Standard Specification for Design of Weight-Shift-Control Aircraft[1]

This standard is issued under the fixed designation F2317/F2317M; the number immediately following the designation indicates the year of original adoption or, in the case of revision, the year of last revision. A number in parentheses indicates the year of last reapproval. A superscript epsilon (ε) indicates an editorial change since the last revision or reapproval.

1. Scope

1.1 This specification covers the minimum airworthiness standards a manufacturer shall meet in the designing, testing, and labeling of weight-shift-control aircraft.

1.2 This specification covers only weight-shift-control aircraft in which flight control systems do not use hinged surfaces controlled by the pilot.

NOTE 1—This section is intended to preclude hinged surfaces such as typically found on conventional airplanes such as rudders and elevators. Flexible sail surfaces typically found on weight-shift aircraft are not considered hinged surfaces for the purposes of this specification.

1.3 Weight-shift-control aircraft means a powered aircraft with a framed pivoting wing and a fuselage (trike carriage) controllable only in pitch and roll by the pilot's ability to change the aircraft's center of gravity with respect to the wing. Flight control of the aircraft depends on the wing's ability to flexibly deform rather than the use of control surfaces.

1.4 This specification is organized and numbered in accordance with the bylaws established for Committee F37. The main sections are:

1.5 The values stated in either SI units or inch-pound units are to be regarded separately as standard. The values stated in each system may not be exact equivalents; therefore, each system shall be used independently of the other. Combining values from the two systems may result in non-conformance with the standard.

1.6 *This standard does not purport to address all of the safety concerns, if any, associated with its use. It is the responsibility of the user of this standard to establish appropriate safety and health practices and determine the applicability of regulatory requirements prior to use.*

2. Referenced Documents

2.1 *ASTM Standards:*[2]

F2339 Practice for Design and Manufacture of Reciprocating Spark Ignition Engines for Light Sport Aircraft

F2506 Specification for Design and Testing of Light Sport Aircraft Propellers

2.2 *Federal Aviation Regulations:*[3]

FAR-33 Airworthiness Standards: Aircraft Engines

FAR-35 Airworthiness Standards: Propellers

2.3 *Joint Aviation Requirements:*[4]

JAR-E Engines

JAR-P Propellers

JAR-22 Sailplanes and Powered Sailplanes

3. Terminology

3.1 *Definitions—Aircraft Weight:*

3.1.1 *design maximum aircraft weight, n*—aircraft design maximum weight W_{MAX} shall be the sum of $W_{WING} + W_{SUSP}$.

3.1.2 *design maximum trike carriage weight, n*—design maximum trike carriage weight, W_{susp}, shall be established so that it is: *(1)* highest trike carriage weight at which compliance with each applicable structural loading condition and each applicable flight requirement is shown, and *(2)* not less than the empty trike carriage weight, W_{tkmt}, plus a weight of occupant(s) of 86.0 kg [189.6 lb] for a single-seat aircraft or 150 kg [330.8 lb] for a two-seat aircraft, plus the lesser of full usable fuel or fuel weight equal to 1-h burn at economical cruise at maximum gross weight.

[1] This specification is under the jurisdiction of ASTM Committee F37 on Light Sport Aircraft and is the direct responsibility of Subcommittee F37.40 on Weight Shift.

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[2] For referenced ASTM standards, visit the ASTM website, www.astm.org, or contact ASTM Customer Service at service@astm.org. For *Annual Book of ASTM Standards* volume information, refer to the standard's Document Summary page on the ASTM website.

[3] Available from Federal Aviation Administration, 800 Independence Ave., SW, Washington, DC 20591.

[4] Available from Global Engineering Documents, 15 Inverness Way, East Englewood, CO 80112-5704

3.1.3 *trike carriage empty weight,* W_{tkmt}, *n*—all parts, components, and assemblies that comprise the trike carriage assembly or that are attached to the suspended trike in flight, including any wing attachment bolts, shall be included in the trike carriage assembly empty weight, W_{tkmt}. These must include the required minimum equipment, unusable fuel, maximum oil, and where appropriate, engine coolant and hydraulic fluid. Trike carriage empty weight, W_{tkmt}, shall be recorded in the Aircraft Operating Instructions (AOI).

3.1.4 *wing weight,* W_{wing}, *n*—all parts, components, and assemblies that comprise the wing assembly, or that are attached to the wing in flight, shall be included in the wing weight, W_{wing}. The wing weight, W_{wing}, shall be entered in the AOI.

3.2 *Abbreviations:*

3.2.1 *AOI*—Aircraft Operating Instructions

3.2.2 *C*—Celsius

3.2.3 *CAS*—calibrated air speed (m/s, kts)

3.2.4 *cm*—centimetre

3.2.5 *daN*—deca Newton

3.2.6 *F*—Fahrenheit

3.2.7 *Hg*—mercury

3.2.8 *IAS*—indicated air speed (m/s, kts)

3.2.9 *in.*—inch

3.2.10 *ISA*—international standard atmosphere

3.2.11 *kg*—kilogram

3.2.12 *kt(s)*—nautical mile per hour (knot) (1 nautical mph = (1852⁄3600) m/s)

3.2.13 *lb*—pound (1 lb = 0.4539 kg)

3.2.14 *m*—metre

3.2.15 *mb*—millibars

3.2.16 *N*—Newton

3.2.17 *psi*—pounds per square inch gage pressure

3.2.18 *s*—seconds

3.2.19 *SI*—international system of units

3.2.20 V_A—design maneuvering speed

3.2.21 V_C—design cruising speed

3.2.22 V_{DF}—demonstrated flight diving speed

3.2.23 V_H—maximum sustainable speed in straight and level flight

3.2.24 V_{NE}—never exceed speed

3.2.25 V_{S0}—stalling speed or minimum steady flight speed at which the aircraft is controllable in the landing configuration

3.2.26 V_{S1}—stalling speed, or the minimum steady flight speed in a specific configuration

3.2.27 V_x—speed for best angle of climb

3.2.28 V_y—speed for best rate of climb

3.2.29 V_T—maximum aerotow speed

3.2.30 W_{MAX}—maximum design weight

3.2.31 *WSC*—weight shift control (aircraft)

4. Flight Requirements

4.1 *Proof of Compliance:*

4.1.1 It shall be possible to demonstrate that the aircraft meets the requirements in this section at each allowable combination of weight, hang point, and trimmer setting.

4.1.2 The test aircraft used to demonstrate compliance with this specification shall be an accurate representation of the production aircraft except in the following case:

4.1.2.1 For the purposes of this test only, the aircraft may be modified to expand the control travel or limits in pitch when establishing V_{DF} or V_{S1}.

4.1.3 Airspeeds shall be corrected to standard atmospheric conditions 1013.25 mb [29.92 in. Hg], 15°C [59°F].

4.1.4 Climb performance requirements shall be met at standard conditions or conditions more adverse.

4.2 *General Performance:*

4.2.1 *Stall Speed in the Landing Configuration (*V_{S0}*):*

4.2.1.1 The stall speed, if obtainable, or the minimum flight speed shall be established with: (*1*) engine idling with the throttle closed, (*2*) hang point that produces the highest stalling or minimum flight speed, (*3*) maximum takeoff weight, and (*4*) trim setting in the landing configuration.

4.2.1.2 V_{S0} shall be determined by flight-testing, in accordance with the following procedures: (*1*) aircraft power at idle, at a speed of not less than V_{S0} plus 2.6 m/s [5 kts], and (*2*) the speed reduced at a rate not exceeding 0.5 m/s [1 kt/s] until the stall is produced as indicated by an autonomous downward pitching motion of the wing or until the control limit is reached.

4.2.1.3 It shall be possible to prevent more than 30° of roll or yaw by normal use of the controls during the stall and the recovery, or, if stall is not achieved before the control limit is reached, during the slowing to V_{S0} and subsequent acceleration to V_{S0} plus 2.6 m/s [5 kts].

4.2.2 *Stall Speed Free of Control Limits (*V_{S1}*):*

4.2.2.1 Where control limits result in V_{S0} being reached before the aircraft stalling, then the stall speed free of control limits (V_{S1}) shall be determined. V_{S1} shall be established with: (*1*) the aircraft in the landing configuration defined in 4.2.1.1, and (*2*) the aircraft may be modified for the purposes of this test, only to expand the nose up pitch control range to the extent necessary for the aircraft to stall when flown in accordance with the procedures detailed in 4.2.1.2.

4.2.2.2 Where V_{S0} as determined in accordance with the procedures of 4.2.1.2 is the speed at which the aircraft stalls, then $V_{S1} = V_{S0}$.

4.2.3 *Minimum Climb Performance:*

4.2.3.1 The gradient of climb at recommended takeoff power at *Vx* shall not be less than 1:12.

4.2.3.2 The rate of climb shall exceed 1.5 m/s [300 ft/min] at *Vy* at recommended takeoff power.

4.2.4 *Flutter, Buffeting, and Vibration*—Flight-testing shall not reveal, by pilot observations, potentially damaging buffeting, airframe, or controls vibration, flutter (with attempts to induce it), or control divergence, at any speed from V_{S0} to V_{DF}.

4.2.5 *Turning Flight and Stalls*—Stalls shall be performed as follows: after establishing a steady state turn of at least 30°

bank, the speed shall be reduced until the aircraft stalls, or until the full nose up limit of pitch control is reached. After the turning stall or reaching the limit of pitch control, level flight shall be regained without exceeding 60° of roll. This shall be performed with the engine at idle. No loss of altitude greater than 152 m [500 ft], uncontrolled turn of more than one revolution, or speed buildup to greater than V_{NE} shall be associated with the recovery.

4.2.6 V_H—Maximum sustainable speed in straight and level flight, knots CAS.

4.2.6.1 V_H shall be established in straight and level flight with: (*1*) maximum allowed continuous engine power, and (*2*) the combination of weight, loading, trimmer setting, and use of the flight controls allowed by the manufacturer that yields the highest sustainable speed.

NOTE 2—In the case where maximum continuous engine power results in a climb at maximum speed, power may be reduced as needed to maintain level flight.

4.3 *Controllability and Maneuverability:*

4.3.1 *General*—When operating in accordance with the recommendations in the Aircraft Operating Instructions, the aircraft shall be safely controllable and maneuverable during:

4.3.1.1 Takeoff at maximum takeoff power,

4.3.1.2 Climb,

4.3.1.3 Level flight,

4.3.1.4 Descent,

4.3.1.5 Landing, power on and off,

4.3.1.6 With sudden engine failure,

4.3.1.7 Turns,

4.3.1.8 Changing speeds between V_{S0} and V_{NE}, and

4.3.1.9 Dive to V_{NE}.

4.3.2 *Longitudinal Control:*

4.3.2.1 Starting at a speed of 1.1 V_{S0}, it shall be possible to pitch the nose downwards so that a speed equal to 1.3 V_{S0} can be reached in less than 4 s.

4.3.2.2 It shall be possible to pitch the nose up at V_{NE} at the most adverse hang point, trimmer setting, and engine power.

4.3.3 *Lateral Control:*

4.3.3.1 Using an appropriate control action, it shall be possible to reverse a steady 30° banked turn to a 30° banked turn in the opposite direction. This shall be possible in both directions within 5 s from initiation of roll reversal, with the aircraft flown at 1.3 V_{S0}.

4.3.3.2 Lateral control forces shall not reverse with increased displacement of the flight controls.

4.3.4 *Trim Speeds*—The speeds to achieve longitudinal trim shall lie between 1.3 V_{S0} and 0.909 V_{NE} at all engine powers and the allowable hang points.

4.3.5 *Ground Handling*—It shall be possible to prevent ground looping, with normal use of controls, up the maximum crosswind component published in the AOI.

4.4 *Stability:*

4.4.1 *Longitudinal Stability:*

4.4.1.1 The aircraft shall demonstrate the ability to sustain steady flight at speeds appropriate for climb, cruise, and landing.

4.4.1.2 A pull force shall be required to attain and maintain any speed above trim and a push force shall be required to attain and maintain any speed below trim. As the control force is reduced, the aircraft shall return to within 20 % the original trim speed.

4.4.2 *Pitch Testing*—A test of the wing pitching moment about the hang point shall be conducted at $V_{S0} \times 0.866$ over the range of angles of attack from 15° above zero lift angle to 10° below zero lift angle of attack. The wing shall exhibit a trim angle above zero lift angle of attack, and a positive pitching moment at any angle below trim, or if trim is not achieved in the test range, the wing shall exhibit a positive pitching moment throughout the range of angles specified.

NOTE 3—This test may be conducted as a taxi test with the wing mounted to the trike carriage.

5. Structural Requirements

5.1 *Strength Requirements:*

5.1.1 Strength requirements are specified in terms of limit loads (the maximum loads to be expected in service) and ultimate loads (limit loads multiplied by prescribed factors of safety as specified in 5.3). Unless otherwise provided, prescribed loads are limit loads.

5.1.2 The structure shall be able to support limit loads without permanent deformation. At any load up to limit loads, the deformation may not interfere with safe operation.

5.1.2.1 The structure shall be able to support ultimate loads with a positive margin of safety (analysis) or without failure for at least 3 s (tests).

5.2 *Fulfillment of Design Requirements:*

5.2.1 Fulfillment of the design requirements shall be determined by conservative analysis, tests, or a combination of both. Structural analysis alone may be used for validation of the structural requirements only if the structure conforms to those for which experience has shown this method to be reliable. Aerodynamic data required for the establishment of the loading conditions shall be verified by tests, calculations, or conservative estimation.

5.2.1.1 For analysis and test purposes, unless otherwise provided, the air and ground loads shall be placed in equilibrium with inertia forces, considering each major item of mass in the aircraft. The loads shall be distributed so as to represent actual conditions or a conservative approximation to them.

5.2.2 If deflections under load would significantly change the distribution or amount of external or internal loads, this redistribution shall be taken into account.

5.2.3 The results obtained from strength tests should be corrected for departures from the minimal mechanical material properties and least favorable material dimensional tolerance values defined in the design.

5.3 *Safety Factors*—The factor of safety is 1.5, except it shall be increased to:

3	on castings and bearings whose failure would preclude continued safe flight and landing of the aircraft or result in serious injury to the occupants
2	on other castings and bearings
2	on cables
2	on lap belts and shoulder harnesses
1.73	on fittings and system joints whose strength is not proven by limit and ultimate tests in which actual stress conditions apply or are simulated.

5.4 *Design Airspeeds:*

5.4.1 The selected design air speeds are calibrated air speeds (CAS):

5.4.1.1 *Maneuvering Speed V_A—V_A* shall be greater than or equal to $V_{S1} \times 2$.

5.4.2 V_{NE} shall be no greater than $0.9 \times V_{DF}$.

5.4.3 V_{DF} shall be greater than or equal to the lesser of $1.11 \times V_A$ or $1.11 \times V_{DMAX}$.

5.5 *Flight Loads:*

5.5.1 Except in the case of dynamic testing, as detailed in the applicable sections of this specification, the limit load factors must have at least the following values:

+4.0
−2.0

5.5.1.1 If V_A is greater than two times V_{S1}, then the minimum positive limit load factor shall equal $(V_A/V_{S1})^2$. The negative load limit factor shall not be required to be greater than −2.0.

5.5.2 Although it is very difficult and very unlikely to achieve sustained negative flight loads on weight shift aircraft, wings shall be tested for such loads to ensure adequate strength to withstand negative loads caused by gusts, landing, and ground handling.

5.5.3 Adequate structure of the wing to ultimate loads as prescribed by the 1.5 safety factor shall be verified via test (static, component, dynamic, or flight).

5.5.3.1 Compliance with special factors above the safety factor of 1.5 may be shown by testing or conservative analysis, or both.

5.5.3.2 For a conventional flex-wing configuration, for the purposes of calculating the positive and negative limit and ultimate load values for test purposes, unless a specific testing protocol listed in this specification or its appendices is used that specifies another method for allocating the weight of the wing, it shall be considered appropriate to include in the weight of the aircraft 50 % of the weight of all components comprising the wing assembly.

5.5.4 For static testing of the wing, in the absence of a more rational analysis, the test shall be conducted in accordance with one of the test protocols as contained in Appendix X1.

5.5.5 Compliance with the positive limit load requirements for the wing may alternatively be shown by a dynamic test of the wing in which the wing is tested at an angle of attack equal to the highest angle at which maximum lift is achieved, at an airspeed equal to the greater of $1.0 \times V_A$ (maneuvering speed), or the speed that will produce a measured load of 3.8 Gs, for a minimum of 3 s without permanent deformation of the structure.

5.5.6 Compliance with the positive ultimate load requirements for the wing may alternatively be shown by a dynamic test of the wing in which the wing is tested at an angle of attack equal to the highest angle at which maximum lift is achieved, at an airspeed equal to the greater of $1.225 \times V_A$ (maneuvering speed), or the speed which produces 1.5 times the load achieved in the limit load test, for a minimum of 3 s without failure.

5.5.7 Compliance with the negative limit load requirements for the wing may alternatively be shown by a dynamic test of the wing in which the wing is tested at a negative angle of attack equal to the highest negative angle at which maximum negative lift is achieved, at an airspeed equal to the greater of $0.707 \times V_A$, or the speed which produces a measured negative load of 1.52 Gs, for a minimum of 3 s without permanent deformation of the structure.

5.5.8 Compliance with the negative ultimate load requirements for the wing may alternatively be shown by a dynamic test of the wing in which the wing is tested at a negative angle of attack equal to the highest negative angle at which maximum negative lift is achieved, equal to the greater of $0.866 \times V_A$ (maneuvering speed), or the speed which produces 1.5 times the load achieved in the limit load test, for a minimum of 3 s without failure.

5.5.9 If dynamic testing is chosen for limit load testing of the wing, compliance with the ultimate load requirements may be shown by conducting a static load test to a load of 1.5 times the loads generated during dynamic limit tests. The wing shall sustain this load for a minimum of 3 s without failure but may show permanent deformation.

5.6 *Pilot Control Loads:*

5.6.1 The pitch and roll control bar shall be designed to withstand as far as to the stops (these included) limit loads arising from the pilot forces in Table 1. Lower pilot forces may be established, provided it can be demonstrated that the forces in Table 1 are unlikely to be able to be applied.

5.6.1.1 Where a backup safety system ensures the ability to continue safe flight in the event of a control system component failure, the forces in Table 1 may be reduced by ⅓ .

5.6.1.2 In roll, in the case in which the rear lower rigging wires bearing against the operators or trike fuselage is the only practicable roll control limit stop preventing damage to the structure, the control frame upright shall be able to achieve an angle within 10º of the vertical centerline of the trike without causing permanent structural deformation. If the upright can reach this angle, it is not necessary to show compliance with Table 1 for control stop purposes.

5.6.2 Dual control systems must be designed to withstand the loads that result when each pilot applies 0.75 times the load specified in Table 1 with the pilots acting together in the same direction, and the pilots acting in opposition.

5.7 *Ground Loads:*

5.7.1 The fuselage shall be able to sustain a static limit load of 2*g* without permanent deformation. The loads shall be distributed throughout the structure in a rational manner,

TABLE 1 Pilot Forces

Control	Pilot Force, daN [lb force]	Method of Force Application
Pitch	66.7 [150]	Push or pull of control bar
Roll	31.1 [70]	Lateral force (roll) applied to the control bar
Foot controls for steering	89 [200]	Apply forward pressure on one pedal
Foot controls for throttle and brake	44.5 [100]	Push of control
Miscellaneous secondary controls	22.2 [50]	Push and pull of control lever

including wing load, engine load, full fuel load, occupant load, frame load, and maximum allowable baggage load.

5.7.2 An ultimate load of 2 g × 1.5 safety factor (3 g) shall be supported for a minimum of 3 s without failure.

5.7.3 *Landing Gear Shock Absorption*—The landing gear shall be capable of absorbing the energy that would result when landing with the specified vertical velocity without either the shock absorber or the tires bottoming.

5.7.3.1 The specified vertical velocity or drop height or both are calculated as follows:

where:

W_{MAX} = design maximum weight specified in kg, and
S = wing area specified in m^2.

then:

specified vertical velocity = $0.9 \cdot (W_{MAX}/S)^{1/4}$ m/s, and
specified drop height = $4.1 \cdot (W_{MAX}/S)^{1/2}$ cm.

where:

W_{MAX} = lb, and
S = ft^2.

then:

specified vertical velocity = $4.4 \cdot (W_{MAX}/S)^{1/4}$ ft/s, and
specified drop height = $3.6 \cdot (W_{MAX}/S)^{1/2}$ in.

5.7.3.2 This may be demonstrated by way of a single drop test from the specified height. No corresponding ultimate test is required. This test shall be performed using a trike carriage loaded to maximum design weight with a normal load distribution and hanging such that the front wheel is 10 ± 2 cm [3.94 ± 0.79 in.] higher than the rear wheels. The drop height is measured from base of the rear wheels to ground.

5.8 *Emergency Landing Loads*—In an emergency landing in which each occupant experiences, separately, the following ultimate inertia forces:

5.8.1 *Upward*—3.0 g.

5.8.2 *Forward*—6.0 g.

5.8.3 *Sideward*—1.5 g.

5.8.4 *Downward*—4.5 g.

5.8.5 Within the constraints imposed by the limitations inherent in an aircraft without an enclosing cockpit, the aircraft shall be designed such that, although it may be damaged:

5.8.5.1 It will restrain the occupants (arms and legs excluded) within the aircraft when proper use is made of safety equipment as prescribed in the AOI, including but not limited to belts and harnesses provided for in the design, and

5.8.5.2 The aircraft shall not undergo permanent deformation to an extent that the aircraft structure would likely cause serious injury to the occupants.

5.8.6 The supporting structure shall be designed to restrain, under loads up to those specified in 5.8.1 – 5.8.4, each item of mass that could injure an occupant if it came loose in a minor crash landing.

5.8.7 For an aircraft with the engine components or fuel tank located behind and above an occupant seat, an ultimate inertia load of 15 g in the forward direction must be assumed for these components.

5.8.8 Fuel tank mounting points shall be capable of sustaining the inertia forces specified in 5.8.1 – 5.8.4 or 5.8.7 as applicable, without failure of the mounts or rupture of the tank.

6. Design and Construction Requirements

6.1 The structure shall be designed, as far as practicable, to avoid points of stress concentration and high stresses and to take account of the effects of vibration. Materials that are inherently unsuited to an application shall be avoided.

6.2 *General*—The integrity of any unusual design feature having an important bearing on safety shall be established by test.

6.3 *Materials*—Materials shall be suitable and durable for the intended use. Design values (strength) shall be chosen so that no structural part is under strength as a result of material variations or load concentration, or both.

6.4 *Fabrication Methods*—Manufactured parts, assemblies, and completed aircraft shall be produced in accordance with the manufacturer's quality assurance and production acceptance test procedures.

6.5 *Self-Locking Nuts*—No self-locking nut shall be used on any bolt subject to differential angular motion between the contact surface on the bolt and the contact surface on the nut during taxi, takeoff, flight, and landing, unless a non-friction locking device is used in addition to the self-locking device.

6.6 *Protection of Structure*—Protection of the structure against weathering, corrosion, and wear, as well as suitable ventilation and drainage, appropriate to operation and maintenance in accordance with the recommended procedures as stated in the AOI, shall be provided.

6.7 *Accessibility*—Accessibility for critical structural elements and control system inspection, adjustment, maintenance, and repair shall be provided.

6.8 *Setup and Breakdown*—Instructions for setup, breakdown, and preflight inspection provided in the AOI shall be sufficiently detailed for a trained pilot to be able to fulfill these actions competently.

6.9 *Control System—Operation Test*—It shall be shown by functional test that the control system installed on the aircraft is free from interference, jamming, excessive friction, and excessive deflection when the control system design loads are applied to the control frame. The control frame stops shall withstand those loads.

6.10 *A Mast (Pylon) Safety Device,* with minimum ultimate strength of 3× WSUSP, shall be provided that connects the wing to the fuselage below the mast in the event of a mast failure.

6.11 *Cockpit Design*—The cockpit and its equipment shall allow each pilot to perform his duties without unreasonable concentration or fatigue. In the design of the cockpit, consideration shall be taken to avoid, where practical, the use of sharp objects that would likely cause serious injury to an occupant in the emergency landing conditions specified in 5.8.

6.11.1 *Controls:*

6.11.1.1 Each cockpit control shall be located and arranged so that the pilot, when strapped in, has full and unrestricted movement of each control.

6.11.1.2 In aircraft with dual controls, it shall be possible to operate the throttle and the ignition kill switch from each pilot's seat.

6.11.2 *Occupant Restraint:*

6.11.2.1 The design of the restraint system shall allow a full range of pilot movement such as is required to control the aircraft under all conditions likely to be encountered in service.

6.11.2.2 A lap belt shall be available to each occupant, capable of restraining the wearer against the inertial forces prescribed for emergency landing conditions specified in 5.8.1 – 5.8.4.

6.11.2.3 Each seat and its supporting structure shall be designed for a maximum occupant weight of 90 kg [198.4 lb] and the maximum load factors corresponding to the specified flight and ground conditions including the emergency landing conditions prescribed in 5.8.1 – 5.8.4.

6.12 *Markings and Placards:*

6.12.1 The aircraft shall be marked with the following placard:

The Aircraft Operating Instructions must be carried with the aircraft. Occupants must be familiar with information necessary for safe operation.

6.12.2 Each marking and placard shall be displayed in a conspicuous place, and may not be easily erased, disfigured, or obscured.

7. Powerplant

7.1 *Installation*—The powerplant installation shall be easily accessible for inspection and maintenance. The powerplant attachment to the airframe is part of the structure and shall withstand the applicable load factors.

7.2 *Fuel System:*

7.2.1 The unusable fuel quantity for each tank shall be established by tests and shall not be less than the quantity at which the first evidence of engine fuel starvation occurs under each intended flight operation and maneuver.

7.2.2 The fuel tanks shall be protected against wear from vibrations and their installation shall be able to withstand the applicable inertia loads.

7.2.3 Fuel tanks shall be designed to withstand a positive pressure of 241.3 mb [3.5 psi].

7.2.4 The fuel filler shall be located outside of the passenger compartment. Spilled fuel shall be prevented from entering or accumulating in any enclosed part of the aircraft.

7.2.5 Each fuel tank shall be vented. The vent shall not siphon in flight and must discharge clear of the engine and exhaust system.

7.2.6 There shall be a means to remove water and debris from the fuel system.

7.2.7 A fuel filter accessible for drainage and cleaning or replacement shall be included in the system.

7.2.8 Fuel lines shall be supported and protected from vibrations and wear.

7.2.9 Fuel lines located in any area subject to high heat shall be fire resistant or protected with a fire-resistant covering.

7.2.10 There shall be a control accessible to the pilot while wearing a seat belt by which the pilot can effectively shut off the flow of fuel.

7.3 *Oil Systems:*

7.3.1 If an engine is provided with an oil system, it shall be capable of supplying the engine with an adequate quantity of oil at a temperature not exceeding the maximum established by the engine manufacturer.

7.3.2 The oil tank or radiator, or both, shall be installed to withstand the applicable inertia loads and vibrations.

7.3.3 The oil breather (vent) shall be resistant to blockage caused by icing.

7.3.4 Oil foam from the breather shall not constitute a hazard.

7.4 *Induction System*—The engine air induction system shall be designed to minimize the potential of carburetor icing.

7.5 *Fire Prevention*—If the engine is fully enclosed, it shall be isolated from the rest of the aircraft by a firewall or shroud. It shall be constructed as far as practical to prevent liquid, gas, or flames or all from entering the aircraft. The use of any one of the following materials shall be acceptable without further testing:

7.5.1 Stainless steel not less than 0.038 cm [0.015 in.] thick,

7.5.2 Mild steel not less than 0.046 cm [0.018 in.] thick, or

7.5.3 Alternative materials that are shown to provide protection equivalent to 7.5.1 or 7.5.2.

7.6 *Powerplant Requirements:*

7.6.1 Engine, transmission, and propeller for aircraft covered under this specification shall meet at least one of the following standards: Practice F2506, Practice F2339, JAR-22 parts H and J, JAR-E, JAR-P, FAR-33, or FAR-35.

7.6.2 Alternatively, if an aircraft has a wing loading no greater than 25 kg/m^2 [5.12 lb/ft^2] or a stall speed (V_{SO}) no greater than 18 m/s [35 kts], a manufacturer may elect to use a powerplant meeting the requirements of 7.6.3 in place of 7.6.1.

7.6.3 Powerplant suitability shall be demonstrated by performing, on at least one aircraft, engine, and propeller, a minimum of 100 flight hours, including 200 takeoffs and landings without failure. The test shall be completed having performed only those maintenance operations listed in the POH for normal service.

8. Equipment Requirements

8.1 *Powerplant Instruments:*

8.1.1 Fuel indicator or means to view fuel quantity from the pilot seat.

8.1.2 Engine instruments as required by the engine manufacturer.

8.2 *Miscellaneous Equipment:*

8.2.1 If installed, an electrical system shall include a master switch and overload protection devices.

8.2.2 The electric wiring shall be sized according to the load of each circuit.

8.2.3 The battery installation shall withstand all applicable inertia loads.

8.2.4 Unless sealed batteries are used, battery containers shall be vented outside of the aircraft.

8.3 *Lap Belts and Harnesses*—Occupant lap belt, harness, and their attachments to the structure shall be designed for the appropriate loads.

8.4 An airspeed indicator shall be provided to enable the pilot to comply with limiting airspeeds, unless V_H is less than V_A and less than V_{NE}.

9. Operating Limitations

9.1 *Load Distribution Limits*—The manufacturer shall select the ranges of weight and hang point within which the aircraft is to be safely operated. This information shall be recorded in the AOI.

9.2 The operating limitations and other information necessary for safe operation shall be made available to the pilot in the AOI.

9.3 All flight speeds shall be stated in terms of indicated air speed readings (IAS). Speeds (CAS) determined from structural limitations should be suitably converted.

10. Keywords

10.1 hang point; trimmer; weight-shift-control aircraft

ANNEX

(Mandatory Information)

A1. DESIGN AND PERFORMANCE STANDARDS FOR LIGHT SPORT AIRCRAFT USED TO AERO-TOW GLIDERS

A1.1 *Applicability*—This annex is applicable to weight-shift-controlled light sport aircraft that are to be used for towing of gliders.

A1.1.1 *Minimum Climb Performance While Towing:*

A1.1.1.1 The aircraft shall be capable of achieving a gradient of climb while towing of at least $1/18$, while not exceeding the maximum placarded towing speed of the towing aircraft or the maximum safe towing speed of the aircraft being towed.

A1.1.1.2 The aircraft shall be capable of achieving a rate of climb while towing of at least 0.762 m/s [150 ft/min], while not exceeding the maximum placarded towing speed of the towing aircraft or the maximum safe towing speed of the aircraft being towed.

NOTE A1.1—Compliance with A1.1.1 shall adequately take into account the performance and control capabilities of both the towing aircraft and the aircraft being towed. To account for varying performance and control capabilities on the part of the towed aircraft, the manufacturer of the towing aircraft may specify a maximum weight and maximum drag for the towed aircraft at each speed for which the towing aircraft is approved for tow operations, such that the required climb performances can be achieved. Compliance with A1.1.1 is then shown when the towed aircraft is safely controllable under tow at a speed for which its drag and weight are within these prescribed maximum weight and drag limits.

A1.2 *Controllability and Maneuverability*—The aircraft shall be safely controllable and maneuverable during all ground and flight operations applicable to normal towing operations, including both deliberate and inadvertent release of the glider being towed.

A1.3 *Stability*—It shall be possible to conduct normal towing operations, including both deliberate and inadvertent release of the glider being towed, without incurring any dangerous reduction in the stability of the aircraft.

A1.4 *Structure and Strength Requirements*—The otherwise applicable structure and strength requirements for the aircraft shall be met, taking into account the effects of loads arising from towing equipment that is included in the design of the aircraft or installed on the aircraft. See A1.5.1.

A1.5 *Design and Construction*:

A1.5.1 *Glider Towing Installations*—The maximum all up takeoff weight of the glider to be aero-towed, including pilot and all equipment, shall be selected by the manufacturer.

A1.5.1.1 The maximum glider towing speed, V_T, shall be selected by the manufacturer. V_T shall be at least 1.3 V_S, where V_S is the stalling speed of the aircraft in the cruising configuration without a glider in tow.

A1.5.1.2 The aircraft shall have limit and ultimate factors of safety not less than 1.0 and 1.5 respectively, when loads equal to 1.2 of the nominal strength of the weak link (see A1.5.1.6) are applied through the towing hook installation in the conditions shown below, simultaneously with the loads arising from the most critical normal accelerations (as defined in the normally applicable requirements for structure and strength) at the speed V_T.

A1.5.1.3 The conditions applicable are: the speed is assumed initially to be at the maximum glider towing speed V_T, and the load at the towing hook installation is assumed to be acting in each of the following directions relative to the longitudinal centerline of the aircraft: horizontally backwards, backwards and upwards at 40° to the horizontal, backwards and downwards at 20° to the horizontal, and horizontally backwards and 25° sideways in both directions.

A1.5.1.4 The towing hook shall be of a quick-release type. It shall be established that with loads equal to 10 and 180 % of the nominal strength of the weak link (see A1.5.1.6) is applied to the towing hook in each direction prescribed in A1.5.1.4*(2)* and the release control is operated:

(1) The tow cable will be released,

(2) The released cable will be unlikely to cause damage to or become entangled with any part of the aircraft, and

(3) The pilot effort required shall not be less than 20 N [4.5 lbf] or greater than 100 N [22.5 lbf].

A1.5.1.5 The release control shall be so located that the pilot can operate it without having to release any of the primary controls.

A1.5.1.6 The maximum strength of any weak link that may be interposed in the towing cable shall be established. For the determination of loads to be applied for the purpose of A1.5, the strength of the weak link shall not be less than 900 N [202.3 lb].

A1.6 *Operating Limitations*—Operating limitations applicable to towing operations must be established, and included in the POH, to include at a minimum:

A1.6.1 The maximum permissible towing speed (V_T),

A1.6.2 The maximum weak link strength (may be specified in terms of the weight of the glider to be towed), and

A1.6.3 The maximum permissible all up weight of the glider to be towed.

APPENDIX

(Nonmandatory Information)

X1. STATIC TEST PROTOCOL WING STRENGTH TESTS

X1.1 *Load Tests Required*—A series of four load tests are to be performed. There are no requirements as to the order in which these tests are performed.

X1.1.1 *Limit Positive Load Test*—After which the test airframe is inspected to ensure no permanent deformation (bending of battens due to the placement of test loads is not considered permanent deformation).

X1.1.2 *Ultimate Positive Load Test*—The test airframe, including the sail, shall withstand the test loading conditions for at least 3 s.

X1.1.3 *Limit Negative Load Test*—After which the test airframe is inspected to ensure no permanent deformation (bending of battens due to the placement of test loads is not considered permanent deformation).

X1.1.4 *Ultimate Negative Load Test*—The test airframe, including the sail, shall withstand the test loading conditions for at least 3 s.

X1.2 *Calculation Of Total Test Loads To Be Applied*:

Test	Required Load Factor, LF	Required Safety of Factor, n	Design Maximum Trike Carriage Weight, W_{susp}	Test Load to be Applied = LF × n × W_{susp}
Limit positive load	4.0	1.0		
Ultimate positive load	4.0	1.5		
Limit negative load	-2.0	1.0		
Ultimate negative load	-2.0	1.5		

X1.3 *Positive Load*—The load distribution in a spanwise direction is calculated outwards from the center of the wing using a triangular shape (see Fig. X1.1).

X1.3.1 *Calculation of Spanwise Loading*—To satisfy the requirements, the two halves (span widths) of the wing are divided into five equal fields. Table X1.1 shall be completed.

X1.3.2 The load is to be distributed in a chordwise direction in such a way that the maximum load is on the T/4 line. This will result in a loaded wing having a 20° positive angle at the root. If this is not the case, when the loaded wing is suspended from the trike carriage attachment bracket, then the load shall be adjusted, in a chordwise direction, until the required angle is achieved.

X1.4 *Negative Load*—The load distribution in a spanwise direction is calculated in proportion to the chord (see Fig. X1.2).

X1.4.1 *Calculation of Spanwise Loading*—To satisfy the requirements, the two halves of the wing are divided into equal fields. Complete Table X1.2.

X1.4.2 The load is to be distributed chordwise in such a way that the maximum load is on the T/4 line. This results in the wing (when under load) having a negative angle of 20° at the root. Because of the load distribution, there will be a tendency for the wing to go pitch-up. This shall be compensated for during the load testing by holding the keel (root) at the required angle.

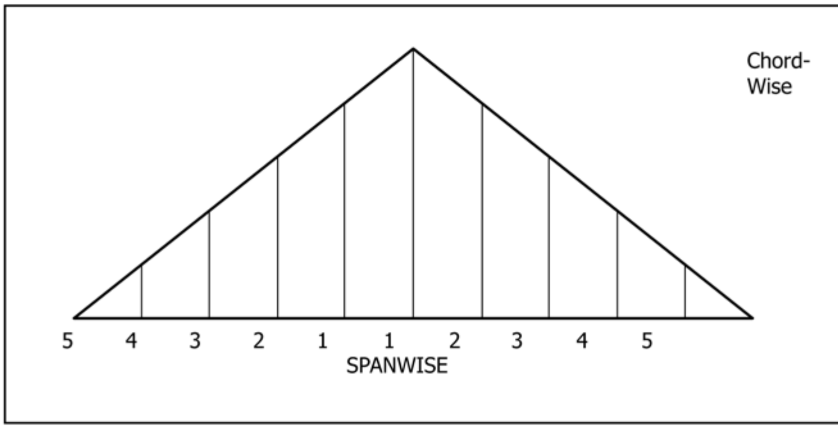

FIG. X1.1 Positive Load Distribution

TABLE X1.1 Positive Load Testing–Load Distribution Per Side

Test load to be applied/ 30: _______ / 30 = _______ = Xp				
Calculate the test load and record in column 3 below.				
Convert into number of sacks (column 4). Where the weight of 25 kg [55.1 lb] test sacks is significantly different to the required test sacks, then smaller 10 kg [22 lb] or 5 kg [11 lb] sacks may be used. Record these details in column 5. Check that the total applied load on each half of the wing is equal to Test Load to be applied / 2.				
Field	Formula	Test Load	25 kg [55.1 lb] Sacks	Extra Weight
1	5 × Xp			
2	4 × Xp			
3	3 × Xp			
4	2 × Xp			
5	1 × Xp			
Sum = _______ = Test load to be applied/ 2				

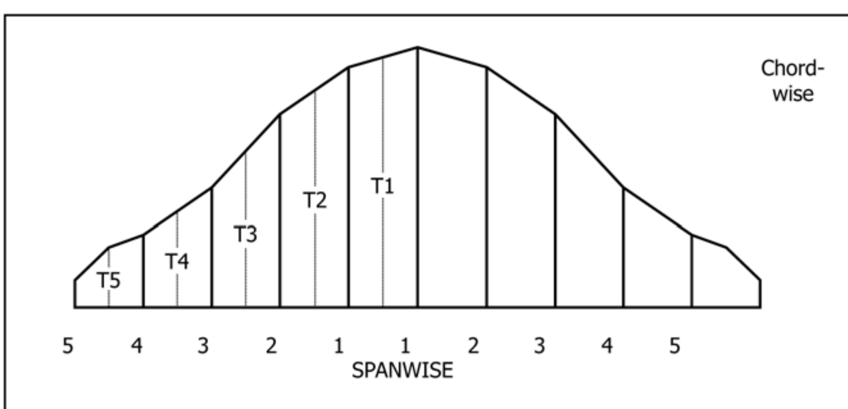

FIG. X1.2 Negative Load Distribution

TABLE X1.2 Negative Load Testing–Load Distribution Per Side

Measure the chord (depth of profile) in the middle of each field (see Fig. X1.2) and record in column 2 below.				
Calculate the load on one half of wing = test load to be applied / 2 = M_{ges} =				
Calculate the test load and record in column 3 below.				
Convert into number of sacks (column 4). Where the weight of 25 kg [55.1 lb] test sacks is significantly different to the required test sacks, then smaller 10 kg [22 lb] or 5 kg [11 lb] sacks may be used. Record these details in column 5. Check that the total applied load on each half of the wing is equal to M_{ges}.				
Field	Chord, Ti	Test Load = $(M_{ges}/T_{ges}) \times Ti$	25 kg [55.1 lb] Sacks	Extra Weight
1				
2				
3				
4				
5				
Add all chord lengths Sum of chords (Ti) = T_{ges}		Total weight on each half of the wing		