AIRCRAFT LOADS AND LOAD TESTING

When trying to get new designs cleared by the LAA, the structural strength part of the submission is usually the source of most difficulty on the part of the applicant.

This leaflet describes what load cases are normally tested and issues involved in working out the magnitude and distribution of the loads. This is intended to show how to avoid some of the pitfalls encountered in trying to get new designs cleared over the years, and we hope that by sharing this material freely we will stimulate more aircraft designs in the UK and help those seeking to bring in existing designs from abroad.

PART 1 AIRCRAFT LOADS

Wing Loads

The loads which are considered acting on a conventional wing are lift, drag, pitching moment and inertia. Hiscocks book 'Design of Light Aircraft' (reference 1) provides a full description of how to calculate the loads acting on a wing, including worked examples. The methods are too lengthy to go into in depth here, so I shall concentrate on the simplifying assumptions that can be made and some of the more common mistakes that crop up.

The crudest acceptable estimate for the magnitude of the lift load on the wings in flight is obtained by multiplying the max gross weight of the aircraft by the load factor for the flight condition in question, multiplied by 1.05. The extra 5% wing lift is an arbitrary margin intended to compensate for the tailplane load which on a conventional aeroplane acts in the opposite direction to the wing lift, eg the tailplane presses down on the rear of the fuselage in order to pitch the aeroplane nose-up into a positive g manoeuvre. The wings have to provide not only sufficient lift to accelerate the mass of the aeroplane upward, but also to overcome the down force generated by the tailplane and elevators in causing the manoeuvre.

The spanwise distribution of the load can be calculated accurately using the Shrenk method which works out the arithmetic mean of an elliptical spanwise distribution and a distribution where the lift coefficient varies proportional to the local chord, modified to account for the effect of twist (washout) in the wing. However a lot of trouble can be saved by assuming a much simpler spanwise distribution, which if sensibly chosen can have surprisingly little effect on the overall results. For example a common choice is to assume the amount of lift is uniform between the wing root and a point one wing chord inboard from the wing tip, and then tapers linearly to zero at the wing tip itself. Reference 2 states that this distribution produces shear and bending results accurate to within very few percent for any typical light aircraft wing platform. This distribution also allows hand calculation of the beam-column behaviour of the spars of strutted and wire braced wings, including biplanes, which would be impossible using the actual Shrenk distribution. See Reference 5 for equations and example of this method. BCAR Section S (reference 4) allows an alternative simplification where the lift is distributed spanwise in proportion to the local chord – so that for example, with a simple parallel chord wing the lift distribution becomes uniform from root to tip, or with a 2:1 taper ratio the wing root generates twice as many pounds of lift per square foot of area than that in the vicinity of the tip.

The wing pitching moment can be calculated from the aerofoil pitching moment coefficient, dynamic pressure of the free stream and the wing span and chord, modified to account for the effect of deflected control surfaces using the methods shown by Hiscocks. A simple equation provided by CS-VLA and similar codes approximates the effect of deflected control surfaces, if this is not included in the aerofoil data. Alternatively BCAR Section S helpfully provides a default range of centre of pressure travel which can be used for microlights. While nowadays it is conventional to think of the lift acting at the quarter chord and an associated pitching moment rather than a shifting centre of pressure, the old fashioned centre of pressure concept still has value when dealing with fabric-covered two-spar wings and in wing load testing where the sandbag distribution has to mimic the effect of the chordwise distribution of the airload.

Inertia relief can be used to reduce the effective wing shear and bending loads, on the basis that the mass of each element of the wing causes an inertia force in the opposite direction to any vertical acceleration. A common mistake is to simply subtract the total wing weight from the aircraft gross weight when working out the wing loads, this assumes that the wing weight is distributed along the span with the same distribution as the wing lift. The correct approach is to divide the wing into a number of spanwise elements (typically ten strips) and then subtract the elemental inertia relief from the small portion of the total lift acting on each individual strip.

A very common mistake is to neglect the so-called anti-drag loads, which are substantial forward loads acting on the wing structure due to the forward component of the combined aerodynamic lift and drag vector when compared to the wing reference plane at high angles of attack. While the aerodynamic force on the wing is always rearward compared to the free stream direction, at high angles of attack the wing reference plane is tipped so far rearward that the aerodynamic force acts forward of the perpendicular. The critical point is usually point A of the flight envelope where the anti-drag force may be 20-25 % of the total lift force. The lower the wing aspect ratio, the higher the stalling angle and the more sever the anti-drag load. The Canadian design code TP10141 specifies an anti drag load of one quarter of the lift load, saving complex calculation. The anti-drag load is normally simulated in wing load testing by tilting the inverted wing nose-down by 12-13 degrees in its test rig when testing the high angle of attack case at point A of the flight envelope.

The fore and aft loading on the wing is also influenced by any wing bracing struts or wires that are arranged with any forward or rearward sweep. In the Kitfox and other folding wing designs for example, the forward lift struts sweep back to meet the fuselage at the rear strut attachment point, directly beneath the wing folding hinge of the rear spar. The result is that the tension in the front struts in positive g flight applies a strong rearwards load to the wing at the strut pick-up point, which tends to balance the effect of the wing anti-drag load. In this case it is usually the rearwards wing load at low angle of attack (point D of the flight envelope) which becomes a critical case, highly loading the wing internal drag bracing and creating significant extra compression loads in the inboard portions of the rear spars and rear spar carry through. With biplanes, the stagger loads must also be considered, these being the forward and aft loads in the individual wings generated due to stagger angle of the interplane struts. In positive g flight, conventional forward stagger causes additional forward load on the upper wing and aft load on the bottom wing, and vice-versa in negative g conditions.

Design codes often call for assymmetric wing lift cases simulating aileron deflection cases giving more lift on one wing than the other. This has implications for the fuselage attachments and wing bracing, as well as for the wings themselves. In a typical strut braced high wing monoplane, the asymmetric wing load tends to parallelogram the fuselage and shear the cabin sideways, loading up the front diagonals on something like a Cub or Auster, or on bolder designs like the Jabiru which don’t include these diagonals, loading up the windscreen itself in shear. Trying to do an asymmetric wing load test on a Jabiru type structure just wouldn’t work without the windscreen being fitted.

A common error is to neglect the extra safety factor applicable to bracing wires which are used internally within a wing making up the drag/anti-drag truss. Section S calls for an ultimate factor of 2.0 (rather than the standard 1.5) compared to the rated breaking strength of all cables used in primary structure, and in the primary flying controls. As it is not usually practical to test wings with twice the limit anti-drag load it is normally necessary to tackle this aspect by calculation. The same approach is needed for cables used for external bracing of wings and tail surfaces.

Another error is to assume that the wing torque will be critical in the nose-down sense due to the negative sign of the wing pitching moment coefficient, perhaps combined with the effect of the downward deflected aileron at high speed. On wings with the mainspar located well aft around 33% chord, you sometimes find that the 'forward centre of pressure' case at high angle of incidence, perhaps combined with the upwards deflected aileron, provides a greater torque about the wing's shear centre. When thinking in terms of wing pitching moment, you have to remember that the torque about the wing's shear centre is what the structure has to deal with, this is the pitching moment about the quarter chord point plus the moment due to the wing lift acting at the quarter chord point, the moment arm being the fore and aft distance between the quarter chord point and the shear centre.

On some aircraft, questions of structural stiffness arise as well as strength, due to concerns about structural divergence or flutter. BCAR Section K, an obsolete British design code for light aircraft, contains simple criteria for recommended torsional and flexural stiffness of wings, fuselages and tail surfaces and these are available from LAA Engineering on request.

Flaps and ailerons.

CS-VLA Appendix A and B provide quick means of estimating the flap and aileron loads and their distribution over the surfaces. It is strongly recommended that these methods are used, where appropriate, as this saves a great deal of time and technical argument. We have come some across wildly inaccurate estimates derived from other approaches over the years, so don't assume that methods others have used are technically valid.

The CS-VLA Appendix A and B methods are based on critical flight envelope airspeeds derived as a function of the aircraft's wing loading. See the section on tail loads, below, for caveats associated with this method.

A common mistake is to calculate the wing loads with flap and aileron deflection and then apply the ailerons and flap loads in addition. This effectively applies the flap and ailerons to the wing twice over, and drastically increases the wing torques. The flap and aileron loads are intended to be used to check the strength of the flap and ailerons themselves, their hinges and control systems. Their effect on the wing itself is dealt with via their effect on the wing pitching moment coefficient.

When checking the strength of aileron and flap hinges, (and also the rudder and elevator) remember that the highest loaded hinge is usually the one adjacent to the pushrod attachment, as this hinge has to react not only part of the control surface airload but also the majority of the pushrod load in a perpendicular ie fore and aft direction. Remember also that most design codes specify a pretty high safety factor for plain hinges, normally 6.67 on bearing strength. Presumably this is due to bad experiences in the past with rapid wear on plain hinges.

Particular care should be taken with flap hinges and flap systems due to the risk of inadvertent overstressing. While the pilot is generally conscious of the other control surface loads through his awareness of the muscular force required to apply the control operating force, flap controls are either mechanically locked in place using a gate system or controlled through an electrical or hydraulic system. If the pilot extends the flaps and then through carelessness allows the airspeed to exceed the flap limiting speed, it is all too easy to significantly overload the flaps and flap operating system. It is sensible to design the flap system so that if it is overloaded, both flaps retract symmetrically rather than risk just one flap raising, which could cause an uncontrollable roll.

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Tail Loads

The load cases considered for the tail surfaces of a conventional aeroplane are aerodynamic cases due to the sum of the balancing loads (the loads required to balance the aeroplane in a steady state condition) plus transient loads due to manoeuvring or gust cases. Fortunately, for simplicity it is not required to consider a combined manoeuvre and gust case (though this can occur in practice – for example when an aerobatic pilot encounters his own wake turbulence when pulling hard in the latter stages of a loop). Inertia relief could be claimed, as with the wings, to reduce the effective loads but for simplicity this alleviation is usually only brought into play were the structure is marginal and the designer needs all the help he can get to try to prove that it is OK. It is also normal to ignore aerodynamic drag and anti-drag loads acting on tail surfaces, but it is important that the designer bears in mind the in-plane loads that will be introduced into the tail surfaces if tailplane or fin bracing wires or struts are swept forward or aft.

Hiscocks in reference 1 provides a full description and examples of how the tail loads can be calculated individually for each corner point on the flight envelope, and then listed out so that the most critical cases (ie highest load values) can be extracted. Fortunately, CS-VLA provides a much simpler method for deriving conservative estimates for the tail loads which have been found over the years to be satisfactory for conventional types of aeroplane, and these are contained in Appendices A and B of CS-VLA. We always recommend that the simplified approach is used wherever possible, as it saves a great deal of time and argument. The method calculates the limit tailplane and fin surface load in terms of pounds per square foot (or kilograms per square metre) simply as a function of the aeroplane's wing loading and the maximum load factor, and then applies a small correction if the selected values of Vc or Va are greater than the minimum 'default' values defined by the simplified method. Having derived the maximum fin and tailplane loads by multiplying the calculated average surface pressure loadings by the appropriate fin and tailplane areas, Appendices A and B then show diagrammatically how the surface loads must be distributed both spanwise and chordwise over the fin and tailplane surfaces to represent firstly the manoeuvring case and secondly the gust case. The spanwise distribution is assumed proportional to the local chord. The chordwise distribution for the manoeuvre case simulates the effect of a deflected control surface with a is triangular distribution having its peak at the control surface hinge line, which throws a fair proportion of the load onto the fin or tailplane rear spar and the rudder or elevator. The gust case, in contrast, simulates the effect of the whole fin or tailplane experiencing a high angle of attack with a distribution peaking at the leading edge which results in the centre of pressure being at just 25% chord.

When using CS-VLA to calculate the tail loads, watch out for some confusing mistakes which have crept into the equations due to errors in 'metricating' it when it was adapted from the American code FAR 23, which is in imperial units. See LAA Technical Leaflet 'Guide to CS-VLA' for details.

When loading tail surfaces, you should fit the elevators and rudder both to check that the control surfaces remain free to move about their hinges when the tail surfaces are loaded, and also to benefit from the extra shear and bending strength provided by the control surfaces themselves. For example where a rudder is hinged to the fin with three or more hinges, the fin post can only bend laterally if the rudder front spar bends equally, therefore the rudder contributes to the strength of the fin.

Common mistakes made by LAA members in working out tail loadings include failing to understand that tail balancing loads and manoeuvring loads are additive, and assuming that the manoeuvring loads in pitch and yaw are only applied to the elevators and rudder control surfaces themselves. When the elevators or rudder are deflected, they affect the airflow and pressure field around the whole tail surface, not just over the control surfaces. Calculating an assumed manoeuvring tail load as if the control surface were in isolation from the tail surface which they are attached to results in seriously underestimating the actual load.

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Another common problem is to neglect the tail upload case and only test with download. Due to the shallower angle of the lower tail bracing wires, wire braced tailplanes in particular are sometimes critical under upload.

Certain design codes specify combined fin and tail loading, others call for special safety factors on hinges when only two hinges are used on control surfaces so you need to check the code carefully before planning your test schedule. A very common mistake is to ignore the special factors which the codes specify on hinges, cables, castings and the like.

Engine loads

The engine load cases required to be checked normally involve firstly, a combination of an inertia download with peak engine torque and secondly, an engine inertia side load case. The design codes call for maximum positive load factor to be combined with the peak engine torque at maximum continuous power, but interestingly only call for the full-throttle (or max power) engine torque to be combined with a somewhat reduced load factor – aerobatic pilots take note !

In the case of an engine fitted with a reduction gear or belt drive to the propeller, the mean engine torques used are the torques developed at the propeller attachment. The torque at the propeller attachment is greater than that at the crankshaft by a factor equal to the propeller reduction drive ratio – in stepping down the propeller rpm the reduction drive steps up the torque. When you take the torque figures from the engine manufacturer's spec sheets, be sure to check whether the figures quoted are at the crankshaft or at the propeller attachment, and correct accordingly. Alternatively, if you calculate the propeller torque by dividing the engine power by the rpm, remember to use the propeller rpm not the engine rpm to account for the effect of the reduction drive.

The peak torques are calculated by multiplying the mean torques by a magnification factor provided in the design codes, depending on the number of engine firing impulses per revolution of the crankshaft. Thus for example the peak torque factor is twice as high for a two cylinder four stroke than for a four cylinder four stroke or a two cylinder two stroke. Interestingly, in specifying the peak torque factors the design codes assume that the engines are arranged to fire their cylinders alternately but that isn't necessarily the case – an opposed two cylinder two stroke might for example be simultaneous-firing in which case the torque peaking would be the same as on a single cylinder two stroke.

The peak torque factor seems quite severe for normal operation, as the inertia of the engine itself probably largely damps out the firing pulses at the engine mount. However it is sensible to include extra robustness in the engine mounting to cope with scenarios such as very rough running (sticking valves etc), sudden engine stops (e.g. when a conrod breaks) or the massive vibration that occurs should the prop throw a blade. Applying a peak torque factor is therefore accepted standard practise even though its original justification seems questionable.

The inertia sideload case involves applying a sideways load to the engine equivalent to the powerplant weight multiplied by a factor which is usually one third of the maximum positive load factor. Hence for the average microlight which has a 4g maximum load factor, the sideload which must be applied to the engine is $4/3 = 1.33$ times the combined weight of the engine, gearbox, propeller, exhaust etc. The sideload must be applied to the effective centre of gravity of the powerplant installation, as seen in side-view.

Where the engine is mounted above and behind the cockpit, where if it were to detach in a crash it might crush the crew, the design codes include additional requirements about security of the engine attachment, normally calling for the engine to remain in place under a 15g horizontal deceleration (i.e. forward load on engine). With this layout, the crash case requirements (the design codes euphemistically call it an 'emergency landing') also call for the engine mountings to withstand prescribed more stringent downloads, uploads and sideloads.

The design codes for simple aircraft do not normally specify that the propeller thrust or propeller gyroscopic loads be taken into account. Nevertheless, there are some circumstances where these may be critical (e.g. an aerobatic aeroplane, or unusual engine mounting configuration) in which case these loads will also have to be checked.

Common mistakes in working out the engine load cases include calculating inertia loads based on engine weight alone rather than total installed powerplant weight (i.e. everything that hangs off the engine mounts), failing to include the effect of the propeller reduction ratio and applying the loads in the plane of the engine mountings rather than in line with the centre of gravity of the powerplant, and failing to consider a casting factors in relation to cast engine crankcases. Where an engine type has previously been approved in an equivalent application it is not necessary to re-check the strength of the engine itself however.

Fuselage loads

The fuselage is normally considered merely as a structure for supporting all the other parts of the aircraft i.e. to transfer loads between the payload and the wings, tail, engines, fuel tanks, undercarriage etc. The airloads acting on the fuselage itself are usually ignored, and with light aircraft it is also common to neglect the inertia relief provided by the mass of the fuselage structure. Where the strength of the fuselage is checked by load testing this is often combined with the testing of other components, for example when strut-braced wings are fitted it is common to load test the wings while mounted onto a fuselage, so that the strength of the wing struts, wing attachments on the fuselage and the wing and strut carry-through structures in the fuselage are tested at the same time. Alternatively, the wings may be tested separately on a test rig and crude dummy wings (normally fabricated from welded steel box sections) are used when load testing the fuselage.

With a conventionally configured aircraft, the bending and torsional strength of the rear fuselage is normally checked by applying the worst case tail loads to the tail surfaces (or dummy tail surfaces) with the fuselage restrained in the vicinity of the wing centre-section or cabin. Similarly, the forward fuselage bending and torsional strength is checked by applying the worst case engine inertia and torque loads to the engine mounting structure while the fuselage mid-section is restrained. In the case of an aircraft with an unusual layout, such as a pusher with a cabin forward of the wings (e.g. Kolb Twinstar) it would be necessary to check the forward fuselage bending strength simulating the maximum inertia load on the forward cockpit and crew.

Rear fuselages need testing separately under the tailplane up load and down load cases - except perhaps in the case of very simple tubular tailbooms where the load carrying capability is clearly equal in either direction. It is also important to recognise that fuselage structures may not be symmetrical to port and starboard of the centreline – particularly with truss-type construction where applying a side load in one direction puts fuselage diagonals in tension, the other way in compression. Changing the orientation of the load changes the sign of the load in all the diagonals, with possibly a quite different structural capability. In this case, side load cases will need to be separately tested under port and starboard fin side load directions, unless it can be agreed beforehand which load direction will be the critical case.

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Common mistakes made in testing fuselages are failing to restrain the fuselage in an appropriate way so that the restraint structure unwittingly contributes to the strength of the fuselage under test, or failing to find sufficiently strong points on the fuselage to retrain it by during the test, or neglecting to cover all the test cases required due to missing the fact that the structure is asymmetric.

Undercarriage

The undercarriage must be designed to absorb the impact energy involved when the aircraft strikes the ground at a specified vertical speed, which approximates to a normal approach to land but forgetting to flare – ouch! The undercarriage must also be compliant enough to shield the remainder of the airframe from excessive loads – it is no good having an undercarriage which is strong but also so stiff that in a heavy landing the engine drops off the front of the aeroplane, or the pilot gets a back injury. The more the suspension travel on the undercarriage, the more soft the suspension can be for a given amount of energy absorption, and the lesser the impact loads that the undercarriage will transfer to the airframe and its unfortunate occupants.

If the suspension force versus deflection curve for the undercarriage has either been calculated or measured, the energy absorption can be checked by calculation. Alternatively it can be checked by carrying out a drop test. In the drop test, either the whole aeroplane, or its undercarriage attached to a suitable rig, is dropped at a prescribed weight from a prescribed height above the ground, and the undercarriage travel and airframe deceleration are recorded as well as checking for signs of damage. In some cases the loaded weight in the drop test is less than the maximum gross weight, which is to compensate for the fact that in an actual landing the wings are still supporting part of the aircraft's weight, rather than dropping like the proverbial sack of potatoes. Confusingly, the amount that the weight must be reduced for the drop test depends on the amount of travel you get in the undercarriage during the test, so you have a chicken and egg situation. The only way to deal with this is to carry out a series of tests which iterate to the correct drop weight corresponding with the amount of deflection obtained.

In the drop test, greased plates are normally required to be placed under the wheels so that when the wheels touch down they are free to slide sideways over the ground. This is because with many undercarriages (e.g. Piper Cub, Cessna 152) the wheels move outboard as they deflect upwards. If the wheel is restrained from sliding sideways over the ground in the drop test by the friction of the tyre (this is called the tyre scrubbing force), the leg will appear to be stiffer in vertical travel than it would be if the tyre is free to slide. In a real landing of course, the wheel is turning which allows the wheel to move sideways over the ground more easily than it can with a static wheel. Drop testing without using greased plates may make a weak undercarriage appear strong enough, hence the certifying authorities usually insist on the greased plates being used.

As well as checking the energy absorption, it is necessary to show that the forces that occur in the specified landing cases are acceptable as limit loads, and that multiplying these forces by 1.5 produces satisfactory ultimate loads. In some cases the ultimate case can be proven by dropping the aircraft from a height 2.25 times as great as the limit case, which is intended to generate 1.5 times the loads in the undercarriage. However you need to be careful with this – if the aeroplane is something like a Cherokee, Condor or Jodel with telescopic type undercarriage legs, or limited travel undercarriages like the Piper Cub, then increasing the drop height much above 'limit case' may mean that the undercarriage runs out of travel and 'bottoms' which will cause unrealistically high loads and wreck the undercarriage and/or the aeroplane. Unrealistic results also occur with aeroplanes with undercarriages with highly non-linear force/deflection characteristics, such as the Cessna 152, where an ultimate drop height test may flatten the undercarriage out completely so that the wheels appear in the side-windows!

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In some design codes the so-called 'reserve energy' case covers this scenario, being a case with 20% more impact energy than the limit case, which is allowed to cause some permanent damage to the undercarriage but not a complete collapse.

The undercarriage checks include coping with landing impacts in different attitudes, including level and tail-down, and also normally include prescribed side load cases (as in, for example, a ground loop or untidy cross wind landing) and also fore and aft wheel load cases. Rearwards wheel loads are obviously generated when landing on rough ground or when braking heavily. These are sometimes referred to as spin-up loads, this is a misleading carry-over from Boing or Airbus designing mentality where significant forces are involved in spinning-up the multiple huge wheels almost instantaneously from zero to the wheel rpm corresponding with touchdown speed. Forward wheel loads are encountered in the rebound situation (wheel pushed back on impact, springs forward on rebound) or when turning 'on the spot' on the ground with brakes on. The design codes call for the various loads to be applied either singly or in combination and it saves time if you can safely assume which are the worst cases which need to be checked. For example if the undercarriage is sharply swept forward (as in the Bucker Jungmann) then it is fairly certain that the tail-down landing will be more critical than the level landing, as the tail-down attitude will accentuate the effect of the rake, whereas with the swept-back gear on an RV4 the level landing case would most likely be the more severe case of the two.

With microlights, the probability that these aircraft will be operated from rough fields has lead to a particularly severe requirement for microlight aircraft's nosewheels to be able to cope with significant rearward impacts, as for example landing in a ploughed field – not that we would recommend such action with today's high-speed microlights. The microlight design code BCAR Section S suggests that in the absence of a better estimate, the nosewheel undercarriage should be able to cope with an impact from the front with energy equivalent to half the total vertical impact energy from the limit drop test. This is usually tested by dangling the aircraft from wires attached to the roof truss of a hangar or workshop building, using a bridle arrangement to keep it in a level attitude. The aircraft is loaded to a selected weight, swung backwards in an arc until the centre of gravity of the aeroplane has been raised by an amount corresponding to the required potential energy. The aircraft is then released allowing it to swing forward and impact the nosewheel against a solid concrete block or steel framework. Great fun, with somebody else's aeroplane! For safety's sake, have an arrestor wire or a judiciously sited pile of hay bales to bring the aircraft safely to rest in case the nosewheel misses the block.

Controls

The design codes call for the control systems to be able to cope with the maximum loadings seen under various flight conditions. CS-VLA Appendices A and B provide simple methods to evaluate the control surface hinge moments, from which the forces in the various elements of the control system can be calculated by simple mechanics, based on the lever ratios of the various cranks and control horns. However with LAA-sized light aircraft the strength of the flying control systems are almost always determined by the 'pilot effort' cases i.e. the need for the system to be able to cope with the maximum forces that the pilot could apply to the controls if he was trying to clear a jam at the far end of the system, for example due to a screwdriver caught up in an elevator bellcrank. These 'jam case' control system forces far exceed the loads encountered by the control system in normal use, as the aerodynamic loads generated by the control surfaces are comparatively small. While the possibility of jamming may seem quite remote, accidents do continue to occur due to control jams and in any case, designing to these pilot effort forces does go some way to ensuring that the system is generally robust enough to cope with years of use and abuse without failure. Given that flying control system failures are likely to have fatal consequences, it's wise to build in generous safety margins.

More substantial control system forces may occur in flap controls particularly if they are power driven rather than manually operated, and it is sensible to include an extra margin with flap systems to cope with inadvertently exceeding the flap limiting speed in flight, or horizontal gusts while flaps are deployed.

The pilot effort load cases specified in the design codes differ from one code to another, for example BCAR Section S requires that the full pilot effort loads only need to be carried between the pilot's controls and the control system stops, and from there on in the system (ie between the stops and the control surface operating horn) only 60% of the pilot effort load need be carried.

Where necessary, control systems can be tested for the aerodynamic loading cases by restraining the stick, rudder pedals or flap lever and then applying load distributed onto the appropriate control surfaces to simulate the aerodynamic load case in question, e.g. using sandbags or other weights tied to elevators, ailerons or flaps, or spring balances or load cells on the rudder. For jam case tests, the control surfaces are restrained in neutral with clamps and the jam case loads applied to the pilot's controls manually (using the local strong-man) or a winch and in each case using spring balances or load cells to measure the applied load. In aeroplanes designed to BCAR Section S, if the controls are cable operated, when planning the load tests of the flight controls the cable factor specified in this code must be taken into account. In aeroplanes with dual controls, the design codes call for the simultaneous application of 75% pilot effort loads on both sets of controls, applied in unison and in opposition. Some codes also call for a special load case with the pilot pressing on both rudder pedals together, as pilots are aft to do when under pressure !

After checking the control systems with the control surfaces clamped neutral, a second set of tests is then made with the control surface clamps released, to check the strength of the control system stops.

Typical problems in checking control systems include buckling of long and slender unsupported elevator or aileron pushrods, the collapse of understrength bellcranks and undue movement in feeble bellcrank or pulley mountings. These issues very often crop up with foreign designed aircraft as microlight design codes used elsewhere in Europe have much less severe jam case requirements than those in BCAR Section S and CS-VLA. Problems with pushrods which buckle in compression can often be sorted out by adding one or more strategically placed intermediate sheaves or an idler crank to support the pushrod at the mid point along its length.

As well as testing the strength of the flying control system, it is often required to check that the system is not too compliant, for example due to stretch in control cables or distortion in the cranks and pivot bracketry under load. Too much springiness in the system means that the pilot doesn't get direct control over the position of the control surfaces, making it more difficult to achieve precise attitude control, particularly noticeable when trying to position the aircraft accurately in relation to another object – when landing, for example, or in formation flying. Springy controls also make control surface flutter much more likely, as any resonance of the control system with the airframe vibrations will be impossible for the pilot to damp out by holding the controls fixed. Finally, if the pilot should need to apply heavy loads to the control system in an emergency, he can't do so if the stick or pedals move to the stop because the control system is stretching. The control system stretch tests are carried out by first measuring the amount of movement available stop-to-stop at the pedals and stick with the control surfaces free, and then locking the control surfaces and measuring the movements of the stick and pedals obtained under prescribed loading conditions, and then comparing the free and fixed deflections. Section S for example calls for the deflections with surfaces fixed to be no more than 25% of those with surfaces free, when subjected to pilot effort loads.

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Seats

The design cases for the seats are normally the download on the seat base due to the inertia of the occupant under positive g load, and for the seat back, the need to react the maximum pilot effort forces on the controls – it is no good having a control system strong enough to allow you to push 150 Lbs on the stick if in doing so the seat back collapses. The maximum positive g case which is used for the vertical load on the seat base may be the crash case (emergency landing) or either a manoeuvring or gust-induced flight condition – you have to study the flight envelope for the particular aeroplane involved and select whichever case gives the greatest vertical acceleration. When looking at the gust case you should use the gust acceleration that results at minimum rather than maximum gross weight. Whereas the gust loads in the wings and airframe structure are greatest at maximum weight, the gust response (i.e. acceleration) of an aeroplane is greatest at minimum weight, so this condition causes the largest forces on so-called ‘dead weight’ items such as crew members, batteries, fuel tanks etc.

When using plastic or glass fibre seats, a composite superfactor must be applied – typically a room-temperature composite superfactor of 1.5 is used. Particularly with aerobatic aircraft with plastic seats, the seat test loads can become very high. With a metal or wood fuselage, it may be prudent to test the seat on a separate rig rather than in a fuselage, to avoid overloading the fuselage in the test by virtue of the composite superfactor associated with the plastic seat.

The crash case accelerations quoted in common design codes are surprisingly moderate compared to what a fit person can survive, so it is worth designing seats which will support the user under 10g or more even though the design code ‘worst case’ may only require 6g. The seat should collapse progressively as a unit, cushioning the pilot’s back from the impact. A plastic seat which shattered and allowed the pilot to thump down onto hard supporting metalwork was sited as a contributory factor in a fatal accident report a few years ago. While the more advanced design codes these days include stipulated energy absorption requirements, due to the cost and complexity of demonstrating compliance we have not yet included these for LAA type aircraft. It is however important that there be energy absorbing structure between the bottom of the aeroplane and the seat, or that the seat itself can absorb crash impacts otherwise the crew’s backs will be at risk in a simple undercarriage collapse.

Seat Harnesses

The seat harness loads are specified in the various design codes in terms of the accelerations experienced by the aircraft in manoeuvre, gust and crash cases (known as ‘emergency landing’). Normally, as one would expect, it is the crash cases which define the seat harness loads but in an out-and-out aerobatic aeroplane the forces in the harness due to the manoeuvring loads on the pilot under negative g will need to be checked. The design codes do not specify how the pilot restraint forces are shared between the lap strap and the shoulder harness. In the absence of better information, with a conventionally configured harness arrangement a conservative assumption is to assume that the lap strap has to carry 100% of the vertical, fore-and-aft and lateral forces on the pilot when checking the lap strap, and that the shoulder harness has to carry 60% of the fore-and-aft load on the pilot when checking the shoulder harness.

It is important to remember that the forces in the belts can be substantially greater than the forces on the wearer, due to the fact that the belts are not in line with the inertia forces on the wearer’s body. For example, if the lap strap passes over the pilot’s thighs at a typical 45 degree angle seen in side-view, by simple trigonometry the tension in the lap belt is be factored up by 41% compared to both the vertical and horizontal inertia forces acting on the pilot. Bear in mind also that design codes often call up a special safety factor for harness attachments which will need to be included in the calculation.

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Fuel Tanks

Fuel tanks are so called 'dead weight' items and the fuel tank mountings need to be checked for the worst case loads caused under manoeuvre, gust and emergency landing conditions as specified in your design code. This assumes the fuel tanks are located somewhere near to the aeroplane's centre of gravity. If the tanks are positioned near to the aeroplane's extremities, such as a wing tip tank, then you must also check the loads induced by angular accelerations about the cg. For example with tip tanks, the roll acceleration which occurs on sudden application of full aileron, or on making a bumpy one-wheel landing, typically causes significantly higher vertical load on the tip tank than would ever be seen in a normal pull-up manoeuvre. Interestingly, tip tanks also experience significant outboard loads when the aeroplane is rolling, due to the centrifugal force on the tank and its contents.

Another issue to consider with fuel tanks is the hydraulic pressure in the tank caused by the fuel in the tank under acceleration. The magnitude of the pressure generated depends on the 'head of fuel' in the direction concerned. For example in the case of a typical wing leading edge tank, in a 'sudden stop' generating a rearward acceleration, the hydraulic pressure is the product of the fuel density, the horizontal acceleration and the front to back dimension of the tank. A tank which is long and flat will produce maximum hydraulic pressures in a 'sudden stop' type accident whereas a tall tank will be 'worst case' under vertical acceleration ie a heavy landing.

Baggage compartments.

Baggage compartments should be checked to ensure that the baggage will be adequately restrained under flight and crash case conditions in much the same way that that the crew seats and seat harnesses are checked. This is particularly important where the baggage compartment is located above or behind the crew, for obvious reasons. When tests are carried out, the load should be distributed over an appropriate area of the baggage compartment floor or front face to simulate the size and density of likely baggage. With a canvas baggage compartment, for example, a 5 kg tent roll will most likely be a very much less severe test of the strength of the canvas than, say, a 5 kg toolbox, because of the latter's smaller footprint. In other words, don't assume that the baggage weight is spread thinly over the entire area of the baggage bay floor, you need to look at a concentrated load.

PART 2 STRUCTURAL TESTING OF AIRCRAFT

Despite the existence of all of the analytical methods which have been developed over the years to predict the strength of aircraft, load testing aircraft structures remains a common practice throughout the industry, from Boeing and Airbus to the smallest manufacturer, and also for the humble amateur aircraft designer working on a one-off project in his garage. In the LAA world we come across load testing most often as kit manufacturers, or their UK agents, seek to prove that their products meet the requirements for LAA acceptance.

Load testing can be used by itself to prove structural strength, and this is the approach taken by many microlight manufacturers and kit suppliers. Alternatively load tests may be carried out to supplement the designer's structural analysis, to give confidence in the theoretical results. Whether it is necessary to carry out load testing in addition to structural analysis depends on many factors, including the type of structure involved, the experience of the designer and the resources he has available by way of finite element programs and stress data sheets. Some type of structure lend themselves to analysis and the strength can be predicted quite quickly and easily with a high degree of confidence – for example the bending strength of a classical wooden box spar. Other structures, such as stress skinned metal wings, are very much more difficult to analyse when the failure modes involve buckling of thin sheet metal panels, where the onset of buckling is critically influenced by the support provided by a variety of stiffeners, plus the stiffening effect of curvature in the panels, and the inevitable presence of access holes, doublers etc which further complicate matters. With this type of structure, even the most confident of stress analysts will probably want to see load tests carried out to check the validity of his theoretical predictions.

Over the years, design codes have been developed which tell designers what are the appropriate strength factors they must work to, to achieve an aircraft safe enough for certification. For LAA type aircraft, the commonly used design codes are CS-VLA for aeroplanes, BCAR Section S for microlights and BCAR Section T for gyroplanes. These codes can be downloaded via links from the Engineering pages of the LAA website.

Load testing is often carried out by designers in two steps – firstly to a reduced load level sufficient to give enough confidence to start the test flying program, and later, once the design has been shown to be otherwise viable, to the full design code requirements for certification. Sometimes the actual flying prototype is used as the test sample for carrying out the preliminary load tests. The benefit of this approach is that it avoids having to make two airframes (one for destruction, one for flight) before the prototype can fly. On the down side, it means that the flight envelope has to be restricted during the flight testing phase and there will be a reduced margin of safety during the flight tests. Also, if structural problems come to light in the final phase of the load testing then it may be much more expensive to carry out drastic modifications at this stage of the program when the aeroplane is otherwise fully developed. It is generally better in the long run to cover as much as possible of the structural load testing, on a sacrificial structural test airframe, before the flying prototype first leaves the ground.

When carrying out load testing, firstly it is important to check that the structure being tested is representative of the design. If the structure is a dummy sacrificial one it may be tempting to cut corners with the build, using cheaper substitute materials, unairworthy parts from the scrap bin and perhaps leaving out secondary components or features which may nevertheless significantly effect the way the structure behaves under load. For example when testing an elevator in torsion, if the elevator has a cut-out in the trailing edge to accommodate a typical trim tab then it will be important that this feature is in place in the structural test elevator as this cut-out may well be the area that will first fail in a torsion test.

An important issue is to decide what load levels are to be used for the testing. Design codes specify two equally critical structural goals that must be achieved, known as limit and ultimate cases. Limit load is the load that the pilot is allowed to apply to the structure from time to time, and at this load level there must be no permanent damage to the structure. It must be possible to take the structure to limit load without causing permanent deformation on a macro scale, for example while the flying surfaces will curve nicely under limit load, once the load is taken off they must go back to dead straight. There must also be no local failures at limit load, for example rivets must not pull out, wood members must not split, or glued joints pull away. Whilst the thin skins of a sheet metal structure will most likely become covered in ripples and dents at limit load, due to compression buckling and diagonal tension fields, once the load is removed the ripples and dents must disappear without trace, leaving the structure as good as new. The only exception to this is that given enough load applications, fatigue damage may accumulate at a microscopic scale and initiate cracks – but fatigue is beyond the scope of this article.

Ultimate load is usually 50% higher than limit, and this provides a quite slender safety margin intended to cover inadvertent overloads by the pilot, errors in the stressing calculations and degradation in the strength of the airframe through errors in build or through ageing effects. Ultimate load is the load which the structure must be able to carry for a very short period (usually 3 seconds) without collapsing. An airframe which has been taken into the region between limit and ultimate load may be permanently bent and have local breakages, creased skins, pulled rivets, stretched lugs etc, but it must be in a fit state to bring the pilot home, whereupon it will probably be fit only for scrap.

When a sacrificial test airframe is load tested, it is always tested to limit and ultimate load, checking compliance with both criteria. The fact that a structure passes the limit load tests without damage does not prove that it will pass an ultimate test without failure – certain types of failure demonstrate collapse with no prior warning by way of yield or local damage. Stressed skin structures in buckling are in this category, as are many composite structures.

Conversely, passing an ultimate load test without collapse does not prove that there was no damage at limit, so it is essential to inspect the structure carefully for distortion and damage after reaching limit load, before raising the load level to ultimate. Where wing spar fittings are bolted to wooden spars, for example, the bolts may well be bent and the bolt holes in the wood 'stretched' at load levels less than half that which eventually causes the bolts to break or the wooden spar to break. Therefore after a limit load test it is essential to check critical fittings for signs of any movement, withdraw critical bolts to check their straightness etc. A quick eye-balling of the structure is not enough.

Design codes sometime specify that the loads used in load testing are increased to cater for variability in the structure's strength, to make sure that safety levels are still preserved even when the structure is weakened by factors such as high temperature (eg a composite wing that has softened through baking in the sun) or unseen internal defects (possibly porous castings). For example when load testing a composite structured microlight it is normal to apply 50% higher loads than would be used if the airframe was manufactured from wood or metal, through applying the so-called 'composite super-factor' of 1.5 as specified by BCAR Section S.

This can cause problems because the composite super-factor strictly only needs be applied to the composite components, and not to the metallic parts of the structure such as brackets, nuts and bolts etc. Thus when the load levels are set by the need to test the composite components effectively, the metallic parts of the structure will be overloaded. Designers wishing to prove the strength of composite airframes by load testing have to either over-design the metallic parts to cope with the unrealistically high load levels they will be subjected to in the tests, or fit beefed-up metallic parts during the load testing just to prevent these parts breaking before the composite parts have reached their factored ultimate load.

When load tests are carried out on an airframe intended for flight, load levels are generally restricted to limit load only, or even less, since clearly it is not sensible to fly an aeroplane which has been purposely loaded to the point where it has been structurally damaged. The only exception to this is with structures that have been designed with very high safety factors, so the designer is very confident that damage will not occur until the loads are substantially above limit. Great care must be taken to avoid causing unseen damage to the structure which could spread later, for examples, cracks or sheared rivets in metal structures, delamination in composite ones, or compression shakes in wooden parts.

While in some countries the authorities favour limit load testing of every single homebuilt aircraft that is built, in this country we rely on our strength investigation of the 'UK prototype' and our inspection system to check the quality of build rather than submitting every homebuilt to the indignity and possible risks of limit load testing.

Calculating the amount of load that must be applied in a load test is an important part of the process. If various simplifying assumptions are made, 'worst case' loads can quickly be calculated. Often it is found that carrying out more complex calculations instead of making simplifying assumptions means that alleviating factors can be brought into play and the load levels come down. Sometimes a structure fails a load test but the designer can go back to his original crude (and hence over-pessimistic) calculations, burn some more midnight oil, and come up with alleviating factors which bring down the required load figure. By this means a clever stressman can sometimes turn a failed test result into a triumphant success.

Loads may be applied to structures using a variety of methods. The traditional sandbag method of testing wings and tail surfaces remains popular amongst amateur designers. Alternatively, the load may be applied using hand winches and hydraulic jacks, often with spring balances or load cells to measure the forces being generated. In this case, loads are often distributed using a system of balancing beams (known as a whiffle tree) where the relative lengths of the arms of the beams determine how the loads are apportioned between the cables attached to either end.

During load testing it is very important to load the structure or component in a representative way. The load must be properly distributed over the surface of a wing or control surface, to model the effect of airloads and inertia loads acting upon it. When applying a point load such as when simulating the pilot effort forces on the control column, the load must be applied at the centre of the stick grip, not at some other arbitrary point beneath the grip, just because the hook on your spring balance won't quite go around the grip part.

When loading bricks or sandbags onto a wing, it is essential that the bricks or sandbags are piled in separate stacks rather than keyed together in an overlapping fashion, like a wall, as the wall would act as a structure in its own right and confer unrealistic effective bending stiffness to the component. Similarly when using plywood sheets to spread the load when piling weights onto a wing or tail, it is important that the plywood is thin enough to flex to conform with the surface, and that the weight of the plywood is taken into account. It is important that the load is applied in the correct location and cannot slide out of place, particularly as the structure deflects. The canard on a composite aircraft broke on test because as the surface flexed under the weight of the sandbags, the sandbags started to slip outboard towards the tip en-masse, increasing the bending moment to the point of failure.

In some cases tapes or strings are used to restrain the weights against sliding. In others, polystyrene wedges between the slippery, sloping wing surface and the sandbags were found sufficient to prevent any sliding. Old bits of carpet underlay or anything else you have to hand which will provide a non-slip surface will help. A little ingenuity goes a long way in this particular field of aeronautical engineering!

The way in which a structure is restrained for testing must also be appropriate, and sometimes this can be very difficult to achieve. The forces in an aeroplane structure are normally shared amongst a great many components and it is often hard to find two or three individual parts strong enough to grab it by – it's rather like trying to jack up a rotten car body without jacking points! Welded steel tube space frame structures are the best in this respect, as suitable node points can usually be found to attach to, whereas monococque wood, composite or sheet metal structures can be very difficult and it may be necessary to build up special fixtures to spread the attachment loads over an adequate area of the structure.

Each part of the structure commonly needs to be tested several times to check different conditions, for example a wing may need testing in worst-case positive shear and bending, negative shear and bending, plus a max torque case, so that covering limit and ultimate conditions for all three cases requires no less than six tests. It is important to think carefully about the order in which the various tests are to be carried out, so as much use can be made of the test article as possible, leaving the high-risk tests until last – you don't want to wreck the structure at the first shot. Breaking the test wing in a torque test may leave the test article repairable, whereas if the spar fails in the bending test you are probably going to ruin the wing completely, so your best plan is to leave the ultimate positive bending test until last.

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Preparing the load schedules beforehand is important. Carrying out load tests is far too stressful an exercise to combine with crunching a lot of numbers on the same day, so you need to work out all the tests loads in advance and write out the schedules carefully so there is no confusion on the big day.

If you are carrying out the tests to satisfy a regulatory body on a certification issue (e.g. as part of a new LAA type acceptance program) then you should get the load test proposals agreed in advance with them – also, find out whether they want to send a representative to come and observe some or all of the tests, to check that what you are doing is valid, or whether they want the tests witnessed by your LAA inspector, for example.

When the time comes to do the tests, where large loads are involved you must find a way of supporting the structure while the loads are being applied, in such a way that the support can be progressively withdrawn once the load is in place. For example, in a wing test it is normal to support the wing tip with a jack which can be remotely inched up and down. Without this simple precaution, firstly the test article may be overloaded by the act of dumping weights on the surface, but more importantly, if the component fails when the weights are being loaded on then the person putting the weights in place may well be injured by the collapsing structure. Much better to place the weights on the wing while it is supported, and then stand all personnel clear while the supporting jack is progressively lowered. Another benefit of this arrangement is that should the structure fail when the supporting jack is being lowered, the surface will only move a short distance before coming up against the jack again, which will prevent it moving any further and so limit the extent of the damage to the test article. In effect, control of the supporting jack governs directly the amount of strain in the structure.

The test set-up should ideally include means of measuring the deflection under load, and the deflections recorded for each load increment. If you choose to take the load up in several steps (2g, 3g, 4g etc) then the deflections are best plotted out as the test is taking place, rather than later, as this way you may be able to spot any discontinuity or kink in the load/deflection curve which indicates where there may have been an internal failure in the structure, or the onset of yield. Plot the deflection after unloading too, which will show up whether there has been any permanent deformation during the test. It's a good idea to load up the structure initially to about half of limit load and then remove it again, before starting on the test proper. This allows the structure and its restraint system to settle in and take up any slack which would otherwise confuse the displacement measurements.

During the testing you must be alert to noises from the structure which tell of damage or impending failure. With composites it is normal to hear occasional clicks or bangs from the loaded structure indicating where the odd over-stretched fibre has pulled away or blob of glue has popped off, but watch out for the more serious sounds of joint separations or delaminating plies. Likewise with wood and metal structures, only by experience will you come to distinguish the sounds of the structure 'taking the load' from the noises made by actual damage, or worse, the ominous warnings of imminent collapse.

Keep an eye on the structure as far as you can, but not at the expense of your own safety. It is most unwise to crawl underneath a wing wobbling under the influence of perhaps several thousand pound weight of breeze blocks, however interesting it might be to have a close-up look at the buckling in the wing's upper skin!

Apart from the perhaps obvious risk of being struck by falling weights and collapsing structures, you must consider other health and safety implications including the risk of being impaled by flying splinters from disintegrating wood or composite components, having your toes crushed by overturning jacks and so on. With such great loads being applied to lightweight structures at or close to breaking point, there are opportunities for injury around every corner so extreme care and a disciplined approach must be taken to contain the risk.

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Strong boots and proper protective clothing will be more appropriate than a tee shirt, shorts and flip-flops, whatever the ambient temperature of your test house! Don't underestimate the physical labour involved in moving several thousand pounds weight of sandbags, perhaps many times over. Lifting this much weight over and over again from floor level to an elevated test rig is a recipe for back injuries, so make sure you have a fit and willing (but above all, disciplined and properly briefed) team of helpers on hand to share the burden. Exhausted volunteers, bent-double with pain tend to make mistakes so don't try to rush the program by doing too much in any one day.

Whilst it is important to plan the tests in detail in advance, you should be prepared to re-jig the program in mid-test if there is an unexpected development. If a failure starts to happen, if you stop the test and carry out a minor repair and structural reinforcement you may find the test article can go on to maximum load without further damage. Or you might choose to move on to other tests on the slightly damaged test article to see if it can be used to 'tick off' some of the other load cases before going back to the critical test for the final 'make of break' load increment.

Load tests have no value if they are not properly recorded so that there is permanent proof of what has been done, so you will need to have photographic records as well as writing up a detailed report. Standard procedure is to include in each photo a board marked-up to identify the test case involved (e.g. 'ultimate symmetrical manoeuvring tail load, 485 Lbs per side') and a common mistake is to use too small or too faint lettering on the identification boards so the numbers are not clear in the photos. Such details are important and in all the excitement of carrying out the tests, such niceties are easily forgotten. Your best plan is to allocate one member of your team solely to recording the results of the testing in writing and photographically. In some cases, such as undercarriage drop testing, a video camera is needed to record the event, and prior experiments with optimum camera angles, illumination, tripod arrangements, background screens etc will pay dividends on the day. It's a great shame when the results of a significant or expensive and time consuming test are spoilt by a dark or indistinct video record where you can hardly make out what the aeroplane component is let alone what is happening to it in the test.

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