended struts.

material are partly attributable to the direct end load applied and partly to the bending action. The stresses due to the bending action are practically identical to those in a beam under bending loads, which have been already described.

115. When a loaded strut of any form is damaged by having a dent made near the centre of its length, the stresses in the material may be very much increased. If we take a tubular steel strut and apply a safe load to its ends, the stresses are fairly even over any section taken through the strut, because the material is of even thickness, and each small section or part of the section is deriving an equal amount of support from the remainder. When a dent is made, it is clear that this support is taken away or much changed over the portion distorted by the dent. The result is that the undistorted portion is taking more than its share of the load, and is therefore unevenly and more highly stressed. This is more easily understood if para. 103 is read, and the dent is considered as a hole. It is therefore obvious that a dent which is short, but which extends a considerable distance round the diameter of the tube, is more dangerous than a narrow depression running some way along the length of the strut.

116. All pin-jointed struts of parallel section are most liable to failure at mid-length; therefore, whenever it can with advantage be done, struts are thickened up at the centre and tapered off towards each end. In some cases, though very seldom in aircraft construction, where the strut is of abnormal length it is braced by struts and wires as shown in fig. 21. This method effectively halves the length of the strut. The

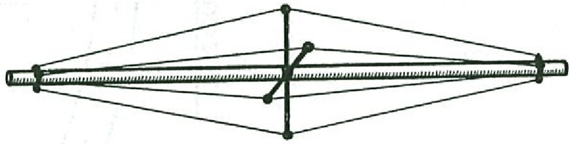


Fig. 21.—Braced strut.

same effect is obtained if an attachment is made from a more secure portion of the structure to the centre of the strut. Some preliminary indication of the failing strength of aircraft struts is frequently given by visible bowing under load, so that it is clear that struts should not be loaded transversely or have any initial set or bend unless originally designed in this manner, since this condition increases the bending moment due to the end load.

In the larger size of struts, a considerable economy of weight of material can usually be made by employing a built-up type. This type is especially applicable to heavily loaded interplane struts of streamline section.

Fittings.

117. Most well designed fittings have the lines of action of all the loads coming upon them meeting at a point. In other words, if an imaginary line is drawn through the centre of the struts, wiring lugs and other load-carrying parts attached to the fitting and produced through to the centre line of the main member, all the centre lines will meet at a point on the centre line of the main member, as shown in fig. 22A any departure from this imposes an off-set load, and therefore a bending moment, on the main member.

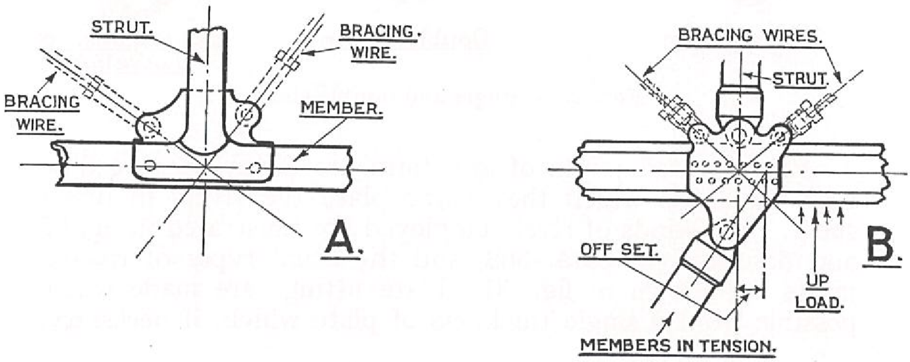


FIG. 22.—Lines of action in fittings.

118. It is not always possible to avoid off-setting a wiring lug or similar attachment, especially in the case of hollow metal spars. In these cases the members affected have to be made sufficiently strong to withstand the bending loads caused by the off-setting. There are also instances where a deliberate off-setting has been arranged for one of the attachments of a fitting, in the manner shown on fig. 22B, so that the bending action caused thereby tends to cancel out a bending load in the opposite direction which is caused by some other action.

119. All fittings are to some extent subjected to vibrational loads, but the durability of those fittings which take the direct vibration from the engine, such as those on the engine mounting, cannot be judged by the ordinary strength calculations. Unless the areas in contact are of ample size, the hammering action produced by the partial or complete reversal of load caused by the vibration will, in time, cause elongation of holes, or fatigue cracks (as explained in paras. 97 to 99), and finally failure.

Bolts used on the principal fittings are always arranged to be either in tension or shear, or a combination of both, and pins in shear only. These parts are never subjected to bending if it is at all possible to avoid it, as the stresses in the part would thereby be very much increased. This points to the necessity of keeping all bolts or pins drawn up tight. Whenever it can be arranged, bolts and pins are placed in double shear as indicated on fig. 23.

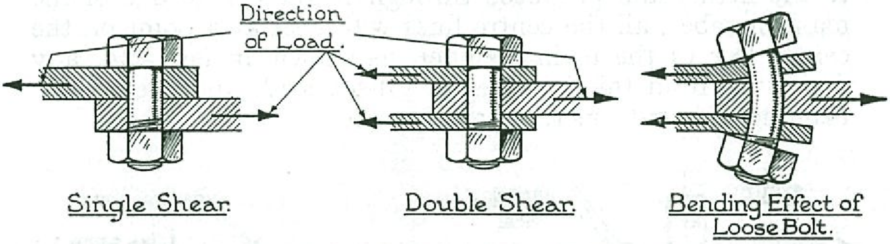


Fig. 23.—Single and double shear.

120. Riveted joints of any form are always arranged so that the loads which they carry place the rivets in direct shear. The kinds of rivets employed are illustrated in fig. 95 and described in para. 388, and the usual types of riveted joints are shown in fig. 96. Plate fittings are made where possible from a single thickness of plate which, if necessary,

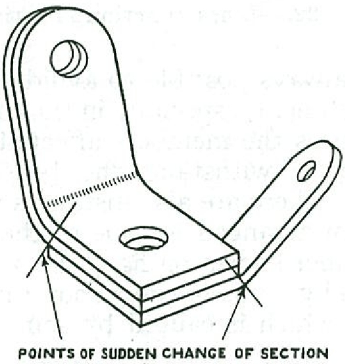


Fig. 24.—Laminated plate fitting.

may be machined down in places to any required thickness. Laminated plate fittings of the type shown on fig. 24 are seldom used, as they are apt to give trouble at the junction of the plates, especially if the parts are highly stressed and there is a large difference in the thickness of material at the change of section. Wiring lugs which are at an angle from the main fittings are usually arranged with the bend close up to the holding-down bolt, as indicated in fig. 25A, but when this is not possible a block or pad is sometimes arranged under the head of the nearest bolt, as shown on fig. 25B, to prevent any distortion. Wiring and other lugs are always bent to the same angle as the wires or struts to which they are attached.

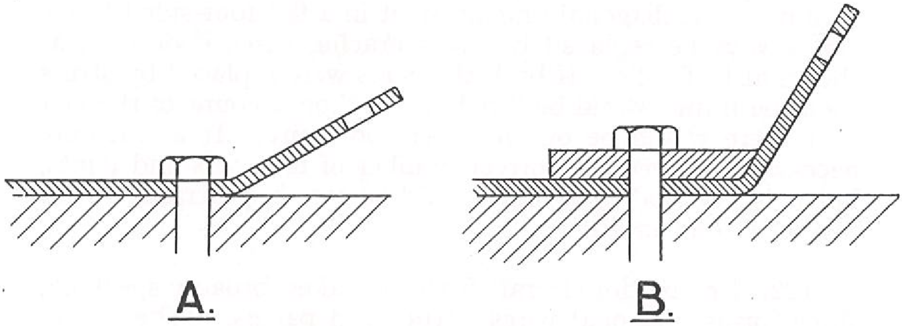


FIG. 25.—Wiring lug attachment.

Braced Structures.

121. Bracing is used in any structure, and particularly so for aircraft, to obtain rigidity with the least possible expenditure of material. Most aircraft parts would need to be a great deal heavier if bracing in one form or another was not employed. In their usual form, aircraft structures consist of open frames built up from struts and ties, pin-jointed together, and designed to be loaded mainly at the joints. A perfect frame has neither too many nor too few members. Fig. 26A represents a perfect frame in its simplest form. If a structure of this type is supported at A and C, and a load applied at B in the direction indicated by the arrow, then this load will be resisted by pure compression and tension in the members A, B, and B, C. Fig. 26B shows a frame which is imperfect, because it would deform if a load were applied

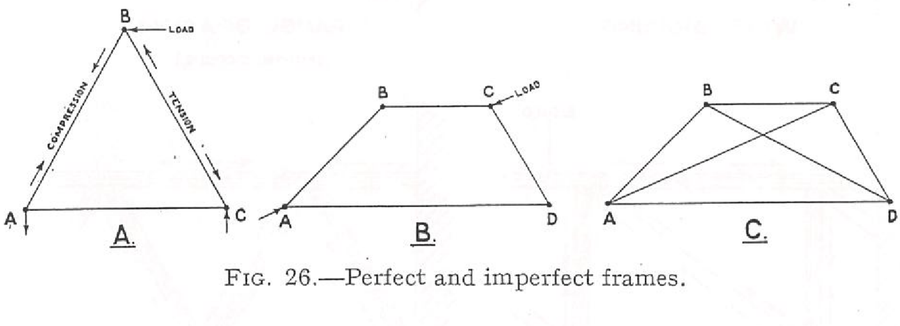


FIG. 26.—Perfect and imperfect frames.

in the direction A to C, although it would be in equilibrium under certain conditions of loading. The number of members required for a perfect frame can be determined by the formula $2n—3$ for flat frames, and $3n—6$ for " solid " or cubical frames, where n equals the number of nodes or joints. Taking the frame as given at B, fig. 26, the number of joints is four, therefore (2×4) — 3 = 5, so that in this instance it would be necessary to have a diagonal strut in addition to the side members. A diagonal bracing strut in a flat four-sided frame can always be replaced by cross bracing wires if desired, as shown at C, fig. 26. If both the wires were replaced by struts then the frame would be "redundant" on account of the fact that there would be one member too many. It is not only necessary to have the correct number of members and joints, but it is essential that they should be correctly arranged with respect to one another.

122. Bracing for aircraft fuselages takes, broadly speaking, three forms, diagonal wires, struts and panels. When wires are used, say, to brace a fuselage bay, these are taken diagonally across from corner to corner, and when the fuselage is under load, one or the other of the wires is in tension, depending upon the direction in which the load is applied, as indicated in fig. 27A. When strut bracing is employed, as at fig. 27C and D, single diagonal struts replace

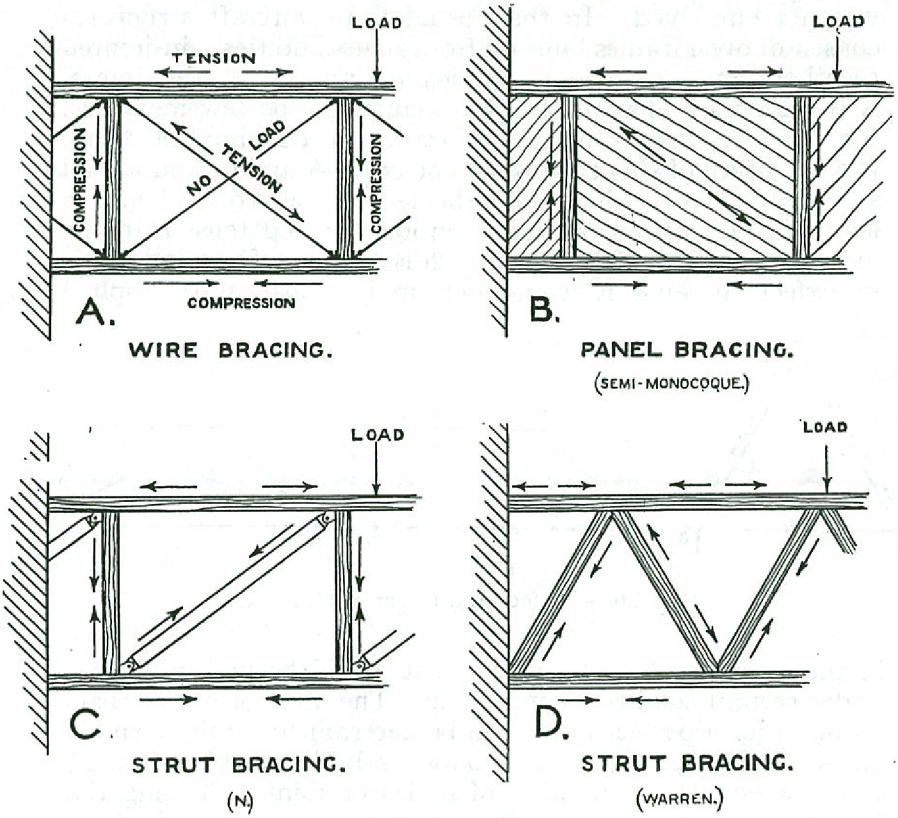


FIG. 27.—Braced frames.

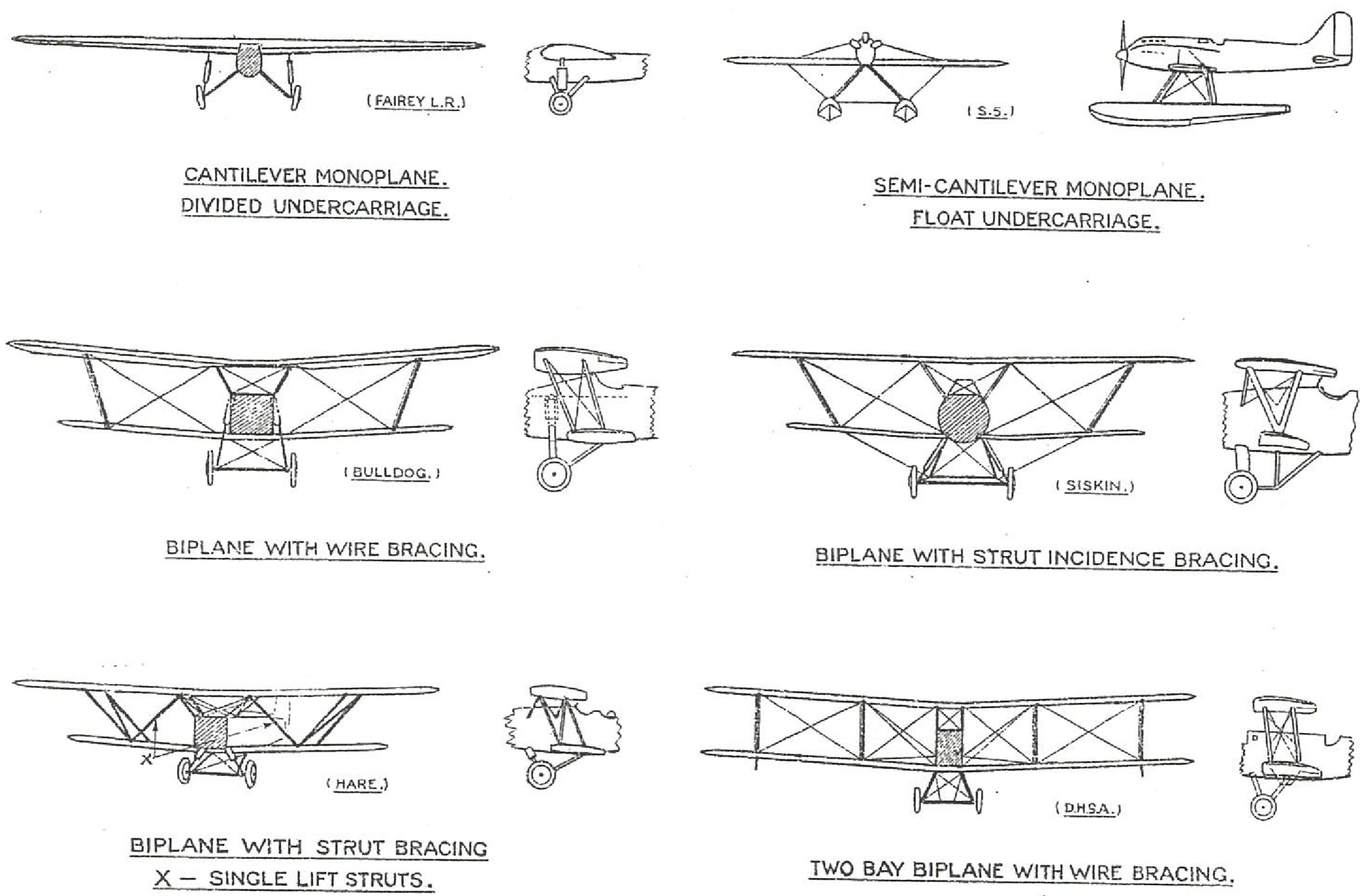


FIG. 28. TYPICAL FORMS OF BRACING.

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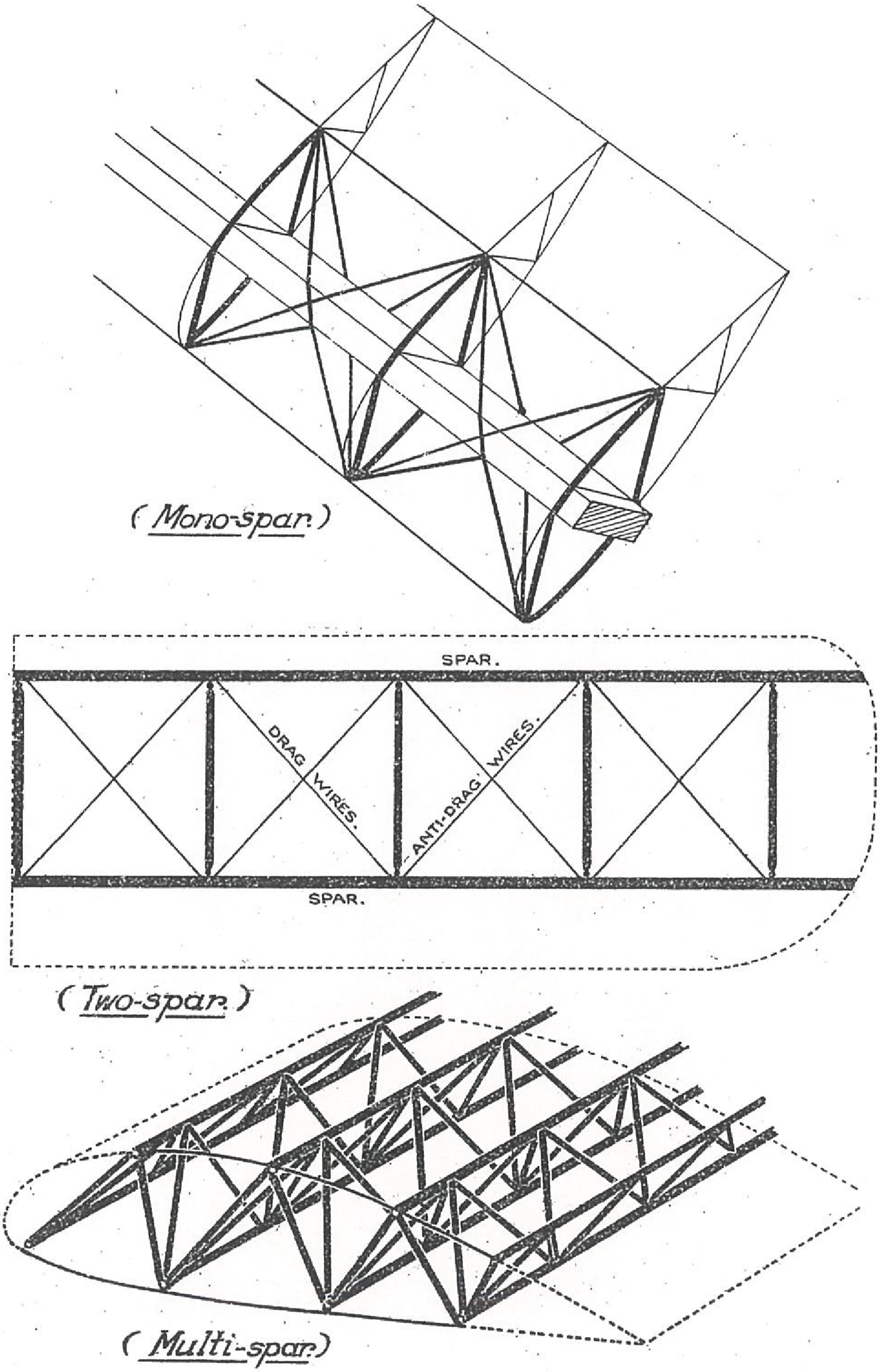

FIG. 29. TYPES OF INTERNAL WING BRACING.

the wires, and when under load, are either in tension or compression, also depending upon how the load is applied. Three-ply (or possibly sheet metal) panels sometimes replace wires or strut bracing in aircraft structures as illustrated at B, fig. 27. The loads in panel bracing are mainly taken in tension, but the stresses are of a complex nature owing to the comression buckling and shear stresses. This type of panel is very effective in maintaining the shape of the structure.

123. Details and materials may vary considerably, but the plan arrangement of any biplane wing generally follows well established economical lines, such as that shown in the centre illustration of fig. 29. It consists of an open frame composed of two spars, drag struts, drag and anti-drag wires. To this frame are attached ribs which are shaped to form the desired wing section and made strong enough to take the air loads imparted by the fabric, with which the whole is covered. Monoplane wings, on the other hand, may vary considerably in general design, especially in the case of the cantilever monoplane, but for the smaller aircraft of this type the same two-spar construction is generally used. On the larger monoplanes there may be one, two or a number of spars and the covering may be of fabric, wood or one of the light alloys. The wings of a cantilever monoplane are thicker than the wings of a biplane of similar weight, chiefly owing to the practical necessity of having to provide a greater depth of section for the internal structure. The greater depth of section is also required to allow adequate precautions to be taken to resist the twisting imposed by the ailerons, which on the larger monoplanes is a serious consideration. When a stronger substance than fabric is used for the plane covering, it is generally used as a strength member, and definitely takes a proportion of the air loads. This does not refer to the solid coverings used as walkways or engine platforms. Fig. 29 shows types of internal bracing used on mono-spar, two-spar and multi-spar wings. In the multi-spar diagram shown, the covering would have to take a large percentage of the loads.

124. When cellule or six-sided cubical bracing is considered, such as is usual in the wings of biplanes, it is not essential for complete rigidity that all six sides should be braced. Provided that the base is rigidly supported, it is only necessary that four other sides should be braced, as shown on fig. 30. As indicated, any side loads imparted to the members of the open end are resisted by the struts and diagonal bracing of the sides. If the base is not rigidly supported, then the structure will be flexible, owing to distortion of the members constituting the base, in a direction at right angles to the plane of that base.

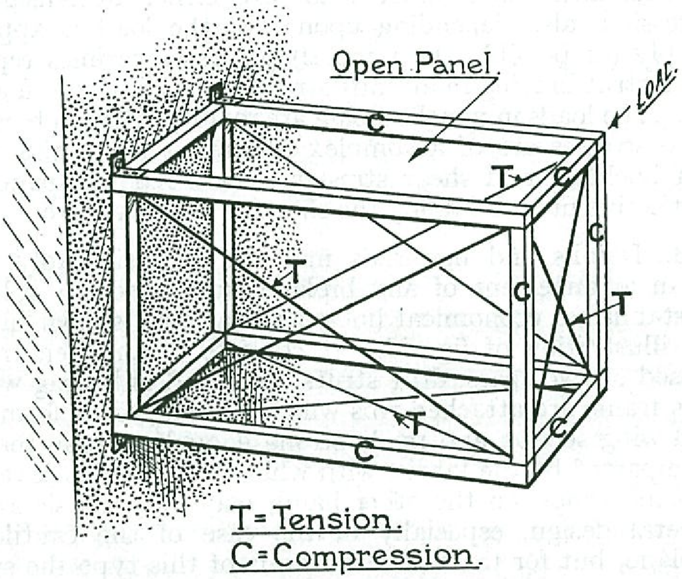

Fig. 30.—Cellule bracing.

125. In order to differentiate between the wires used for bracing aeroplanes, they are generally indicated in terms of the purpose for which they are used. In this way, as indicated in fig. 4, the "Lift" wires are so called because the principal function of these wires is to transfer the lift of the wings to the body or other part of the aeroplane, the "anti-lift" or "landing" wires resist forces in the opposite direction to the "lift" wires, and are usually under maximum load only during landing. "Drag" wires are those which transfer the drag of the planes to the body or other part of the structure, "anti-drag" wires resist forces in the opposite direction to the drag wires, and the "incidence" wires brace a main plane cellule in the plane of a pair of interplane struts. Some of the various forms of wing bracing in common use are shown in fig. 28; all the types illustrated being tractor aeroplanes.

126. An undercarriage bracing is usually similar in principle to one of the types of bracing described in paras. 246 to 248, and generally takes the form of cross bracing wires or cables, but the kind of bracing depends largely upon the type of undercarriage employed, many modern undercarriages being strut-braced. Undercarriage bracings and structure are sometimes slightly complicated by having universal joints at the ends of the articulated struts or radius rods. These joints are necessary in order to permit the movements required for the shock absorbers. The types of undercarriage can be

classified as through axle, divided axle, twin, and float, all of which are shown diagrammatically in fig. 28.

Shock absorbers.

127. Before an aeroplane can come to rest, during the operation of landing, the forward velocity must be absorbed in the work done in travelling over the ground, but the downward velocity must be absorbed in the structure itself; therefore, some form of shock absorber is necessary.

128. In all properly designed shock absorbers, the principle is very similar, and that is to convert the kind of energy possessed by the aeroplane when landing, called kinetic, into some other form which can be easily dissipated. Shock absorbers take the form of steel springs, rubber in tension or compression, or some type of oil or air dashpot. Steel springs, or rubber in tension, operate by storing up the energy imparted to them during landing, and retaining the energy until released. Oleo shock absorbers convert the kinetic energy into heat by the work done in the dashpot, and give off the heat by conduction and radiation. The functioning of the compression rubber shock absorbers is a combination of the principles of the spring and the oleo, most of the energy being stored, and the remainder dissipated as heat. Figs. 47, 67, 68, 69 and 70, show the various types of shock absorbers used.

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