A method for predicting the rate and effect of approach to the stall of a microlight aeroplane.

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The stall and immediately post-stall behaviour of a microlight aeroplane are shown to be a function of the deceleration rate prior to the stall; therefore, it is necessary to use a representative deceleration rate when determining the acceptability of stall and post-stall handling qualities. This research has found means by which the range of deceleration rates likely to be seen in a particular type can be estimated, so that flight test programmes can ensure these rates are included, and thus aircraft are confirmed to have acceptable stalling characteristics. Recommendations are made towards the use of this research for all aircraft type, and of further work which might usefully be carried out.

Nomenclature

α	Wing angle of attack
ρ	Air density

σ	Relative air density
$\frac{dV_{ind}}{dt}$	Rate of change of Indicated Air Speed (IAS) with respect to time.
τ_d	Deceleration time factor
A	Arbitrary value used in calculation, no physical significance.
BCAR	British Civil Airworthiness Requirements
CAS	Calibrated Air Speed
C_D	Drag coefficient of aircraft
C_{Di}	Induced drag coefficient
C_{Do}	Zero lift (or profile) drag coefficient of aircraft
C_{Ds}	Drag coefficient of aircraft at point of stall
CG	Centre of Gravity (Centre of Mass)
C_L	Lift Coefficient of aircraft
$C_{L.max}$	Maximum (stall point) lift coefficient of aircraft
C_{LE}	Lift coefficient at the best range glide condition
C_M	Pitching Moment Coefficient of aircraft
ETPS	Empire Test Pilots School (based at Boscombe Down, Wiltshire, UK)
g	Acceleration due to gravity (9.80665 N/kg, or m/s²)
G	Best glide ratio
ht	Altitude
IAS	Indicated Air Speed
ISA	International Standard Atmosphere (also sometimes known as US Standard Atmosphere).
k	Gradient of C_{Di}/C_L^2 ($\frac{\partial C_D}{\partial (C_L^2)}$)
M	Mass
n	Normal acceleration

POH	Pilots Operating Handbook
R^2	Coefficient of determination, defining the quality of a line fit[1], has value $R^2=1$ for perfect line fit, $R^2=0$ for no measurable correlation.
S	Reference wing area (including a canard, if fitted, but not tailplane)
sHp	Standard Pressure Altitude (using 1013.25 hPa altimeter setting)
t	Time
TAS	True Air Speed
V	Aircraft translational velocity
V_E	Best range glide speed
V_S	Stalling speed
W/S	Wing loading (normally quoted for MTOW)

The fact and significance of stall entry rate.

The stall entry rate of any aircraft is critical in determining the stall and post-stall characteristics. This is because of the "deepness" of the stall, i.e. the minimum airspeed actually achieved before the aircraft starts to recover, and its being affected by the deceleration rate prior to the stall. This may be demonstrated by examining the stalling characteristics of an X'Air Mk.1 (see **Figure 1**) aircraft shown in Table 2 below.

[1] Defined by $R = \frac{n\sum xy - \sum x \sum y}{\sqrt{n\left(\sum x^2\right) - \left(\sum x^2\right)}\sqrt{n\left(\sum y^2\right) - \left(\sum y^2\right)}}$

Table 1, X'Air Mk.1 stalling characteristics

Stalling Characteristics, G-BYCL Type: X'Air 582(1), mid CG, MTOW, flight idle. Source of data: Type Certification flight test reports.		
Engine Power	Stall Entry Rate	Stall Characteristics
Flight idle (throttle closed)	1 kn/s	Ran out of control authority in level flight attitude
Flight idle (throttle closed)	2 kn/s	5° nose down pitch at the point of stall
Flight idle (throttle closed)	5 kn/s	20° nose down pitch at the point of stall

Figure 1, X'Air Mk.1 aircraft

In general, more rapid stall entries tend to cause greater nose-down pitching moments at the point of stall, whilst slower stall entries (typically the conventional 1 kn/s deceleration primarily used during certification testing) causes a reduced pitching moment, but in some circumstances (e.g. the Aviasud Mistral, which is a biplane with

an all-flying aileron-effect lower mainplane without strong centring, so that a gradual deceleration can often result in one wing stalling before the other) a greater tendency for the aircraft to suffer a wing-drop. During the flight test parts of this research, no general relationship between the stall entry rate and any tendency to enter a spin has been observed, but certain types of aeroplane (for example the Spectrum T1 as shown in Figure 2 below) will certainly enter an incipient spin mode from a rapid stall entry, whilst this does not occur following a more gradual deceleration.

Figure 2, Aviasud Mistral aircraft

Figure 3, Spectrum T1 aircraft

The definition of the stall and stall warning from the perspective of the pilot.

It is important to appreciate that the stall, as seen by the pilot, is not identical to the stall as would be understood classically by an aerodynamicist. The following definition, which is extracted from BCAR Section S**[1]**, is typical of the definitions contained in most civil certification standards:-

(From S201(a)) *Stall demonstrations must be conducted by reducing the speed by approximately 1kn/s from straight and level flight until either a stall results as evidenced by a downward pitching motion or downward pitching and rolling motion not immediately controllable or until the longitudinal control reaches the stop.*

A more simple definition, which is a variation upon that taught in the military test pilots schools such as the Empire Test Pilots School at Boscombe Down, Wiltshire (ETPS), is that *a stall is the point following deceleration at which the pilot ceases to*

have full control over the aeroplane. This is compatible with the definition above, since an uncontrolled motion or the longitudinal control being on the stop are clear indicators that the pilot does not have full control over the aircraft in all axes; however, wing rocking (undemanded rolling oscillations, initially of low amplitude but potentially enough to roll an aircraft inverted if not controlled), or other low-speed departures from controlled flight may also be included.

This definition is different to the stall as commonly explained in purely aerodynamic terms. Such conventional explanations (for example section 8.2. of **[2]**) would most normally either define the stall when considering lift versus α characteristics as the point at which lift ceases to increase with increasing α, by reference to a flow visualisation as the point where a given degree of flow detachment occurs from the lifting surface, or as the point at which there is a marked increase in the gradient of $\frac{\partial C_M}{\partial \alpha}$. However, whilst these features are essential to aerodynamic research, not all (or sometimes any) of these will be immediately apparent in those forms to a pilot and depending upon severity may not be considered by a pilot to mark the stall in any case.

During a test programme, the test team must define the stall for a specific aircraft. Notwithstanding that other definitions may be useful in certain circumstances, the three most common definitions are:-

- The longitudinal control being on the nose-up control stop (often termed "mush" by pilots). This is most common at forward CG / hangpoint states where insufficient nose-up control authority exists to fully aerodynamically stall the wing.

- A downward pitching motion (often termed a “pitch break”). This is caused by a loss of lift at the mainplane (or canard) altering the balance of forces and moments on the aircraft and causing a net nose-down pitching moment. This is most common at aft CG/hangpoint states, where there is sufficient nose-up control authority to fully aerodynamically stall the wing.
- A wing drop, sometimes accompanying a pitch break. This occurs where the two sides of the mainplane do not stall simultaneously and may be caused by a small amount of uncorrected sideslip, a rigging asymmetry in the wings and airframe, or by an inadvertent control input.

The term stall warning describes those characteristics of the aircraft which indicate to a pilot that he or she is flying at conditions close to the stall and caution may be needed. Stall warning characteristics will vary between aircraft and should normally be noted in the operators manual. The following are typical stall warnings:-

- Airframe buffet, as localised airflow starts to detach.
- Stick buffet, as localised airflow, usually over the wing root in a conventional 3-axis/tailplane aircraft, detaches and strikes the tail control surfaces.
- Artificial stall warning devices, normally either based upon an α sensor **[3]**or a localised airflow pressure sensor**[4], [5]**.
- An aircraft pitch attitude which is perceptibly more nose-up than that normally seen in level flight.
- The aircraft’s primary pitch control being noticeably displaced in the nose-up sense compared to its position in level flight.
- Lack of control responsiveness.

During the airworthiness evaluation process for any aircraft, the following questions need to be addressed:-

- What are the stalling characteristics at representative deceleration rates? Are these characteristics acceptable?
- What are the stall warning cues? Are they adequate?
- Is the aircraft fully controllable during deceleration down to the point of stall?
- Can the aircraft, post-stall, be returned to controlled flight without the use of exceptional piloting skill, or whilst suffering an unacceptable degree of height loss or uncommanded manoeuvre?

Finally, operating data (most particularly the Pilots Operating Handbook, or POH) must be confirmed to accurately and safely address the stalling characteristics of the aeroplane.

The significance and magnitude of the stall entry rate

Historical experience**[6]** is that in most light aircraft, the combination of inertia and drag are such that in the event of mishandling or sudden loss of power, the rate of deceleration can reasonably be expected to be around the 1kn/s used for the determination of stall speed (and acceptable handling characteristics at the point of stall) contained within most certification codes; it is also near-optimal for recognition of stall warning cues. However, for microlight aeroplanes, this is not necessarily true; the combination of low mass (not greater than 450g for a 2-seat landplane, or 300kg

for a single seat landplane) and relatively high drag (particularly caused by unfaired or externally braced structures) can result in far higher deceleration rates. The consequence of this is that the handling characteristics following a genuinely inadvertent stall, can differ significantly from those which would be found if testing was only carried out at 1kn/s deceleration.

Realising this, most accepted test schedules such as **[7]**,**[8]** circa 1999 were modified following unpublished work by the author to insist upon acceptable stalling characteristics at increased deceleration rates of up to 5kn/s. This value however was entirely empirical and the reason for this value has not historically been justified. To address this lack of rigour, the following investigation seeks to establish a means to estimate a deceleration rate, representative of what would occur in a mishandling or sudden loss of power case, which might be used during certification flight testing to determine whether stalling characteristics are acceptable.

Measurement and Estimation of the Stall Entry Rate

The following assumptions are made:

- In this class of aircraft, the pilot will initially either enter a descent or maintain level flight in the event of a sudden engine failure. (In high energy aircraft such as fighters the immediate action would be to climb to increase potential energy; however this behaviour is inappropriate and thus not taught in small light aeroplanes.)

- C_{Do} is constant between V_S and V_E

- The partial derivative of lift with respect to induced drag squared is constant between V_S and V_E
- The aircraft is moving within a constant velocity air mass (i.e. inertial effects due to movement of that air mass are insignificant).

Basic equations:

Basic lift equation $L = ½ \rho V^2 S C_L$ (1)

Basic drag equation $D = ½ \rho V^2 S C_D$ (2)

Components of Drag $C_D = C_{Do} + k C_L^2$ (3)

Note that the term k, the lift-dependent drag coefficient factor, above is treated here as a constant value for $\frac{\partial C_D}{\partial (C_L^2)}$ and its greater physical significance will not be discussed.

A detailed discussion of the significance of this constant may be found particularly in chapter XI of reference **[9]** and also repeated in more recent texts.

Now consider the aircraft at the stall:-

Drag at the point of stall

$$C_{Ds} = C_{Do} + kC_{L\max}^2 \quad (4)$$

[from (3)], assuming C_{Lmax} occurs at the stall.

Re-arranging (1):-

$$C_{L\max} = \frac{Mg}{\frac{1}{2}\rho V_S^{\,2} S} \quad (5)$$

Inserting (5) into (4):-

$$C_{Ds} = C_{Do} + k\left(\frac{Mg}{\frac{1}{2}\rho V_S^2 S}\right)^2 \quad (6)$$

Inserting (6) into (2):-

$$D_s = \frac{1}{2}\rho V_S^2 S\left(C_{Do} + k\left(\frac{Mg}{\frac{1}{2}\rho V_S^2 S}\right)^2\right) \quad (7)$$

Applying Newton's second law to (7)

$$\left(\frac{dV}{dt}\right)_S = -\left(\frac{\frac{1}{2}\rho V_S^2 S}{M}\right)\left(C_{D0} + k\left(\frac{Mg}{\frac{1}{2}\rho V_S^2 S}\right)^2\right) \quad (8)$$

Where the acceleration rate, $\frac{dV}{dt}$ should have a negative sign, indicating deceleration.

In order to solve equation (8) we only require C_{Do} and k, since all other parameters are known from either design or flight conditions. These missing terms will be found by use of the best range glide condition - since at this condition $C_{Do} = kC_L^2$ and the best

glide ratio, G, exists (For proof of these statement, please see Appendices A and B). G will normally have been determined and is quoted in the aircraft operating manual

Hence, at this condition: $C_{Do} = k C_{LE}^2$ (9)

And also, from (1), and assuming level flight $L = Mg = \frac{1}{2}\rho V_E^2 S C_{LE}$ (10)

Therefore, $C_{LE} = \frac{Mg}{\frac{1}{2}\rho V_E^2 S}$ (11)

We know that at this condition, $C_{Do} = k C_{LE}^2$ thus:- $k = \frac{C_{Do}}{C_{LE}^2}$ (12)

And since $G = \frac{C_L}{C_D}$ at this point $G = \frac{C_{LE}}{C_D} = \frac{C_{LE}}{2C_{Do}}$ (13)

Thus:- $C_{Do} = \frac{C_{LE}}{2G}$ (14)

Substituting (14) into (12) gives:- $k = \frac{C_{LE}}{2G C_{LE}^2} = \frac{1}{2G C_{LE}}$ (15)

and substituting (11) into (15) gives:-

$$k = \left(\frac{1}{2G}\right)\frac{½\rho V_E^2 S}{Mg} \tag{16}$$

So, from (16) one may now calculate k, since all other terms are known.

Now, from (9), (16) and (11):-

$$C_{Do} = k\,C_{LE}^2 = \left(\frac{1}{2G}\right)\left(\frac{½\rho V_E^2 S}{Mg}\right)\left(\frac{Mg}{½\rho V_E^2 S}\right)^2$$

$$\therefore C_{Do} = \left(\frac{1}{2G}\right)\left(\frac{Mg}{½\rho V_E^2 S}\right) \tag{17}$$

Then, inserting (16) and (17) into (8) this gives an estimate for the aircraft's longitudinal acceleration at the point of stall:-

$$\left(\frac{dV}{dt}\right)_S = -\left(\frac{½\rho V_S^2 S}{M}\right)\left[\left(\frac{1}{2G}\right)\left(\frac{Mg}{½\rho V_E^2 S}\right) + \left(\frac{1}{2G}\right)\left(\frac{½\rho V_E^2 S}{Mg}\right)\left(\frac{Mg}{½\rho V_S^2 S}\right)^2\right] \tag{18}$$

It may be seen that all terms in M, ½ ρ ,S cancel out in (18), giving:-

$$\left(\frac{dV}{dt}\right)_S = -V_s^2\left(\left(\frac{1}{2G}\right)\left(\frac{g}{V_E^2}\right)+\left(\frac{1}{2G}\right)\left(\frac{V_E^2}{V_S^4}\right)g\right) \tag{19}$$

$$= \frac{-g}{2G}\left(\frac{V_S^2}{V_E^2}+\frac{V_E^2}{V_S^2}\right) \tag{20}$$

This gives a value, from readily available aircraft data for the maximum magnitude of acceleration (which will have a negative sign) immediately prior to the stall event, when an aircraft is not in manoeuvring or climbing flight. The airspeed values, since they divide into each other may be treated in any convenient unit, g is conventionally in ms^{-2} and the value is for all normal purposes fixed. However, the equation (20) will give a value in ms^{-2}, which is inconvenient for flight use. Therefore a standard value of g=9.80665 will be applied and a conversion of 0.514 from ms^{-2} to kn/s will be applied. This gives the following:-

Therefore:- $$\left(\frac{dV}{dt}\right)_S = \frac{-9.54}{G}\left(\frac{V_S^2}{V_E^2}+\frac{V_E^2}{V_S^2}\right)\text{kn/s} \tag{21}$$

Before progressing further, it is appropriate to consider the nature of the airspeeds under discussion. An aircraft will indicate results in IAS, which for current purposes will be treated as CAS (Calibrated Airspeed) and the errors disregarded. It is theoretically possible that deceleration could instead be measured using an

accelerometer, but the combination of a comparatively low rate of deceleration and likely presence of pre-stall airframe buffet are such that this is not considered a sensible possibility. This would also entail fitting non-standard flight instrumentation; this has therefore not been explored. The origin of this analysis - equations (1) to (4) use TAS. In equation (20) the values are worked upon as ratios and so it is unimportant whether they are TAS or CAS since the ratio will be identical. But the result is expressed as TAS. Since for flight purposes TAS is rarely useable, it is necessary to transform this into a value in CAS. So, considering equation (21):-

The relationship between CAS and TAS is (by standard result):-

$$CAS = (TAS)\sqrt{\sigma} \tag{22}$$

So, a more useful form of equation (21) incorporates (22) allowing the result to be expressed in terms of CAS:-

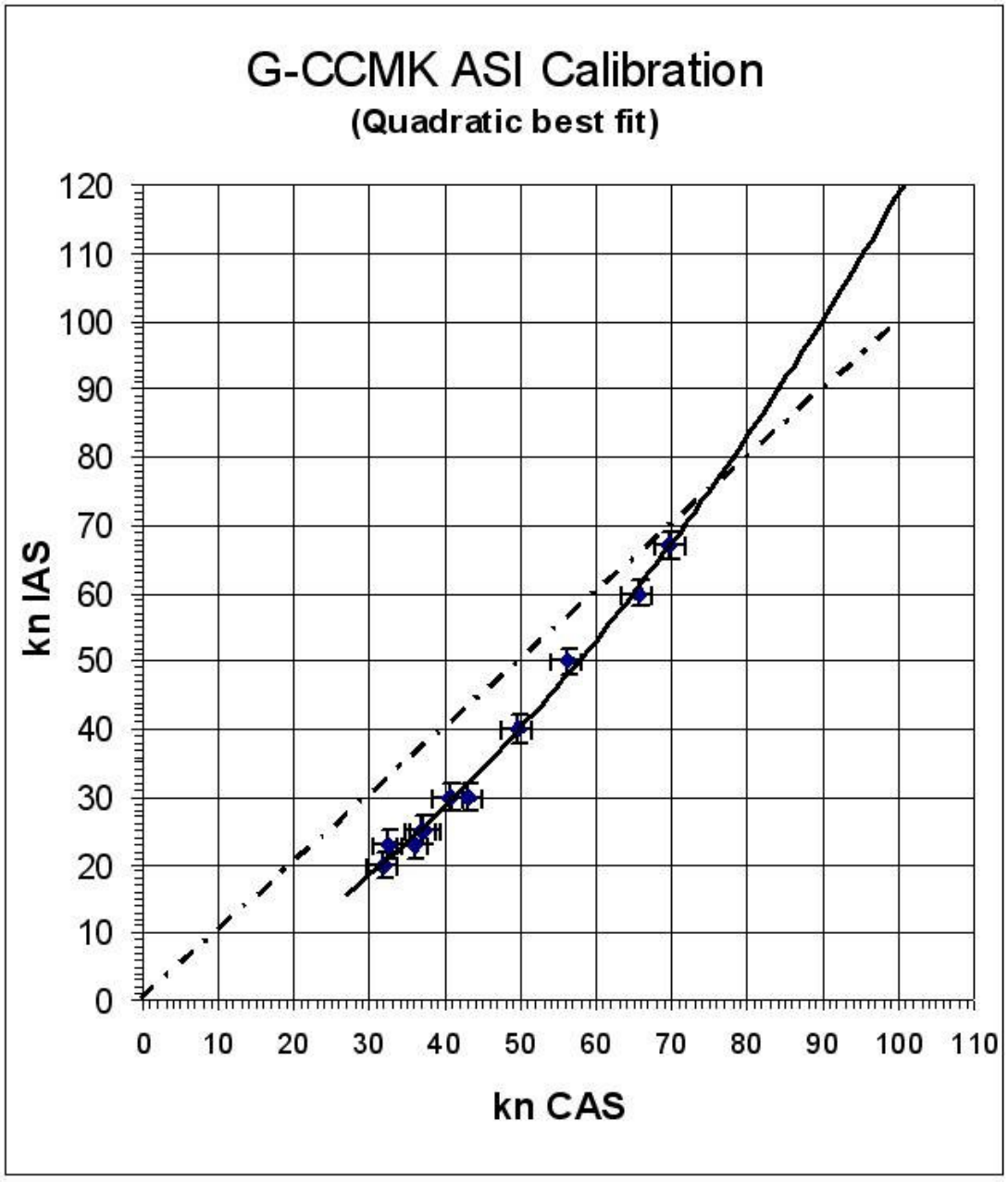
$$\frac{dV_{ind}}{dt} = -\frac{9.54\sqrt{\sigma}}{G}\left(\frac{V_S^2}{V_E^2} + \frac{V_E^2}{V_S^2}\right)$$

However, it has been found regularly that the form of the ASI calibration curve (for example **Figure 4** below) is such that the gradient of IAS versus CAS is not near to unity. Therefore for test work this gradient must be known, and incorporated into this transitional result, to become:-

$$\frac{dV_{ind}}{dt} = \frac{-\partial IAS}{\partial CAS}_{S} \cdot \frac{9.54\sqrt{\sigma}}{G}\left(\frac{V_S^2}{V_E^2} + \frac{V_E^2}{V_S^2}\right) \tag{23}$$

Figure 4, Typical microlight ASI Calibration curve

(dashed line represents IAS=CAS)



ISA defines σ by an exponential equation in terms of height (which should be borne in mind for any computer modelling purposes) however for the current purpose of considering overall altitude effect, look-up tables will suffice, as shown in Table 2 below.

Table 2, CAS:TAS comparison for different heights

Assuming that a value of 2.4 kn/s TAS had been obtained.

Standard Pressure Altitude (ft) (a)	TAS deceleration (kn/s) (b)	σ (c)	$\sqrt{\sigma}$ (d)	CAS Deceleration = TAS $\sqrt{\sigma}$ (kn/s) (e)
0	2.4	1	1	2.4
5,000	2.4	0.862	0.928	2.23
10,000	2.4	0.738	0.859	2.06
15,000	2.4	0.629	0.793	1.90

Thus: (1) The sea level condition (represented by TAS) is the worst case

(2) Up to 10,000 ft the IAS stall entry rate may reduce by up to 14% - which is significant enough to require adjustment of flight test results. However,

since an accuracy of deceleration rate of 30% is as good as might reasonably be hoped for from a test pilot, the sea level result may be used when calculating the stall entry rates to be used for flight test planning at any altitude. Microlight flight testing will not normally be carried out above 10,000 ft because above that height supplementary oxygen is required, which is not normal equipment in this class of aircraft. In any case, a normal height bracket for stall tests would be 3,000 to 5,000 ft sHp (Standard Pressure Altitude) where the maximum error is trivially small.

Notwithstanding the table above, a -1kn/s acceleration rate (1 kn/s deceleration) towards the stall event will still be required (for determination of performance stalling speeds). The worst case sea level value of deceleration rate should therefore be used when determining the safe proof case for flight test purposes (i.e. that is the deceleration rate into the stall up to which the aircraft must not show unacceptable stalling characteristics). Any further adjustments for CAS should be performed only where quantitative comparison with actual flight test data is required.

In order to provide any confidence in this result, it is essential to compare this to actual flight test data. **Table 3** following is based upon flight test data for individual aircraft as listed.

Table 3, Comparison of theory with test data for stall deceleration rates

Type	Reg.	Vs (kn CAS)	V_E (kn CAS)	ht (ft sHp)	$\sqrt{\sigma}$	G	$\frac{\left(\frac{dv}{dt}\right)_{calc.}}{2}$ (kn/s)	δt [3] (s)	$\frac{\left(\frac{dv}{dt}\right)_{true.}}{4}$ (kn/s)
X'Air 582 (1)[5]	G-BYCL	33.5	43	3000	0.949	6.7	3.05	7.75	1.23
Spectrum[6]	G-MWTE	35	30[7]	1500	0.992	7.4[2]	2.68	6.25	2.08
Thruster TST[8]	G-MTGR	28	45	2000	0.971	8.66	3.18	8.0	2.13
Cyclone AX3-503	Several	27	34	2800	0.959	6.92	2.93	4.75[9]	2.74
Avasud Mistral	G-MWIB	30	44	3000	0.957	11.1	2.15	8.38	1.67
Goldwing	G-MJRS	30	35	2000	0.971	12.1	1.60	6.3[10]	3.17
X'Air Jabiru (1)[11]	G-HITM	33	43	1800	0.974	6.8	3.12	6.0	2.50
Thruster TST Mk.1[12]	G-MVBT	33	40	2000	0.971	8.1	2.45	3.5	3.43
SkyRaider II(UK)[13]	G-SRII	38	48	1500	0.992	8.2	2.56	12	1.17

[2] From (23).
[3] Mean value from several tests for the time to decelerate from V_E to the stall whilst maintaining level flight.
[4] $dv/dt._{true}$ represents the best available approximation to the acceleration at the stall, given by $(V_E-V_S)/\delta t$
[5] From certification testing of first UK example.
[6] From testing by the author in a privately owned example.
[7] These are estimated values by extrapolation of test data, the aircraft stalled whilst still on the right hand side of the drag curve. Stalls were carried out from a trim speed of 43 kn.
[8] From testing a modified aircraft for approval under MAAN 1404. ASI calibration not available, so IAS is used.
[9] Deceleration in the AX3 was from 50 mph IAS (43 kn). Apparent stall was at 35 mph IAS = 30kn, which compares only moderately well to the TADS value of 31 mph at MTOW.
[10] Deceleration from 50 kn IAS. (Data obtained during performance testing of an example privately owned by the author).
[11] From certification flight test reports, aircraft was trimmed to 48 kn CAS prior to throttle closure.
[12] Example modified by fitment of BMW R100 engine, enclosed rear fuselage and doors, data extracted from flight testing for approval of the modifications. Throttle closed at 45 kn V_{trim}.
[13] During certification testing of the first UK example, flown at light weight (345kg), trim speed 55 mph IAS = 52 kn CAS. This aircraft developed into the Easy Raider before certification.

During flight tests, it was often noted that for microlight aircraft, the stalling characteristics are often poorly defined, such that there is some uncertainty concerning the precise starting moment of the stall event. Therefore there was probably significant lag between the aerodynamic stall and the perception of the stall. It must be remembered that at all times, apparent characteristics must be used in flight testing. Also however, it is known from published literature on unsteady aerodynamics that C_{Lmax} is greater when a rapid pitch-up occurs; clearly the greater the deceleration rate, the greater the pitch rate and so a greater deceleration rate is likely to result in a lower apparent stalling speed. A lack of appropriate facilities (e.g. a 15m+ section wind tunnel combined with a movable sting capable of pitch rates better than 30°/s nose-down motion in order to meaningfully simulate the post-stall pitch break) for conducting tests for this on wings with a 9 - 12m wingspan prevent this being quantified.

Therefore it is proposed to insert an additional term into (23), as shown below:-

$$\frac{dV_{ind}}{dt} = \frac{-9.54\tau_d\sqrt{\sigma}}{G}\left(\frac{V_S^2}{V_E^2}+\frac{V_E^2}{V_S^2}\right)\text{kn/s} \qquad (24)$$

Where the new term, τ_d is introduced, which will be termed the "deceleration time factor". This is estimated for the types previously considered in **Table 4** below.

Table 4, determination of deceleration time factor

Type	Reg.	Vs (kn CAS)	$\left(\frac{dv}{dt}\right)_{s\ calc.}$ (kn/s)	$\left(\frac{dv}{dt}\right)_{s\ true.}$ (kn/s)	$\left(\frac{dv}{dt}\right)_{s\ true} \Big/ \left(\frac{dv}{dt}\right)_{s\ calc} = \tau_d$
X'Air 582 (1)	G-BYCL	33.5	3.05	1.23	0.403
Spectrum	G-MWTE	35	2.68	2.08	0.776
Thruster TST	G-MTGR	28	3.18	2.13	0.670
Cyclone AX3-503	Various	29	2.93	2.74	0.935
Aviasud Mistral	G-MWIB	30	2.15	1.67	0.777
Goldwing	G-MJRS	30	1.60	3.17	1.98
X'Air Jabiru(1)	G-HITM	33	3.12	2.50	0.801
Thruster TST Mk.1	G-MVBT	33	2.45	3.43	1.40
SkyRaider II(UK)	G-SRII	38	2.56	1.17	0.457

At first sight this shows a very large variation in values of τ_d, hence this was explored further. Personal experience had shown that aircraft in this class tend to show a far more well-defined stall at higher wing loadings, and so the relationship with wing loading was explored. **Table 5** shows the wing loading of each of the test aircraft described above, and **Figure 5** plots the determined value of τ_d versus the wing loading W/S at the time of each test. (The figure omits the results for the Goldwing and Thruster TST.1, which otherwise significantly skew the best-fit curve away from all other points. Both of these are older designs which are known to have pitch control characteristics that might not necessarily be accepted if current practices were followed – very shallow apparent longitudinal static stability in the case of the Goldwing, and a very wide trim speed band in the case of the Thruster TST. It is

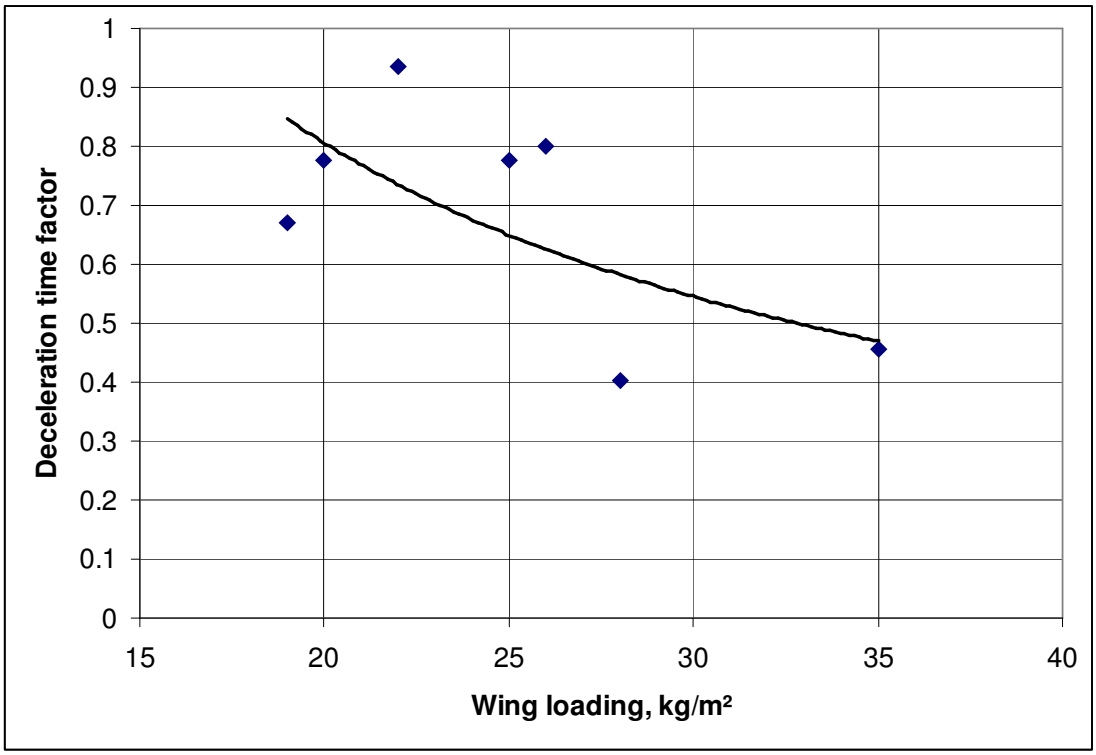
suspected that the unusual pitch control characteristics of these aircraft significantly affect the pilot's perception of the stalling characteristics.)

Table 5, Wing loadings for test aircraft at time of each stalling test

Type	Reg.	W/S (kg/m²)
X'Air 582 (1)	G-BYCL	28
Spectrum	G-MWTE	25
Thruster TST	G-MTGR	19
Cyclone AX3-503	Various	22
Aviasud Mistral	G-MWIB	20[14]
Goldwing	G-MJRS	20[15]
X'Air Jabiru (1)	G-HITM	26
Thruster TST Mk.1	G-MVBT	25
SkyRaider II(UK)	G-SRII	35

[14] The Aviasud Mistral is a biplane.
[15] Including Canard. The Goldwing is the only canard aircraft listed.

Figure 5, Deceleration factor versus wing loading (Goldwing and Thruster TST Omitted)



The curve shown is a power regression of the form:

$$y=A(W/S)^{-1} \quad (25)$$

which gives a moderate (R^2=0.42) fit.

Where A is a derived term of value A=16.4 m²/kg.

Final form of the equation

We therefore find that the acceleration rate of a microlight aircraft as it approaches the stall, is defined by the following equation, where the aircraft has suffered a sudden power failure and the pilot attempts to maintain level flight.

$$\frac{dV_{ind}}{dt} = \frac{-9.54\tau_d\sqrt{\sigma}}{G}\left(\frac{V_S^2}{V_E^2}+\frac{V_E^2}{V_S^2}\right) \qquad (27)$$

Where, τ_d, the deceleration time factor is estimated by τ_d=16.4/(W/S); G is the best glide ratio for the aircraft; V_s is the stall speed; and V_E is the best range glide speed. Although the term is retained for analysis purposes, when planning flight tests, it is safe and more convenient to assume that $\sigma = 1$. The accuracy of (27) is investigated in **Table 6** below.

Table 6, Evaluating the accuracy of (27)

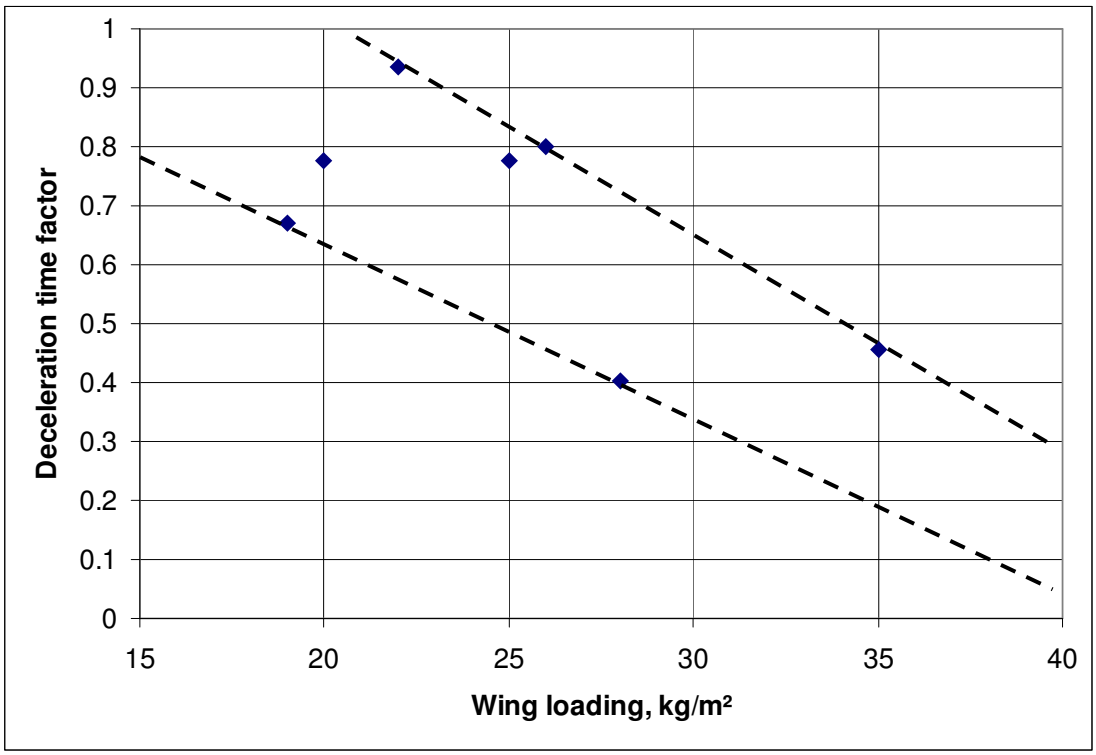
Type	Reg.	Vs (kn CAS)	V_E (kn CAS)	ht (ft sHp)	$\sqrt{\sigma}$	G	W/S (kg/m²)	$\left(\frac{dv}{dt}\right)_{calc.}$ (kn/s)	$\left(\frac{dv}{dt}\right)_{true.}$ (kn/s)
X'Air 582 (1)	G-BYCL	33.5	43	3000	0.949	6.7	28	1.78	1.23
Spectrum	G-MWTE	35	30	1500	0.992	7.4	25	1.76	2.08
Thruster TST	G-MTGR	28	45	2000	0.971	8.66	19	2.74	2.13
Cyclone AX3-503	Various	27	34	2800	0.959	6.92	22	2.18	2.74

Type	Reg.	Vs (kn CAS)	V_E (kn CAS)	ht (ft sHp)	$\sqrt{\sigma}$	G	W/S (kg/m²)	$\left(\frac{dv}{dt}\right)_{calc.}$ (kn/s)	$\left(\frac{dv}{dt}\right)_{true.}$ (kn/s)
Aviasud Mistral	G-MWIB	30	44	3000	0.957	11.1	20	1.76	1.67
Goldwing	G-MJRS	30	35	2000	0.971	12.1	20	1.32	3.17
X'Air Jabiru (1)	G-HITM	33	43	1800	0.974	6.8	26	1.97	2.50
Thruster TST Mk.1	G-MVBT	33	40	2000	0.971	8.1	25	1.61	3.43
SkyRaider II(UK)	G-SRII	38	48	1500	0.992	8.2	35	1.32	1.17

However this formula (demonstration of the accuracy of which is given in the next table) gives the best estimate; this is by definition since it uses the best fit curve to the available data. In order to determine test conditions for certification testing, bounds of greatest and least magnitude deceleration rates are required.

Given that in virtually all cases the stalling characteristics are more severe at higher deceleration rates (and if they are not, then the 1kn/s case must in any case be examined so as to satisfy specific certification requirements) an alternative approach is to determine a value of τ_d which will give the greatest magnitude value of deceleration. This can be achieved by defining the linear relationship (available data does not justify a higher order curve in this case) which gives the greatest value of deceleration amongst the values in the analysis above. A worst-case straight line may be marked on the previous figure as shown in **Figure 6**:-

Figure 6, Deceleration factor versus wing loading (Goldwing and Thruster TST Omitted), with straight lines plotted giving greatest and least magnitude acceleration rates.



These two lines define the bounds of maximum (upper line) and minimum (lower line) acceleration that should be experienced in the event of level flight being maintained following an engine failure. These may be defined by the following:-

Greatest magnitude acceleration: $\tau_d = 1.817 - 0.0389 W/S$ (28)

Least magnitude acceleration: $\tau_d = 1.223 - 0.0295 W/S$ (29)

Inserting (28) and (29) into (27) one obtains bounds for the range of level-flight acceleration rates that are likely to be experienced prior to an inadvertent stall, which are:-

$$\frac{dV_{ind}}{dt} = \frac{-\left(17.3 - 0.37 W/S\right)\sqrt{\sigma}}{G}\left(\frac{V_S^2}{V_E^2} + \frac{V_E^2}{V_S^2}\right) \qquad (30)$$

and,

$$\frac{dV_{ind}}{dt} = \frac{-\left(11.7 - 0.28 W/S\right)\sqrt{\sigma}}{G}\left(\frac{V_S^2}{V_E^2} + \frac{V_E^2}{V_S^2}\right) \qquad (31)$$

Physical significance of τ_d

Whilst an investigation has not been attempted into the physical significance of τ_d, the fact that it is shown to be a function of wing loading indicates that there must be some relationship to an aircraft's design and loading; it is likely that other variables will also be significant – for example the apparent longitudinal static stability, and the severity of the aircraft's post-stall gyrations (in particular of any pitch break) are likely to be significant in determining τ_d's value. Whilst not explored herein, it is likely that the physical significance, and the factors leading to a given value of τ_d will adopt greater importance within any subsequent development of this work.

Factors affecting the test results within this work

It was impracticable during the course of this research to turn off the engines of the aircraft under test, since a number did not possess any ready means of airborne-restart. Therefore, in each case engines were set to the minimum achievable idle setting (generally the lowest which does not lead to any risk of engine stoppage whilst stationary on the ground); this provides a reasonable approximation to the behaviour of an aircraft with the engine stopped but some, unquantified, effects may nonetheless exist.

A further area where users of this data should apply caution is that all data is from normal cockpit instrumentation, which must inevitably contain indication and lag errors. The lack of availability (or affordability) of flight-test instrumentation on microlight aeroplanes makes this inevitable; this is likely also to be the case with any future work.

It should also be noted that the extreme case of an engine failure during a full power climb has not been addressed. This is an important and extreme case, which has been known to cause loss of control, particularly in weightshift controlled aeroplanes **[10]**; however is a separate issue to that which is addressed in this work.

Use of this work.

This work presents a tool by which the greatest deceleration rate in the event of an engine failure of a microlight aeroplane may be predicted. Since existing test schedules for microlight aeroplanes already cover a range of decelerations from 1kn/s to 5kn/s it is unlikely that test planning would commonly be changed by this. However, it provides a mechanism by which the validity of the test conditions, for a

particular type, may nonetheless be checked, and in this context usefully ensure the validity of test results in ensuring the suitability of the aircraft for normal use.

Further research.

This work has potential to be adapted to other classes of lightweight aircraft – for example to consider the immediate deceleration and consequent effects upon rotor speed of a gyroplane following an engine failure, or to consider the potential consequences of a cable failure during a glider launch. It is very likely that such further work will require the researcher to investigate the physical significance of, and factors affecting the deceleration time factor, τ_d .

Conclusions

Because stalling characteristics are a function of deceleration rate, it is important to ensure that the deceleration rates used in certification testing of aeroplanes are representative of the range of rates which may be met in service. For microlight aeroplanes, the maximum anticipated deceleration rate is that associated with a sudden loss of power following which the pilot attempts to maintain altitude, sacrificing airspeed to do so. The deceleration rate may be described by (27):-

$$\frac{dV_{ind}}{dt} = \frac{-9.54\tau_d\sqrt{\sigma}}{G}\left(\frac{V_S^2}{V_E^2} + \frac{V_E^2}{V_S^2}\right)$$

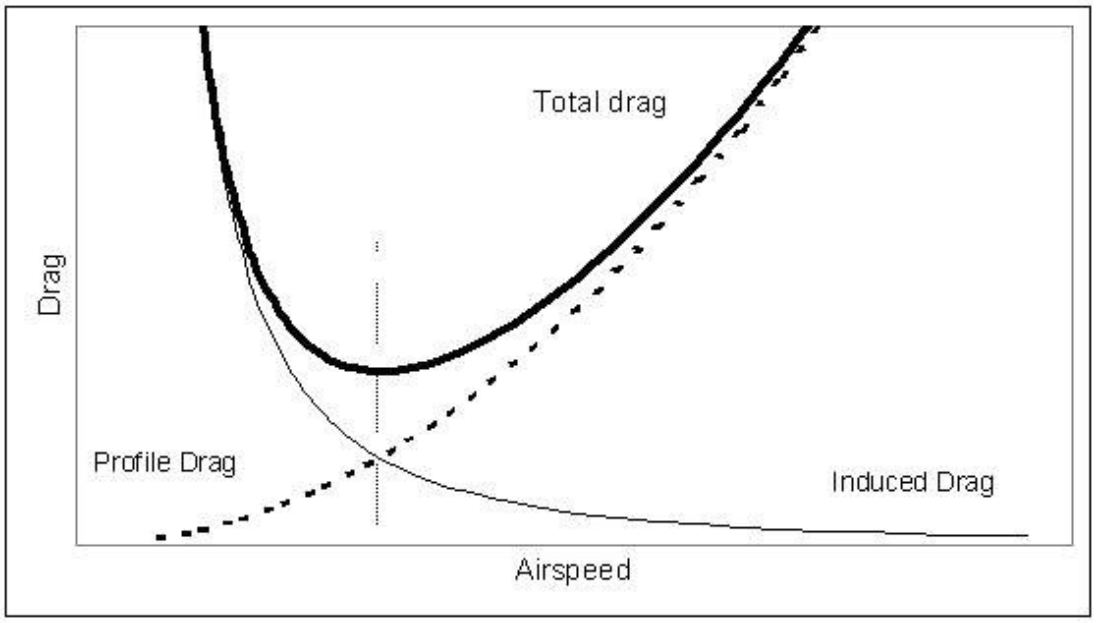
Where τ_d is a function of aircraft characteristics, but has a maximum value estimated as (28):-

$$\tau_d = 1.817 - 0.0389 W/S$$

Appendix A - Proof that $C_{Do} = kC_L^2$.

Total subsonic aircraft drag is conventionally regarded as being made up of two components **[11]** which are induced drag, defined by $D_i = \frac{1}{2}\rho V^2 S.k.C_L^2$ and profile (or form) drag which is defined by $D_p = \frac{1}{2}\rho V^2 S C_{D_o}$. **Figure 7** below is shown a generic graph for these two components and the total value of drag, defined by $D = D_i + D_P$.

Figure 7, Generic polar for total drag upon a subsonic aircraft



By inspection, total drag is at a minimum at the airspeed where Profile drag is equal to induced drag. Therefore at this speed, $\frac{1}{2}\rho V^2 S.k.C_L^2 = \frac{1}{2}\rho V^2 S C_{Do}$ and hence, $k.C_L^2 \equiv C_{Do}$.

Appendix B - proof that $(L/D)_{MAX}$ is identical to the best glide ratio.

The curve of total drag against speed is known from all available experimental data to show a clear minimum. Since $D = L\left(\frac{D}{L}\right)$ *and assuming level flight or a shallow glide angle* $L = W$, $D = W\left(\frac{D}{L}\right)$. *Thus the speed at which the minimum value of drag occurs is co-incident with the point where L/D is at a maximum. It is known that L/D is identical to the glide ratio, and thus to the best glide ratio since it is at a maximum at this speed.*

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