Lesson 17

Stall Speed and $C_{L_{max}}$ from Test Flight Data

The lift coefficient is

$$C_L = \frac{2nW}{\rho V^2 S} \quad \text{when } \theta \approx 0$$

$$V_{cruise} > V_{s_{1g}} > V_{min}$$

How slowly should stall be approached? FAA says $-1\ ^{kt}/_{s}$

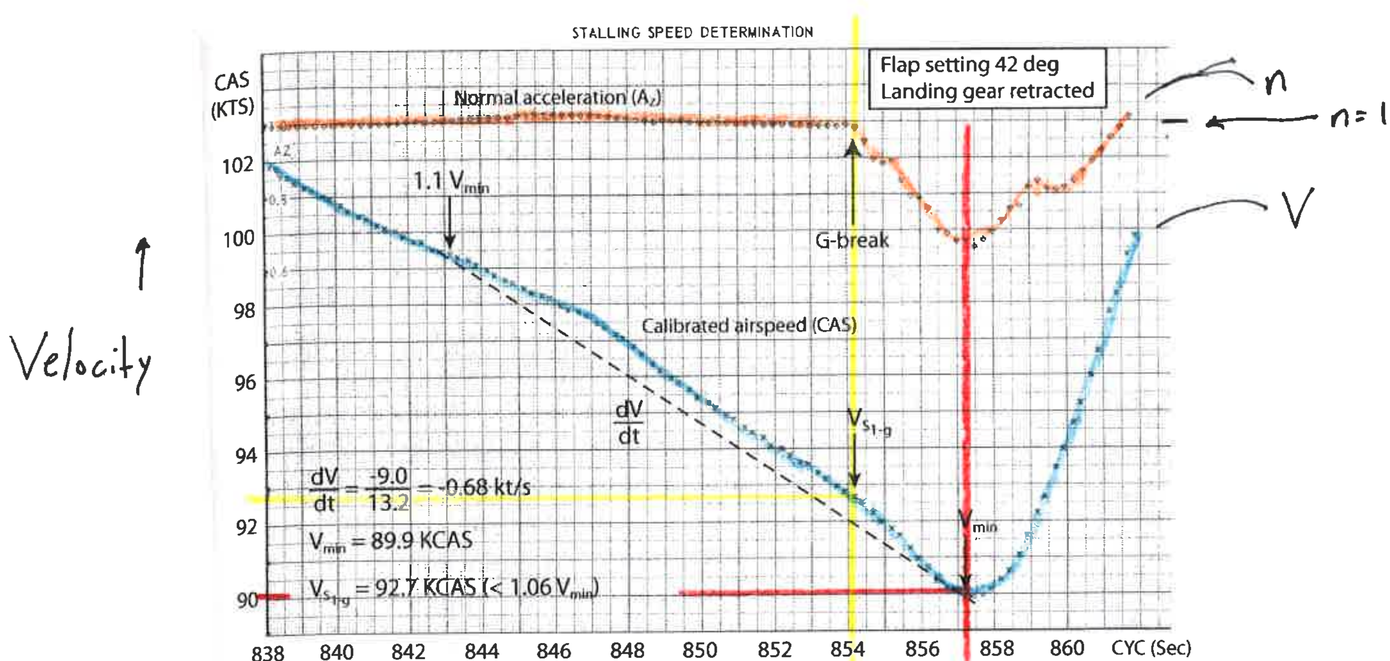


Figure 26.2 - Stall speed determination of a Fokker F-28 Mk 4000.
Source: Fokker Report H-28.40-27.001

Time →

Source: Aerodynamic Design of Transport Aircraft, Obert

Q: For an aircraft certified after 2002, what is the stall speed?

V_{min} ✗ , $\boxed{V_{s_{1\text{-}g}}}$, Something else? ✗

Q: Given the following data (flight test) for the Boeing 747, determine the stall speed if the aircraft were under the 2002 rules (it's not).

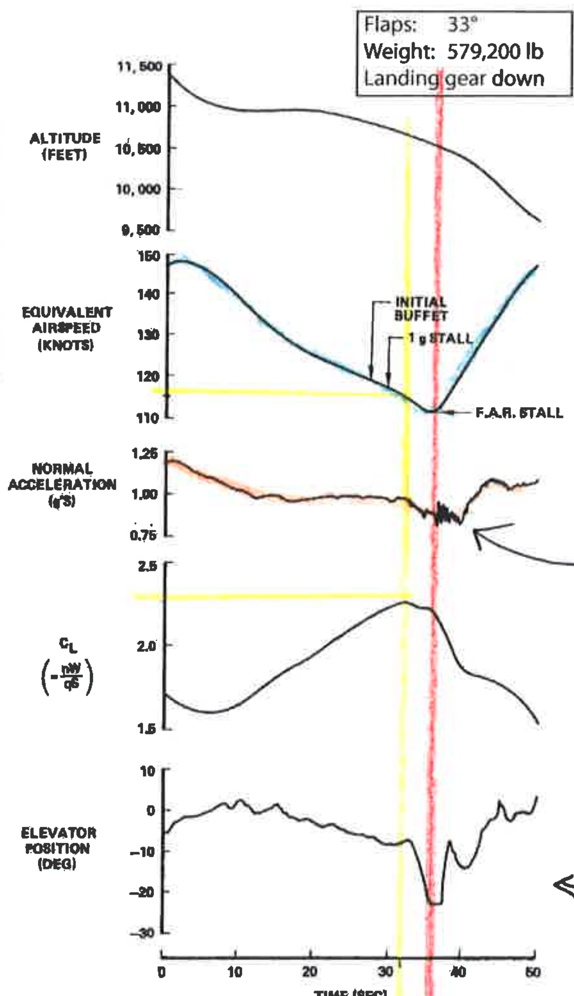


Figure 26.1 - Flight record of a stall manoeuvre of a Boeing 747
Source: AGARD CP-102, Paper No. 21

← $V_{s_{1g}} \approx 116$ KEAS

Noisy?! What is this?

← $C_{L_{max_{1g}}} \approx 2.25$

← Elevator position!

CONTROLLABILITY AND MANEUVERABILITY

§ 23.143 General.

(a) The airplane must be safely controllable and maneuverable during all flight phases including—

(1) Takeoff;

(2) Climb;

(3) Level flight;

(4) Descent;

(5) Go-around; and

(6) Landing (power on and power off) with the wing flaps extended and retracted.

(b) It must be possible to make a smooth transition from one flight condition to another (including turns and slips) without danger of exceeding the limit load factor, under any probable operating condition (including, for multiengine airplanes, those conditions normally encountered in the sudden failure of any engine).

(c) If marginal conditions exist with regard to required pilot strength, the control forces necessary must be determined by quantitative tests. In no case may the control forces under the conditions specified in paragraphs (a) and (b) of this section exceed those prescribed in the following table:

Values in pounds force applied to the relevant control	Pitch	Roll	Yaw
(a) For temporary application:			
Stick	60	30	....
Wheel (Two hands on rim)	75	50	....
Wheel (One hand on rim)	50	25	....
Rudder Pedal	....	....	150
(b) For prolonged application	10	5	20

Stick Position Gradients

From earlier (lesson 16), we found that at trim,

$$\frac{d\delta_{trim}}{dC_{L_{trim}}} = \frac{C_{m_\alpha}}{C_{L_{\delta_e}} C_{m_\alpha} - C_{L_\alpha} C_{m_{\delta_e}}}$$

Using the chain rule, we can find the stick position vs ~~...~~ Velocity

$$\frac{d\delta_e}{dV} = \frac{d\delta_e}{dC_L} \frac{dC_L}{dV} \qquad \text{with } C_L = \frac{2W}{\rho V^2 S} \Rightarrow \frac{dC_L}{dV} = -\frac{4W}{\rho V^3 S}$$

$$= \frac{C_{m_\alpha}}{C_{L_{\delta_e}} C_{m_\alpha} - C_{L_\alpha} C_{m_{\delta_e}}} \cdot \frac{-4W}{\rho V^3 S}$$

$$\frac{d\delta_e}{dV} = \underbrace{\frac{4W}{\rho V^3 S}}_{\text{"+"}} \cdot \frac{\overbrace{C_{m_\alpha}}^{cg}}{\underbrace{C_{L_\alpha}}_{+}\underbrace{C_{m_{\delta_e}}}_{-} - \underbrace{C_{L_{\delta_e}}}_{+}\underbrace{C_{m_\alpha}}_{cg}} \qquad \text{with } C_{m_\alpha} = C_{L_{\alpha_w}}\left(\frac{X_{cg}}{\bar{c}} - \frac{X_{ac}}{\bar{c}}\right) - \text{tail contribution}$$

pilots <u>strongly</u> demand $\frac{d\delta_e}{dV} > 0$

elevator TED ("push") → Faster

" TEU ("pull") → Slower

$C_{m_\alpha} < 0$

$$\frac{d\delta_e}{dV} = + \frac{-}{(-)-(-)} = + \frac{-}{-} = + \qquad \text{with } C_{m_\alpha} < 0$$

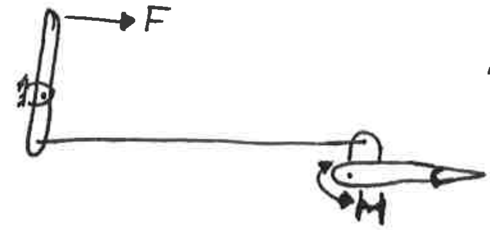
$C_{m_\alpha} > 0$

$$\frac{d\delta_e}{dV} = + \frac{+}{(+)(-)-(+)(+)} = + \frac{+}{-} = -$$

A stable aircraft naturally has $\frac{d\delta_e}{dV} > 0$.

An unstable aircraft can be both unstable in the pitch angle <u>and</u> velocity.

Stick Forces



A force on the stick generates a moment on the surface.

Look first at the hinge moment generated by the aerodynamics

$$H = q S_e \bar{c}_e C_{h_e} \approx q S_e \bar{c}_e \Big(\underbrace{C_{h_0}}_{\substack{\text{Static}\\\text{offset}\\\text{"Camber?"}}} + \underbrace{C_{h_{\alpha_t}} \alpha_t}_{\text{AOA}} + \underbrace{C_{h_{\delta_e}} \delta_e}_{\text{elevator}} + \underbrace{C_{h_{\delta_t}} \delta_t}_{\text{trim tab}} \Big)$$

From before, the AOA at the tail

$$\alpha_t = \alpha - \epsilon + i_h = \alpha\left(1 - \frac{d\epsilon}{d\alpha}\right) + i_h - \epsilon_0$$

$$H = q S_e \bar{c}_e \left(C_{h_0} + C_{h_{\alpha_t}} \left(\alpha\left(1 - \frac{d\epsilon}{d\alpha}\right) + i_h - \epsilon_0 \right) + C_{h_{\delta_e}} \delta_e + C_{h_{\delta_t}} \delta_t \right)$$

How does the Moment H vary with Velocity?

$$\frac{dH}{dV} = \underset{\substack{\uparrow\\\text{calculate}}}{\frac{dH}{d\delta_e}} \; \underset{\substack{\uparrow\\\text{already}\\\text{derived}\\\text{this}}}{\frac{d\delta_e}{dV}} = \underbrace{\left(q S_e \bar{c}_e C_{h_{\delta_e}} \right)}\underbrace{\left(\frac{4W}{\rho V^3 S} \right)\left(\frac{C_{m_\alpha}}{C_{L_\alpha} C_{m_{\delta_e}} - C_{L_{\delta_e}} C_{m_\alpha}} \right)}$$

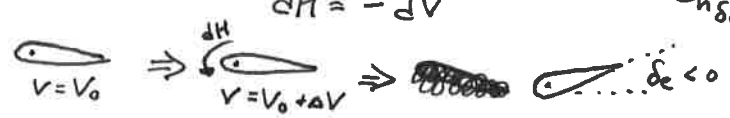
$$= \eta \frac{2W}{V_\infty} \left(\frac{S_e}{S} \right) \bar{c}_e C_{h_{\delta_e}} \frac{C_{m_\alpha}}{C_{L_\alpha} C_{m_{\delta_e}} - C_{L_{\delta_e}} C_{m_\alpha}}$$

In the small δ_e region and a stable aircraft ($C_{m_\alpha} < 0$)

$$= \left(\frac{+}{+}\right)\left(\frac{+}{+}\right)(+)(-) \frac{(-)}{(+)(-) - (+)(-)} \qquad = + \qquad = (-)$$

depends only on $C_{h_{\delta_e}}$ when $C_{m_\alpha} < 0$

Increasing the velocity decreases H: $dH = -dV$



$\Rightarrow \Delta C_m > 0$
nose rises

$$\boxed{\frac{dH}{dV} < 0 \text{ is a speed stable aircraft in pitch}}$$

Stick Forces

By conservation of energy, $F\, l_s \Delta\theta_s = H \delta_e$ for small δ_e and $\Delta\theta_s$

$$F = \underbrace{\frac{\delta_e}{\Delta\theta_s}\frac{1}{l_s}}_{\text{Gearing ratio and stick length}} H = G_e H$$

(The stick force counteracts a hinge moment H)

$$F = G_e\, q \eta S_e \bar{c}_e C_{h_e} = G_e\, q \eta S_e \bar{c}_e \left(C_{h_0} + C_{h_{\alpha_t}}\left(\alpha\left(1 - \frac{d\epsilon}{d\alpha}\right) + i_h - \epsilon_0\right) + C_{h_{\delta_e}} \delta_e + C_{h_{\delta_t}} \delta_t \right)$$

But write α, δ_e as:

$$\alpha = \frac{\partial \alpha}{\partial C_L} C_L + \alpha_0 = \frac{\partial \alpha}{\partial C_L}\frac{W}{qS} + \alpha_0 \qquad \text{where } \alpha_0 \text{ is } C_L = 0$$

$$\delta_e = \frac{d\delta_e}{dC_L} C_L + \delta_{e_0} = \frac{d\delta_e}{dC_L}\frac{W}{qS} + \delta_{e_0} \qquad \delta_{e_0} \text{ is for } C_L = 0$$

$$F = G_e q \eta S_e \bar{c}_e \left(C_{h_0} + C_{h_{\alpha_t}}\left(1 - \frac{d\epsilon}{d\alpha}\right)\alpha_0 + C_{h_{\alpha_t}}(i_h - \epsilon_0) + C_{h_{\delta_e}} \delta_{e_0} \right)$$

$$+ G_e \eta S_e \bar{c}_e \left(C_{h_{\alpha_t}}\left(1 - \frac{d\epsilon}{d\alpha}\right)\frac{d\alpha}{dC_L}\frac{W}{S} + C_{h_{\delta_e}}\frac{d\delta_e}{dC_L}\frac{W}{S} \right)$$

Notice! One term has the static offsets and dynamic pressure.
The other term has wing loading and <u>no</u> dynamic pressure!!

The stick force is <u>approximately</u>

$$F = \eta G S_e \bar{c}_e \left(q A + \frac{W}{S}\frac{C_{h_{\delta_e}}}{C_{m_{\delta_e}}} SM_{free} \right)$$

$$A = C_{h_0} + C_{h_\alpha}\left(1 - \frac{d\epsilon}{d\alpha}\right)\alpha_0 + C_{h_\alpha}(i_h - \epsilon_0) + C_{h_{\delta_e}} \delta_{e_0}$$

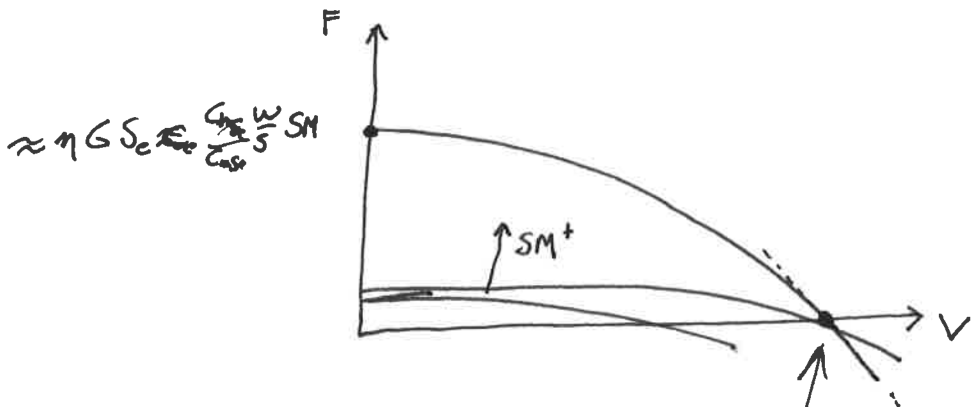
and SM_{free} is the static margin (elevator free)

Stick Force Gradients

How does F change with velocity?

$$\frac{dF}{dV} \approx \eta G_e S_e \bar{c}_e \tfrac{1}{2}\rho V \cdot 2 \cdot A = C \cdot V$$

plot F vs V

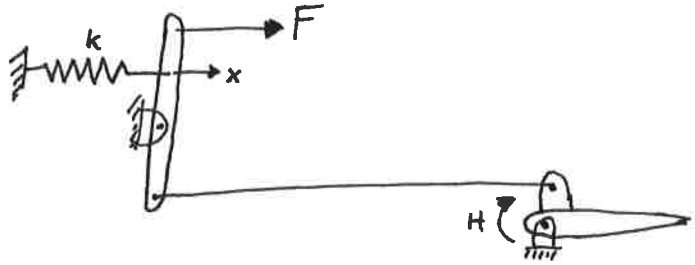


At this point, $qA + \frac{W}{S}\frac{C_{h\delta_e}}{C_{m\delta_e}} SM_{free} = 0$

Thus, $A < 0$ and thus $\frac{dF}{dV} < 0$

Notice that given a particular trim speed,
a <u>larger</u> Static Margin gives a
<u>more</u> speed stable aircraft (i.e. the slope of F wrt V is more negative)

Adding a spring



The spring adds

$$F_{spring} = Kx = \bar{k}\,\delta_e \quad \text{where } \bar{k} \text{ is the geometric gear ratio}$$

The total stick force is $F = F_{aero} + F_{spring}$

$$F = \eta G S_e \bar{c}_e \left(qA + \frac{W}{S} \frac{C_{h_{\delta_e}}}{C_{m_{\delta_e}}} SM_{Free} \right) + \bar{k}\,\delta_e$$

$$\delta_e = \frac{d\delta_e}{dC_L} C_L + \delta_{e_0} = \frac{d\delta_e}{dC_L} \frac{W}{qS} + \delta_{e_0}$$

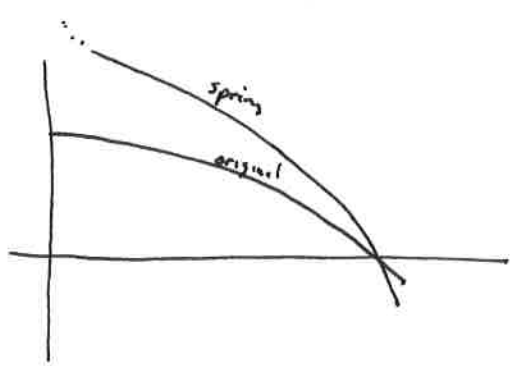
$$= \eta G S_e \bar{c}_e \left(qA + \frac{W}{S} \frac{C_{h_{\delta_e}}}{C_{m_{\delta_e}}} SM_{Free} \right) + \bar{k} \frac{1}{C_{L_{\delta_e}}} \frac{W}{qS} + \bar{k}\,\delta_{e_0}$$

Now

$$\frac{dF}{dV} \approx \underbrace{C_1 \cdot V}_{\text{from previous}} + \bar{k} C_2 \frac{1}{V^3}$$

Thus at low speeds, adding a spring keeps $\frac{dF}{dV}$ negative (and stable)

$\boxed{\text{This is a common fix to overly sensitive control systems}}$



Stick Force per "g"

How many pounds does the pilot need to pull to generate a load factor n:

$$\frac{\partial F}{\partial n} \approx \eta \bar{q} S_e \bar{c}_e G_e \left(C_{L_{trim}} \frac{C_{h_{\delta e}}}{C_{m_{\delta e}}} SM_{free} + \left(\frac{g(X_{ac} - X_{cg})}{U_\infty^2} \right) \left(C_{h_\alpha} - \frac{1.1 C_{h_{\delta e}}}{\tau_e} \right) \right)$$

A fighter aircraft is different than a bomber.

- too small and the pilot can "pull the wings off"
- too high and the pilot struggles to perform maneuvers

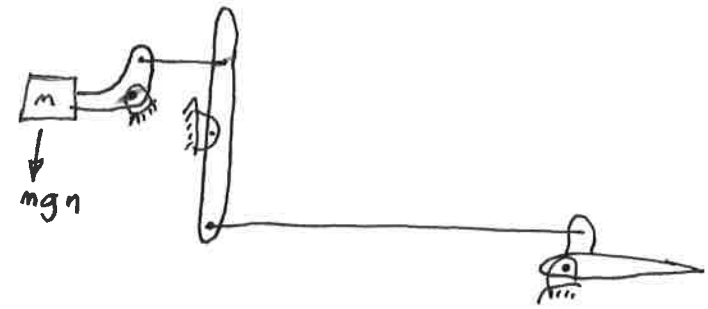
FAR 23 and 25 give requirements based on weight

$$F_{n\text{-}max} > 20 \text{ lbs} \quad \text{stick}$$

$$> 15 \text{ lbs} \quad \text{yoke}$$

"There must be no excessive decrease in the gradient of the curve of stick force versus maneuvering load factor"

Bob Weight (fix a low $\frac{dF}{dn}$)



As the load factor increases, the weight of the bob increase

This pulls the stick forward, which <u>increases</u> $\frac{dF}{dn}$.

See:

tiny.cc/AEM368Bob

30 Tail surface design

Tail surfaces perform three functions:

1. They provide static and dynamic stability.
2. They enable aircraft control.
3. They provide a state of equilibrium in each flight condition.

The first and second items are covered in detail in textbooks and courses on Stability and Control.
The ability to maintain a state-of-equilibrium is often taken for granted, yet to cover extreme flight conditions it often sets design requirements for horizontal and vertical tail surfaces and their control surfaces.

Examples are minimum control speed with a failed engine (V_{MC}), extreme out-of-trim conditions or maximum cross-wind capability.

In general the following design requirements can be formulated for tail surfaces:

1. They shall provide a sufficiently large contribution to static and dynamic longitudinal, directional (and sometimes lateral) stability. This determines primarily their lift gradients

   $\frac{dC_{L_h}}{d\alpha_h} S_h$ and $\frac{dC_{L_v}}{d\alpha_v} S_v$.

   This requires a maximum aspect ratio and for high aspect ratios minimum sweep.

2. They shall provide sufficient control capability, which again determines their lift slope. This also requires a maximum aspect ratio and for high aspect ratios minimum sweep.

3. Control shall be possible with acceptable control forces. This requires a maximum aspect ratio because control force

   $$F = C_h \tfrac{1}{2}\rho V^2 S_c \bar{c}_c$$

   where

   C_h = hinge moment coefficient
   $\tfrac{1}{2}\rho V^2$ = dynamic pressure
   S_c = control surface area
   $\bar{c}_c$ = control surface mean aerodynamic chord

4. The tail surfaces shall be able to cope with high tailplane angles-of-attack, both for the horizontal tail (in particular at higher speeds with flaps deflected) and for the vertical tail surface (high cross-winds). In this case a low aspect ratio is required and sweep is beneficial. The requirement to be able to cope with high tailplane and fin angles-of-attack is aggravated when flight in icing conditions is possible.

5. The tail surfaces shall be able to provide a maximum force sufficiently large to balance the total tail-off forces and moments so that static equilibrium is achieved in all flight conditions. This leads to specific requirements on tail surface areas and on the maximum lift coefficient for the tail surfaces with varying degree of control surface deflection, including the effect of ice roughness.

6. For high-speed aircraft the Mach-number at which serious flow separation occurs shall preferably lie above the design dive Mach-number M_D. Serious flow separation on the stabilizer will aggravate the effect of the changes in tail-off pitching moment due to changes in the wing flow (See pages 29.1-29.5). This applies in particular to aircraft with reversible control systems.

Therefore on stabilizers for high-speed aircraft the sweep angle is often about 5 deg higher than on the wing. Furthermore the section is often 1 or 2% (relative to the chord) thinner than on the outboard wing.

Note that flow break-down on the tail surfaces should preferably not occur below M_D also with deflected control surfaces required for small side slip corrections or pull-up manoeuvres with $n = 1.5$.

Furthermore the following shall be kept in mind:

1. A high aspect ratio has an adverse (although relatively small) effect on weight. Also, in particular for T-tails the flutter analysis requires extra care. A few degrees anhedral (negative dihedral) has a very beneficial effect.

2. Excessive taper ratio may lead to premature tip stall. This risk is higher when sweep is applied although the stall is then more gradual with less loss in lift. On the other hand tapering leads to lower weight.

Source:

Aerodynamic Design of Transport Aircraft

Obert

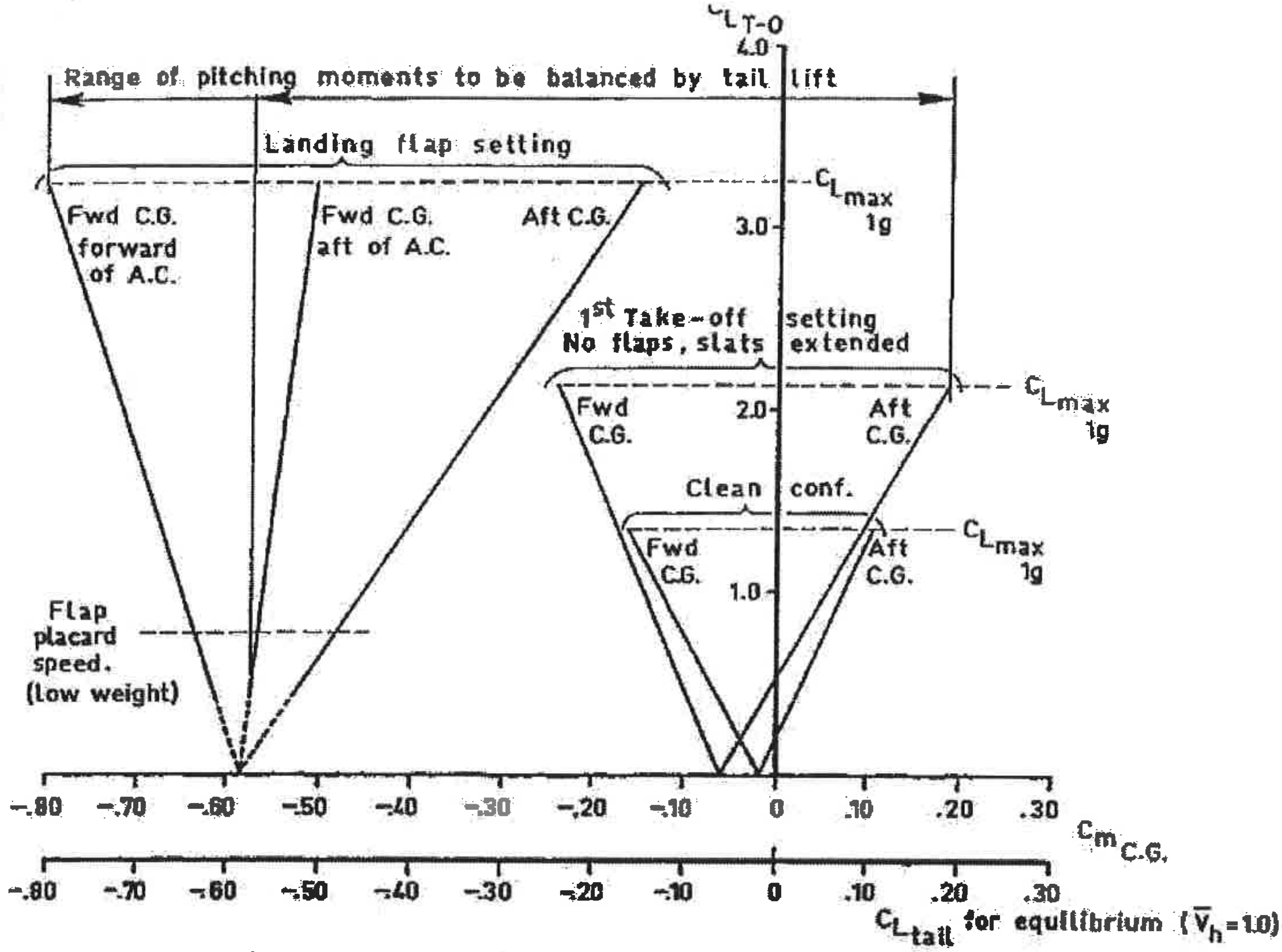
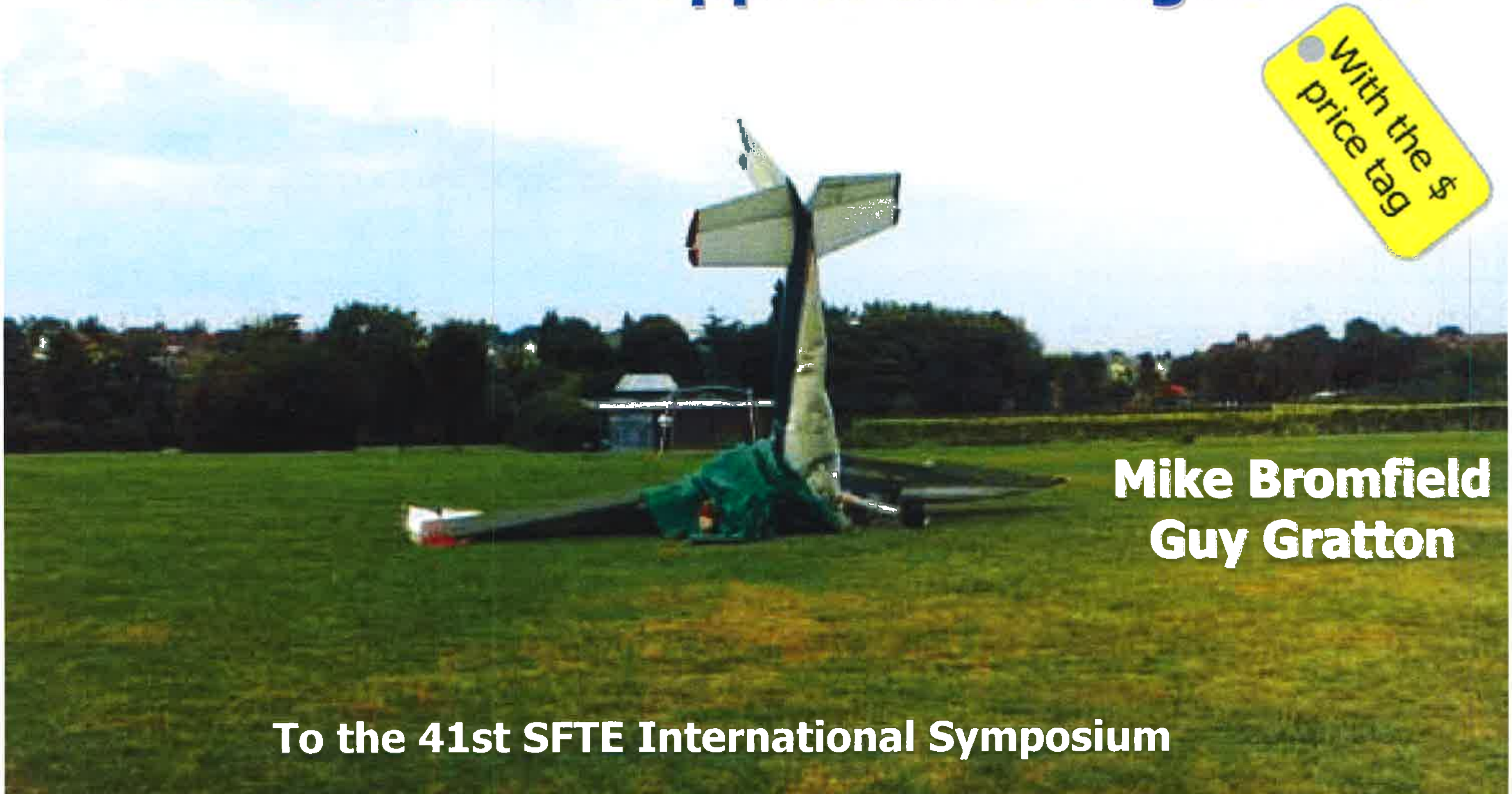


Figure 31.11 - Pitching moment, aircraft less tail. Source: AGARD CP-160, Paper No. 10
Flap placard speed = maximum speed at which aircraft is allowed to fly with flaps deployed

Loss of Control of Light Aircraft: A cost effective approach to Flight Test

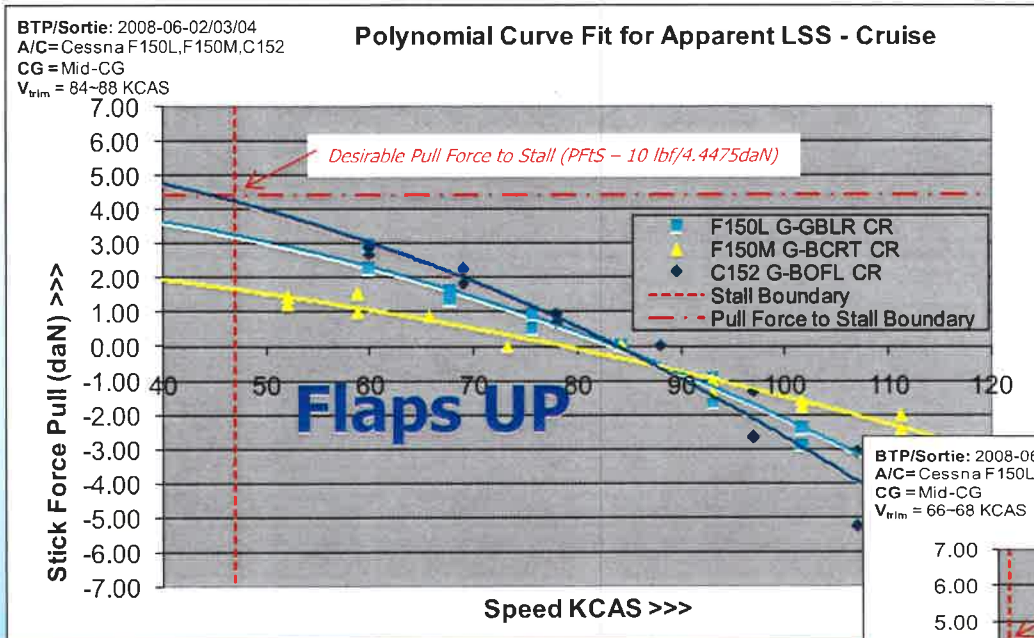


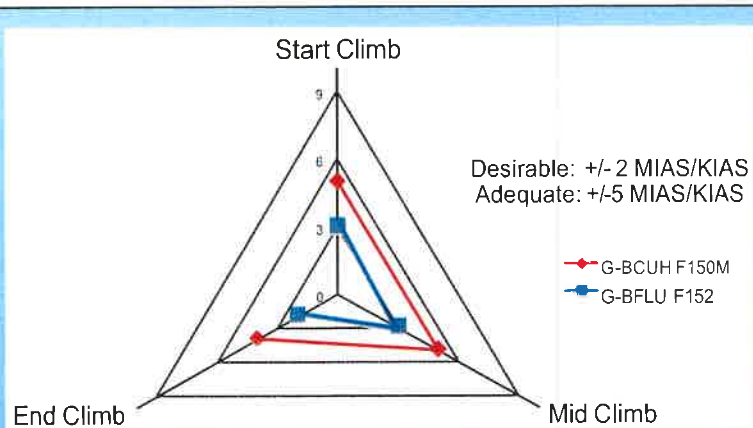
Mike Bromfield
Guy Gratton

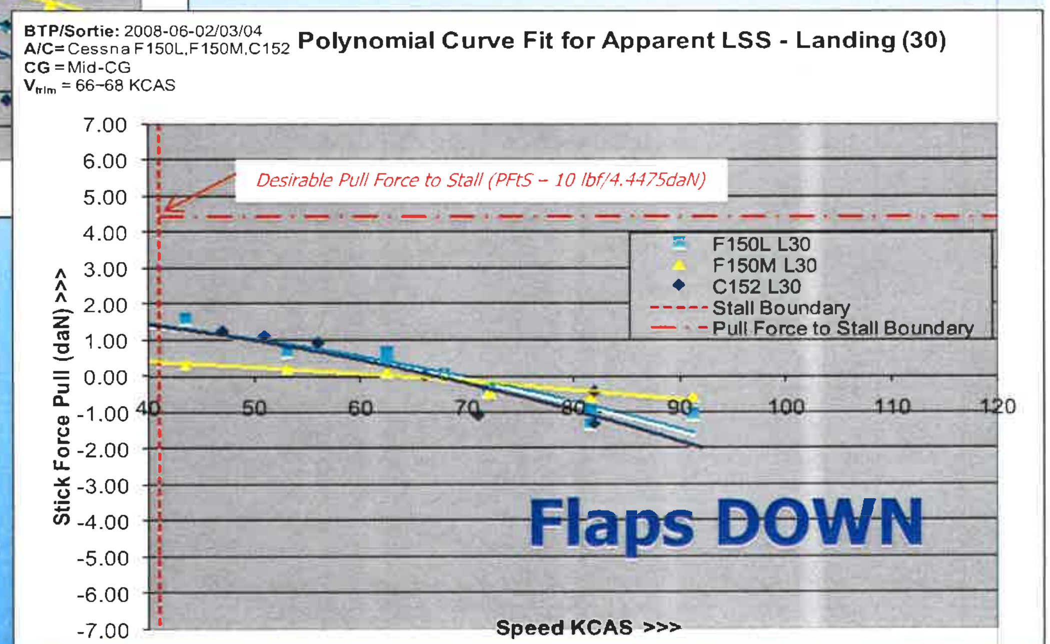
To the 41st SFTE International Symposium

Spot the difference…?

Stick Force to Change Airspeed
Cessna 150L,M & 152 with Flaps a) UP b) DOWN (L30)







http://www.pprune.org/flight-testing/484274-minimum-stick-force-gradient.html

Siai Marchetti 1019

"And here is the view of the elevator I took from the back seat while training the company pilot. Unaccelerated flight, trimmed to 60KIAS, flaps 60, and no more nose down pitch control available! (If you want to nose down, you have to either remove some flaps or some power)."

Load - CG diagram

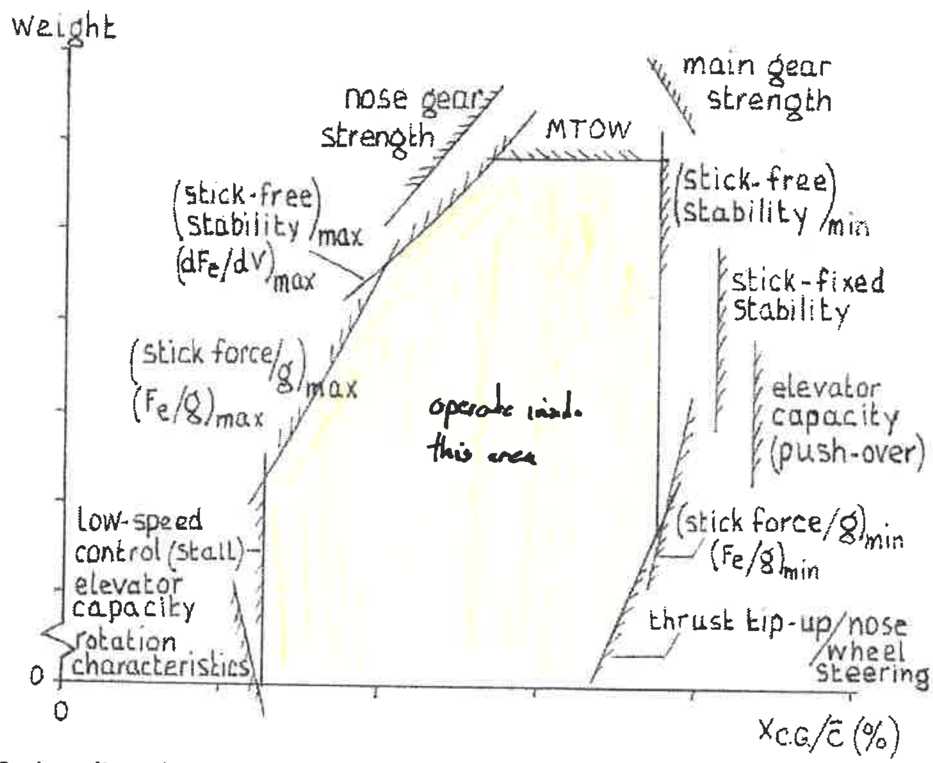
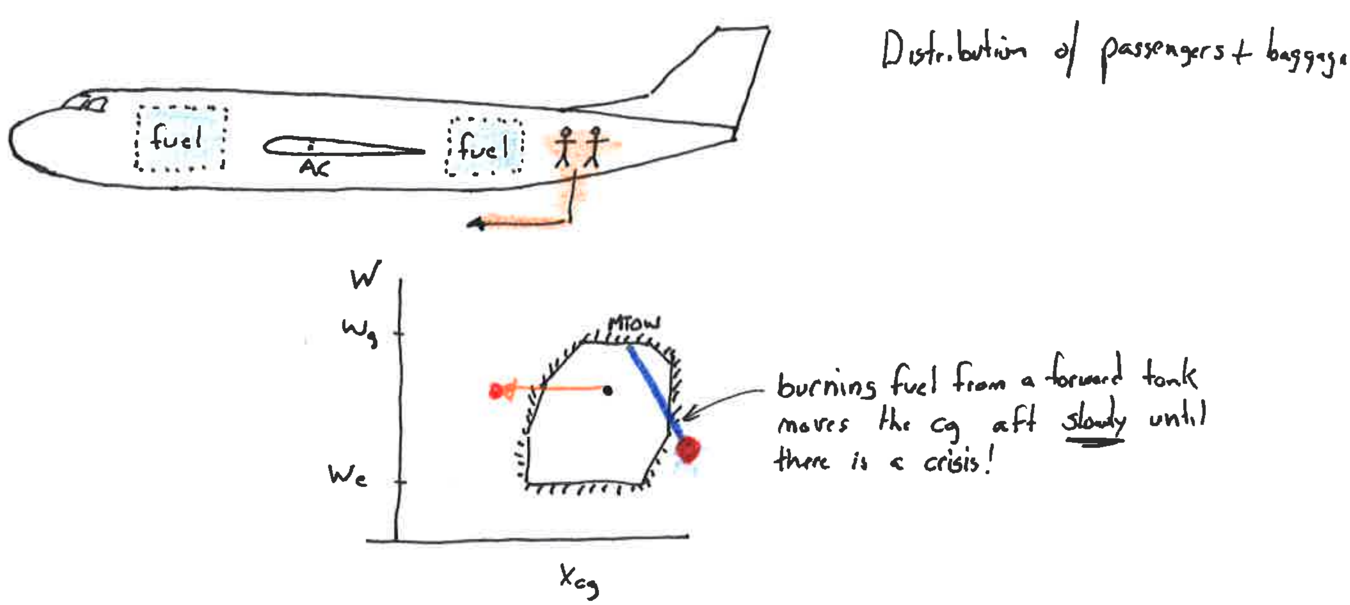


Figure 31.12 - Loading diagrams of some jet transport aircraft: Limits of the loading diagram.

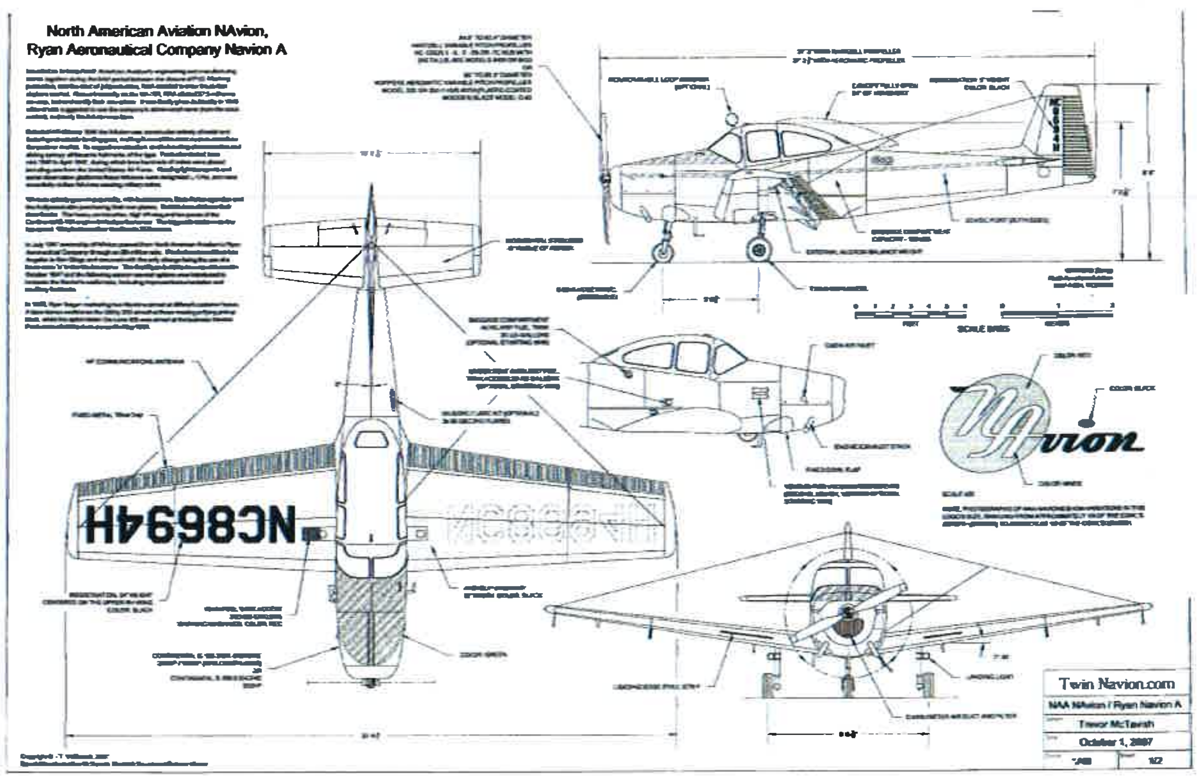
A common failure is to feed from the wrong fuel tank during flight.



North American Aviation: Navion

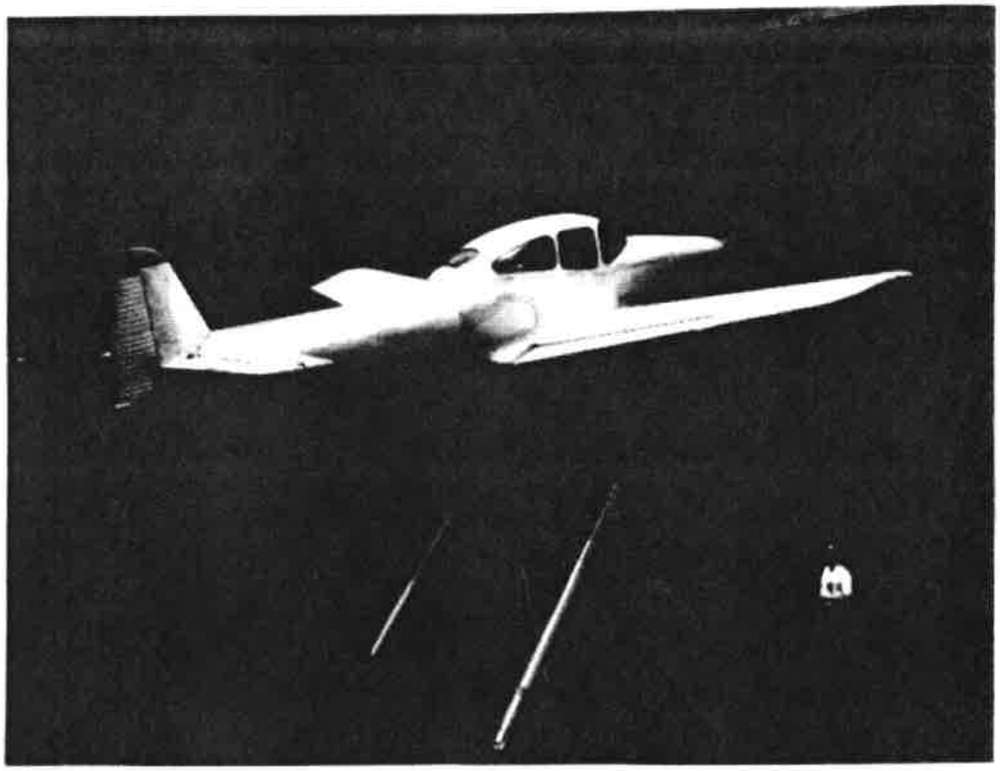
Between the P-51 and the F-86 (ca. 1950)

“General Aviation Airplane” in Appendix B1.



Full-scale wind tunnel (NASA Langley 30'x60')

- Electric motor (266 hp) driving propeller (Tc=T/qS)



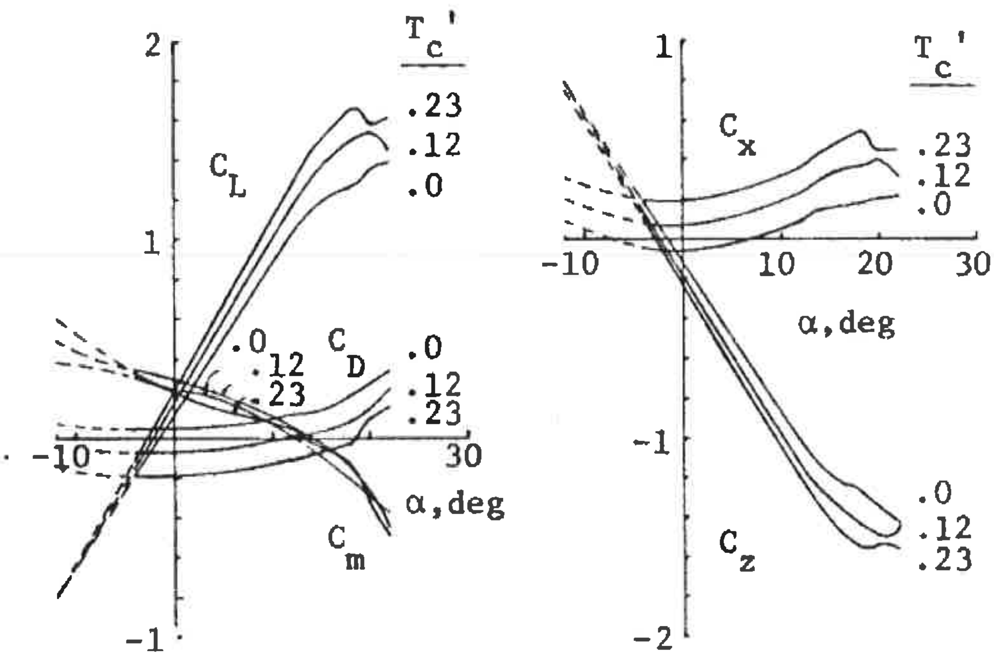


FIGURE 2. Static Longitudinal Coefficients from Full-Scale Wind Tunnel Tests.

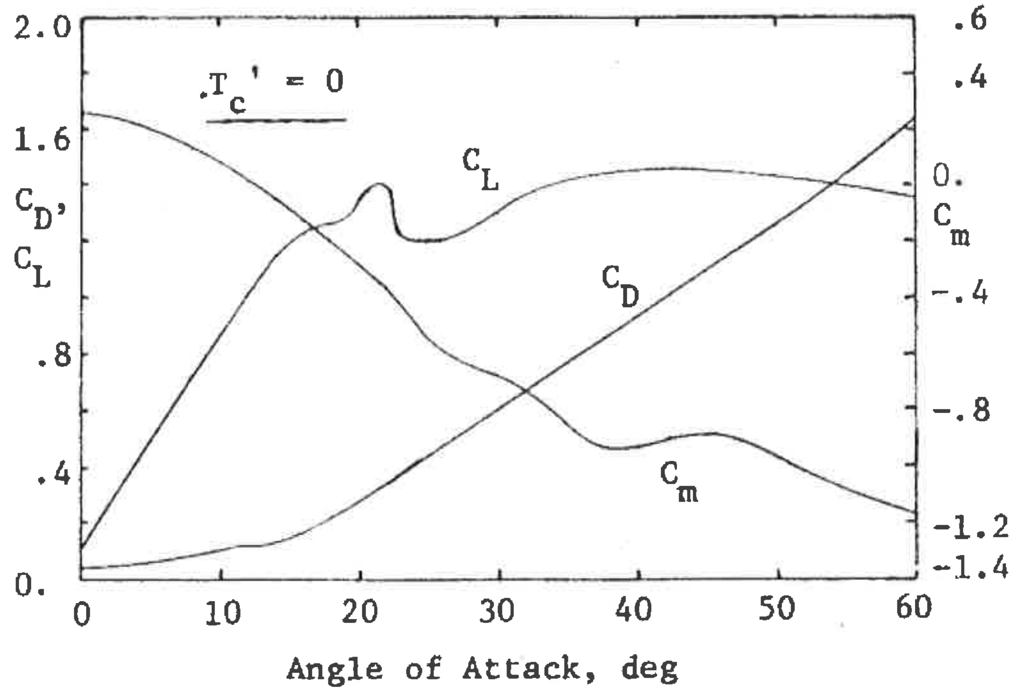


FIGURE 3. Combined Static Longitudinal Model (zero thrust).

The combined zero-thrust static longitudinal coefficients, illustrated in Fig. 3, have three characteristics of note. The wing stalling effects are confined principally to C_L, which shows a pre-stall flattening at $\alpha = 18°$ and a stall break at $\alpha = 22°$. C_D is quadratic in α for low angles, but it is linear in the 20°-to-50° range. The pitching moment slope changes sign for $\alpha = 38°$ to 45°, implying a region of local instability and the possibility of a superstall equilibrium beyond $\alpha = 45°$; however, neither problem can occur unless the aircraft can trim in the region. With the rotational center at 25% m.a.c. (as shown), the maximum ΔC_m that can be produced by the elevator is 0.54, far short of the 0.9 required for trim at high α. An aft center-of-gravity (c.g.) shift of more than 20% would be required to experience pitch instability or superstall with full negative δE; this would place the c.g. well behind its certified aft limit.

Stall strongly depends on power setting:

-Case A: full power & up elevator

-Case B: cut power & up elevator

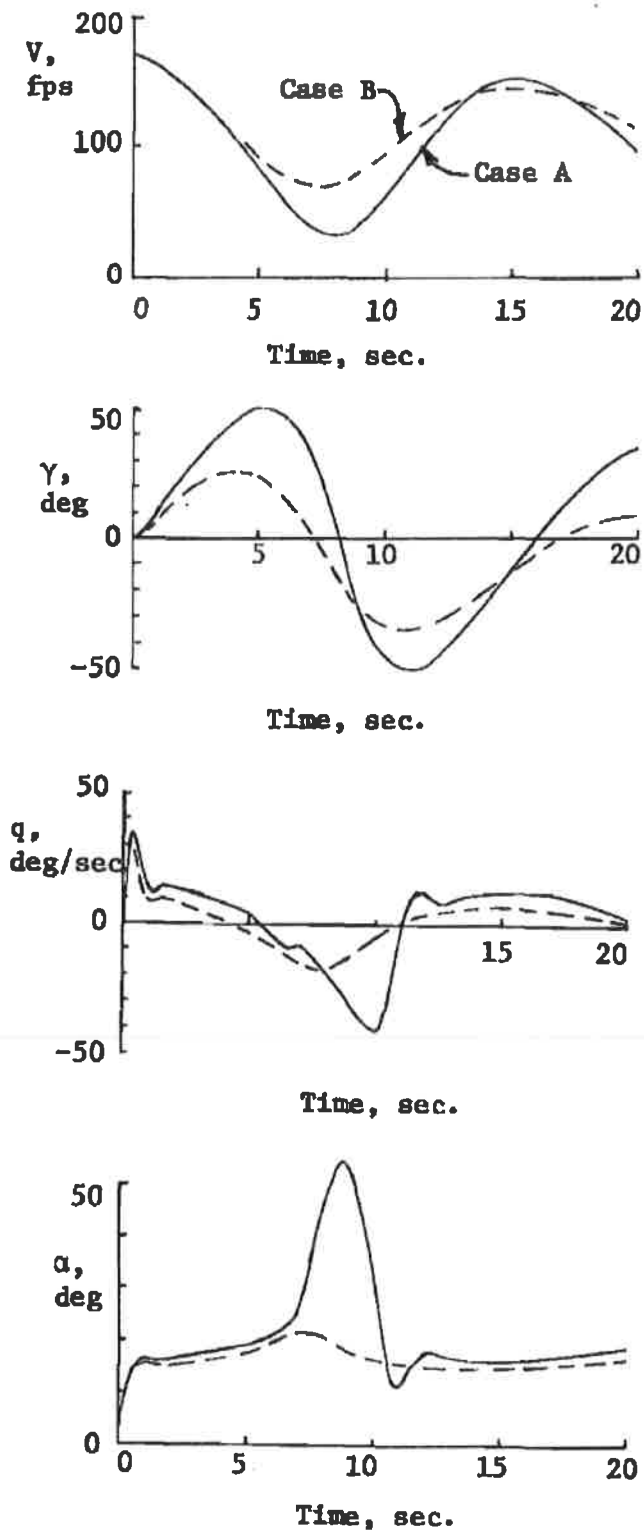


FIGURE 15. Effect of Power on the Abrupt Stall. Altitude = 2286 m(7500 ft).

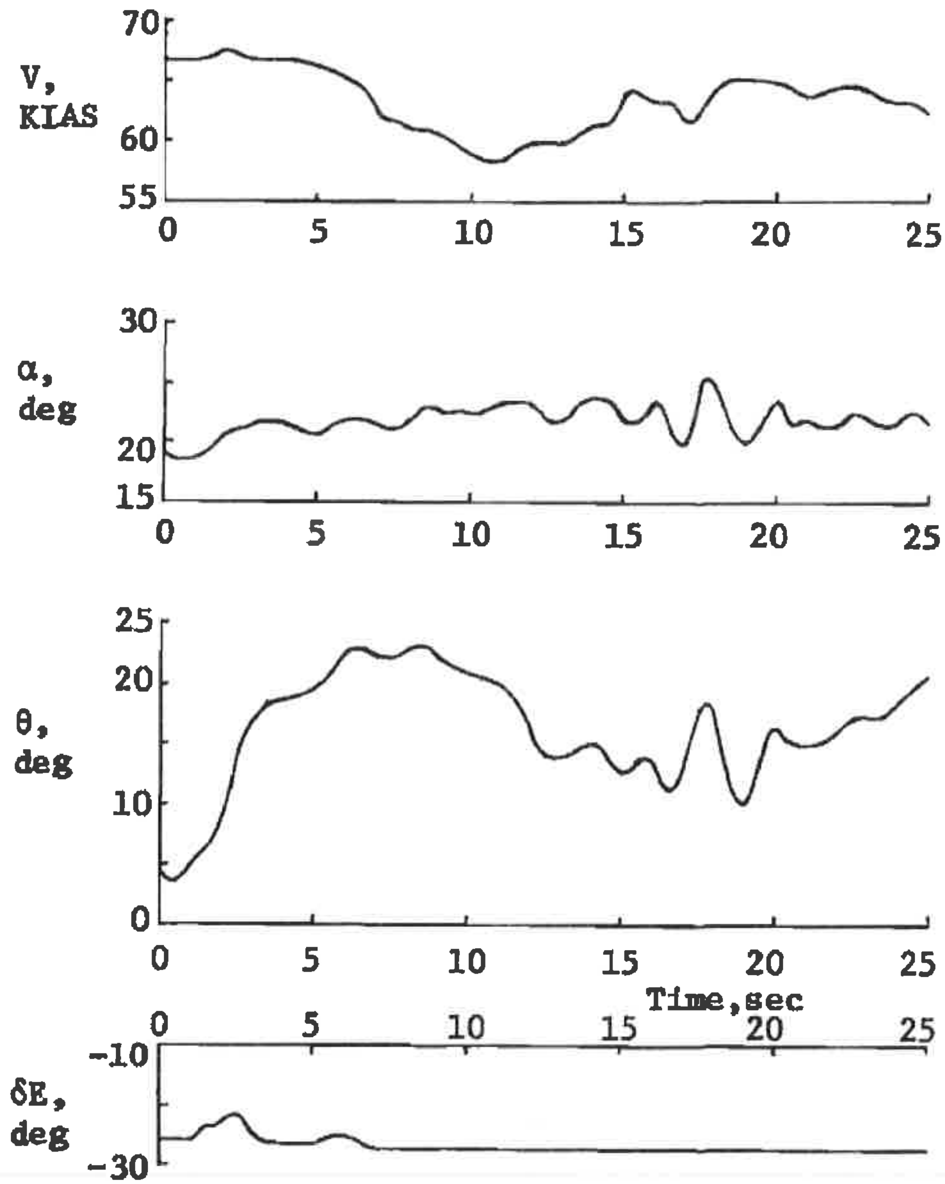


FIGURE 21. Effect of Adding Power During the Stall.

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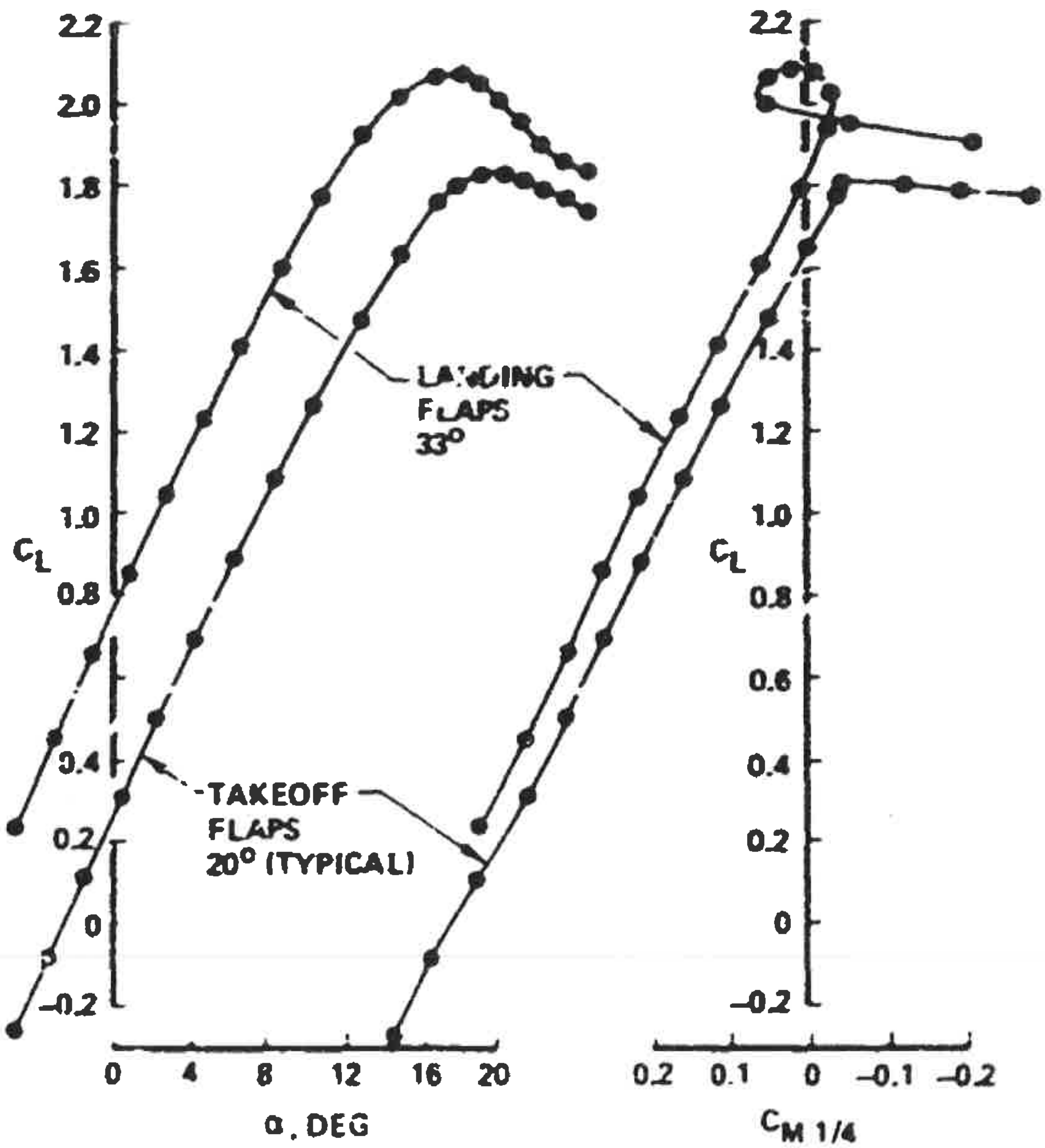
Flight Research Laboratory
Department of Mechanical and Aerospace Engineering
Princeton, New Jersey, U.S.A.

Boeing 747



FIGUPE 3 HIGH LIFT WIND TUNNEL DATA (LOW REYNOLDS NUMBER)

Fokker F-28

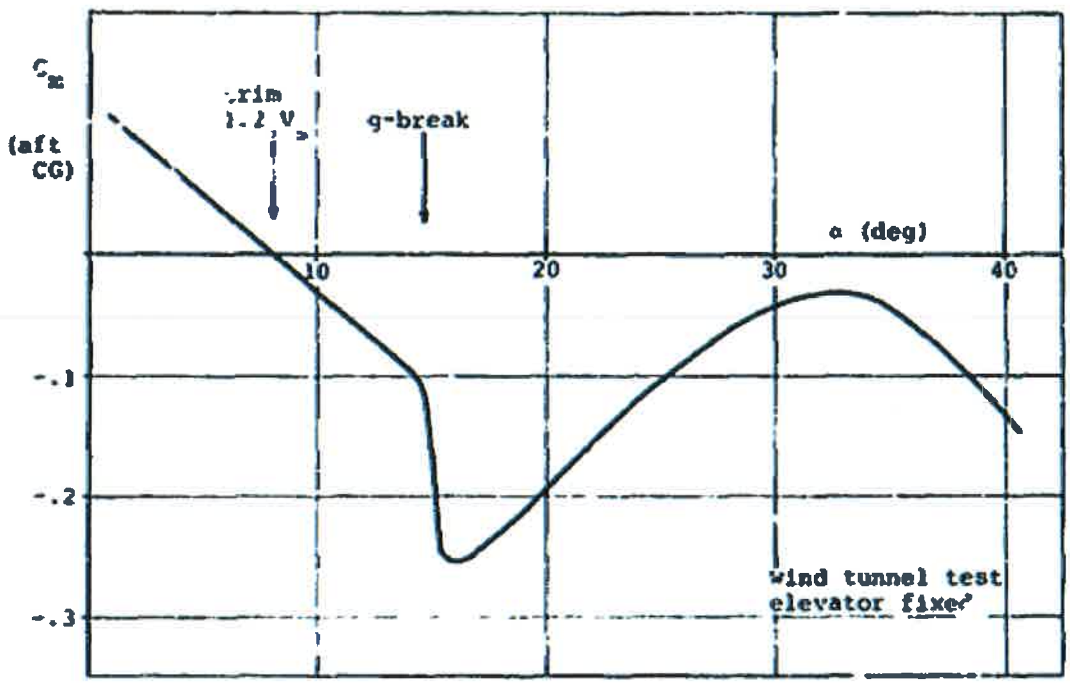


Fig. 3 F28 pitching moment characteristics

If the tailplane is immersed in the wake of the separated flow of the main wing, the aircraft will loose its longitudinal stability. It will remain unstable and pitch-up until a new equilibrium is found at a very high angle-of-attack. This may result in a "locked-in stall" or "deep stall", from which recovery may be extremely difficult. The principle of deep stall will be explained with the aid of figure 26.35.

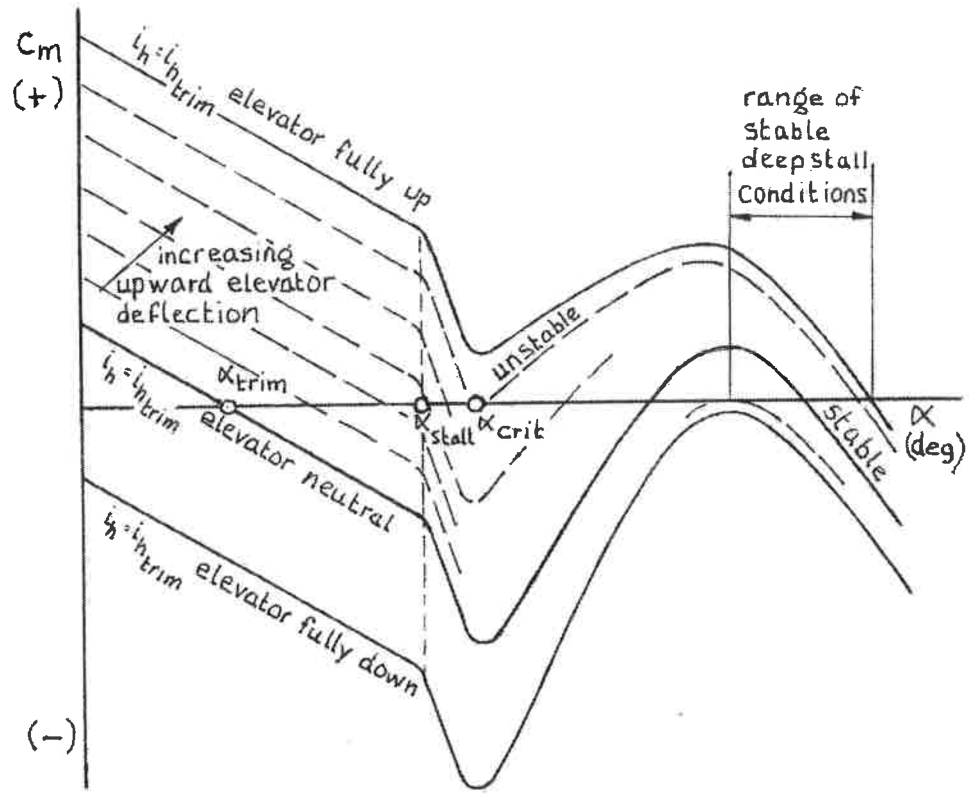


Figure 26.40 - Stability and controllability on an aircraft with a T-tail

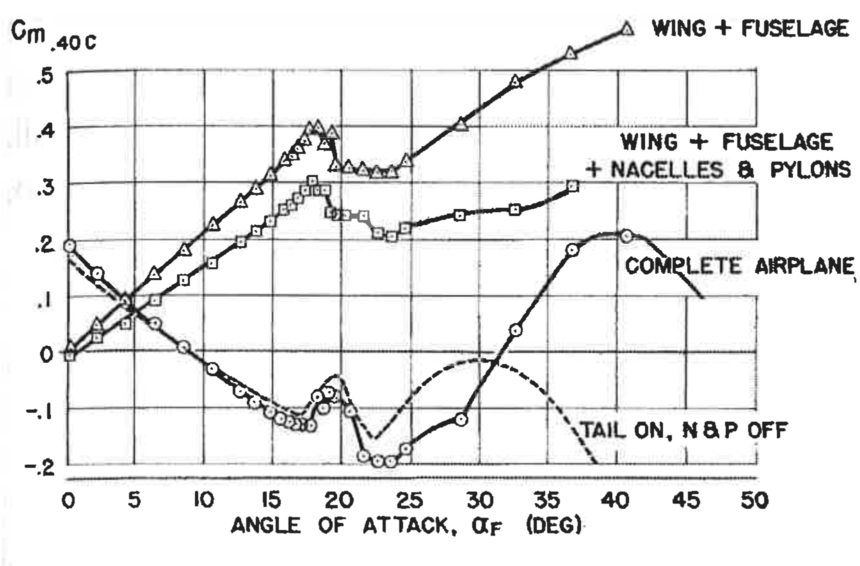


Figure 26.35 - Pitching moment buildup, flaps up, typical model with deep stall problems. Source : AIAA Paper No. 65-738