Aircraft Configuration Influences on Flight Performance and Operations

Guest Lecture: AEM 368
Notes: http://tiny.cc/AEM368-Config

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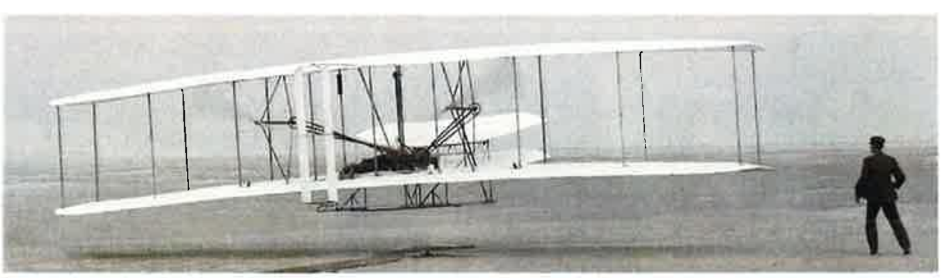
Why do these aircraft fly differently?
.... or do they?



Missions:

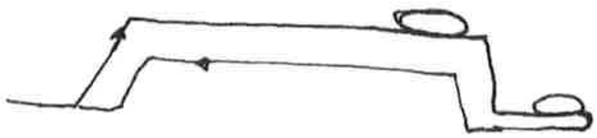
Aircraft perform missions to justify their existance

Commercial Aircraft / Cargo

Objective
Constraints

~~Cost~~ profit to carry # people from A → B

Military (fighter)



Win

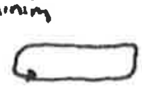
Military (ISR)

Fly to a location and surveil for a given time

Acrobatic

Controllable aircraft maneuvers ~~with~~ perfection of shapes

Training



. . . ?

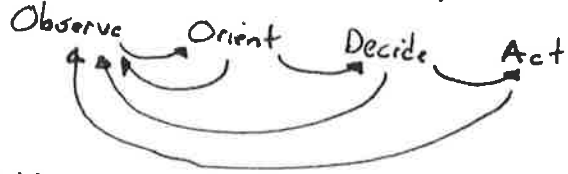
Energy Management

"When you maneuver an aircraft, you need energy... you lose energy either in gaining altitude, airspeed or both"

— Col. John Boyd

Notice how a pilot describes energy... never the less, this statement created a new concept of air combat from which the F-16 emerged.

The other main product from Boyd was the OODA loop



His insight was that the aircraft/pilot combination with the fastest OODA cycle ~~tends to win~~ wins.

The original program name for the F-16 was the lightweight fighter (LWF).

- The 1960s USAF trend was for heavier aircraft with high (M>2) capabilities. This path eventually led to the F-15.
- The F-16 took a different strategy. Lower top speeds (M<2) and much lighter.

Fact: Most air combat occurs below Mach 1.

Consider 3 representative maneuvers and a size factor

		want
• Sustained turn rate		high
• Top speed	$\longrightarrow$ vs $\rightarrow$	high ↑
• Climb rate	↗	high
• Size / Cost	 vs 	low

• Sustained turn

$$\omega_{max} = g\sqrt{\frac{\rho}{W/S}\left(T/W\right)\frac{1}{2k} - \underbrace{\sqrt{\frac{C_{D_0}}{K}}}_{\approx 1}} \approx g\sqrt{\frac{\rho}{(W/S)}\left(\frac{T}{W}\right)\frac{\pi A R e}{2} - 1}$$

$$\text{at } V = \sqrt{\frac{2}{\rho}\frac{W}{S}}\,\sqrt[4]{\frac{K}{C_{D_0}}} \approx \sqrt{\frac{2}{\rho}\frac{W}{S}}$$

• high T/W
• low W/S
• high AR

• Top Speed

$$V_{max} = \sqrt{\frac{\frac{T}{W}\frac{W}{S} + \frac{W}{S}\sqrt{\left(\frac{T}{W}\right)^2 - 4C_{D_0}K}}{\rho C_{D_0}}} \approx \sqrt{\left(\frac{T}{W}\right)\left(\frac{W}{S}\right)\frac{2}{\rho C_{D_0}}}$$

• high T/W
• high W/S
• low C_{D_0}

• Climb rate

$$\dot{h}_{roc} = ROC = \sqrt{\left(\frac{W}{S}\right)\left(\frac{2}{3}\right)\frac{1}{\rho C_{D_0}}}\left(\frac{T}{W}\right)^{3/2}\left(1 - \frac{Z}{6} - \frac{3}{2\left(\frac{T}{W}\right)^2\left(\frac{L}{D}\right)_{max}^2 \cdot Z}\right)$$

$$Z = 1 + \sqrt{1 + \frac{3}{(L/D)^2\left(\frac{T}{W}\right)^2}} \approx 2$$

$$= \sqrt{\left(\frac{W}{S}\right)\left(\frac{2}{3}\right)\frac{1}{\rho C_{D_0}}}\left(\frac{T}{W}\right)^{3/2}\left(1 - \frac{1}{3} - 0\right) \quad \text{at } V_{roc} \approx \sqrt{\left(\frac{T}{W}\right)\left(\frac{W}{S}\right)\left(\frac{2}{3}\right)\frac{1}{\rho C_{D_0}}}$$

• ~~[illegible]~~ moderate W/S
• high (T/W)

• Size

Size $\approx S$

$\approx T$ since $T = \dot{m}\,\Delta V = \rho V A \Delta V$

• low S
• low T

Match $\underbrace{\text{sustained turn Velocity}}$ with $\underbrace{\text{Climb rate Velocity}}$

$$V_{\omega} = \sqrt{\frac{2}{\rho}\frac{W}{S}}\sqrt[4]{\frac{K}{C_{D_0}}} \approx \sqrt{\frac{2}{\rho}\frac{W}{S}} \qquad V_{\dot{h}} \approx \sqrt{\left(\frac{T}{W}\right)\left(\frac{W}{S}\right)\left(\frac{2}{3}\right)\frac{1}{\rho C_{D_0}}}$$

$$\frac{V_{\dot{h}}}{V_{\omega}} = \sqrt{\left(\frac{T}{W}\right)\left(\frac{W}{S}\right)\left(\frac{2}{3}\right)\frac{1}{\rho C_{D_0}} \cdot \frac{\rho S}{2W}}$$

$$= \sqrt{\frac{T}{W}\frac{1}{3}\frac{1}{C_{D_0}}} \approx \sqrt{\frac{1}{3}\cdot\frac{1}{0.02}} = \sqrt{\frac{1}{0.06}} \approx 4!$$

If $T/W \approx 1$, then a relatively high C_{D_0} is ok.

If top speed is not as critical, C_{D_0} can be larger and W/S can be lower.

Conclusion:

A lightweight fighter has turn rate advantages.

has T/W advantages ($W\downarrow$ and simple engine inlet $T\uparrow$ for $M<1$

has less visible/radar area

has a slower top speed

is much less expensive

This is the F-16's description

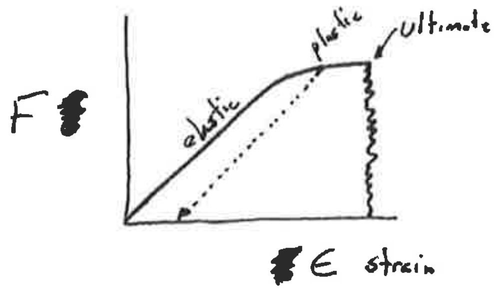
V-n diagram "Flight Envelope"

Where aerodynamics and structures interact.

From structures, you design an aircraft to meet a particular load factor

e.g. $n^+ = 5 \quad n^- = 3$ at a particular weight

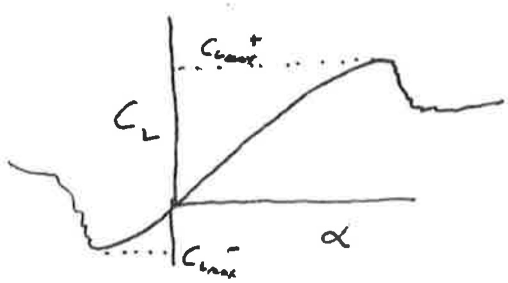
You design a buffer region between the load limit (plastic) and the ultimate load (break



From aerodynamics:

$$L = \frac{1}{2}\rho V^2 C_L S \quad \Rightarrow \quad n = \frac{L}{W} = \frac{1}{2}\rho V^2 C_L \left(\frac{W}{S}\right)^{-1}$$

And a maximum C_L is determined by aero



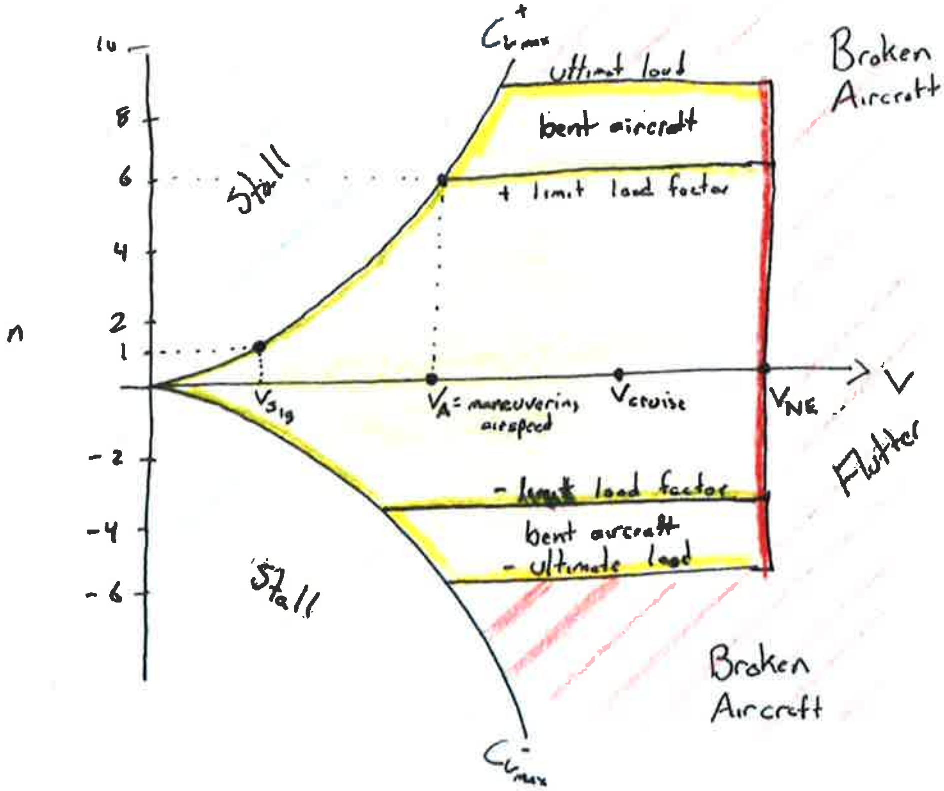
$$\Rightarrow \quad n_{max} = \frac{1}{2}\rho V^2 C_{L_{max}} \left(\frac{W}{S}\right)^{-1}$$

From aerodynamics + performance + aero-structural dynamics

- The aircraft has a maximum "red-line" airspeed. V_{NE}
  Above this airspeed, parts may not remain on the aircraft!

- ~~Certain~~ All aircraft will exhibit "flutter" above a certain dynamic pressure at certain flight conditions

V-n



$V_{s_{1g}}$ is the stall speed at $n=1$

V_A = maneuvering speed, the intersection of aero and structural limits.

At V_A, the aircraft can not be broken/bent by "normal" acceleration (e_n).

At V_A, the aircraft **can** be broken/bent by control deflections and non e_n accelerati

V_C = Cruise speed (FAA certification)

V_{NE} = Never exceed.

The aircraft can not sustain operation outside of the stall $C_{L_{max}}$ curves.

Load - CG diagram

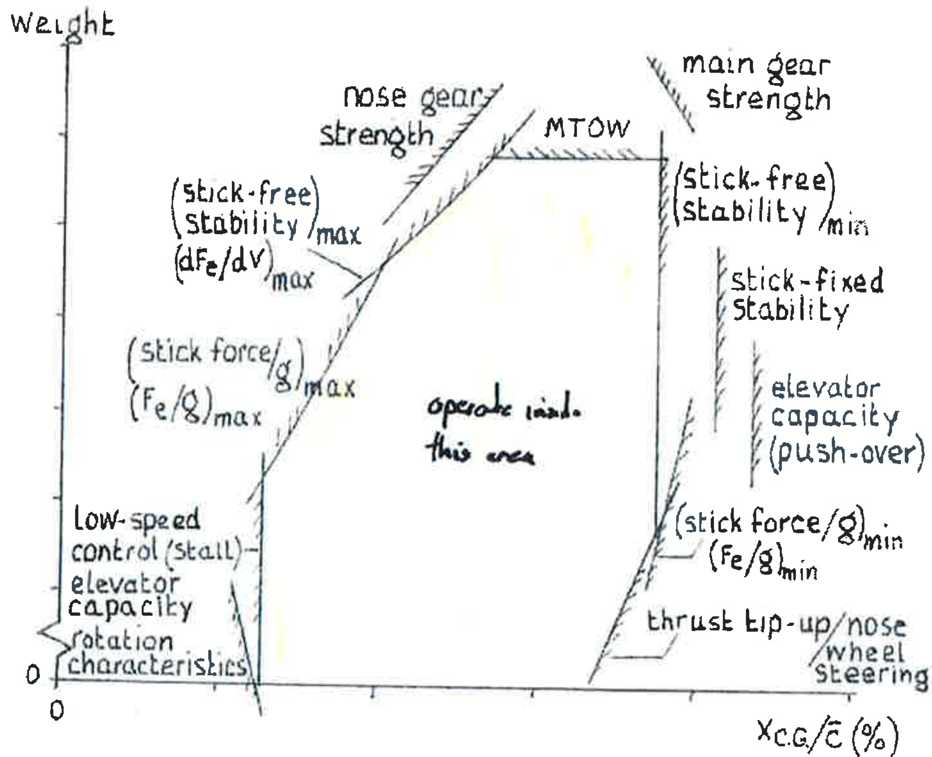
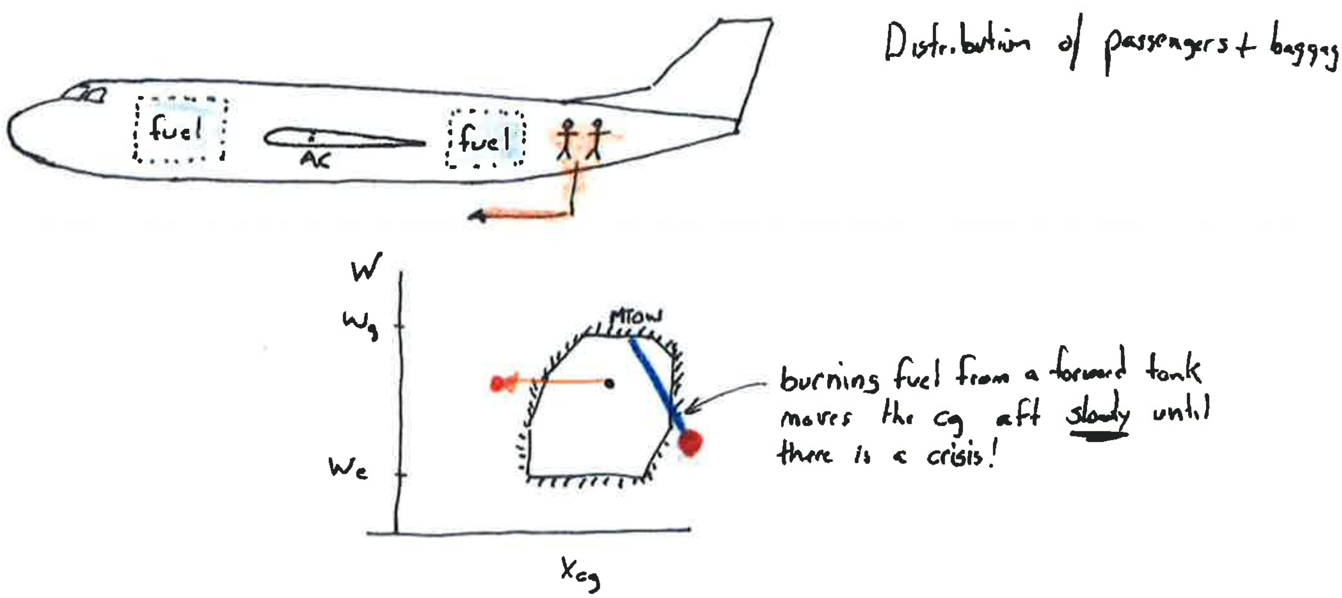


Figure 31.12 - Loading diagrams of some jet transport aircraft: Limits of the loading diagram.

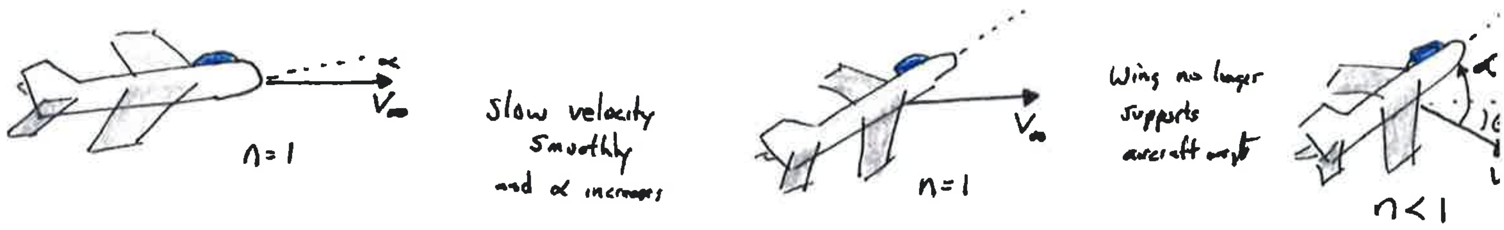
A common failure is to feed from the wrong fuel tank during flight.



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

The lift coefficient is

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



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

How slowly should stall be approached? <u>FAA says $-1\ \frac{kt}{s}$</u>

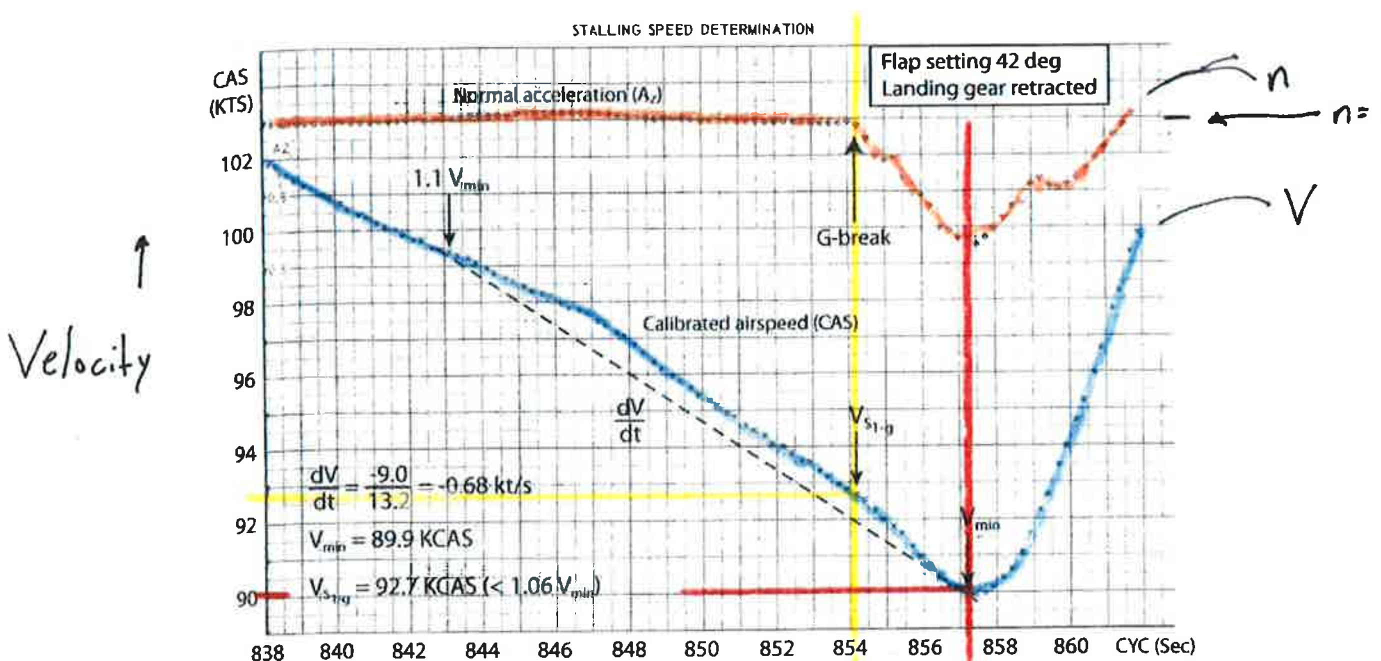


Figure 26.2 - Stall speed determination of a Fokker F-28 Mk 4000.
Source: Fokker Report H-28.40-27.001

Time →

Source: Aerodynamic Design of Transport Aircraft
Obert

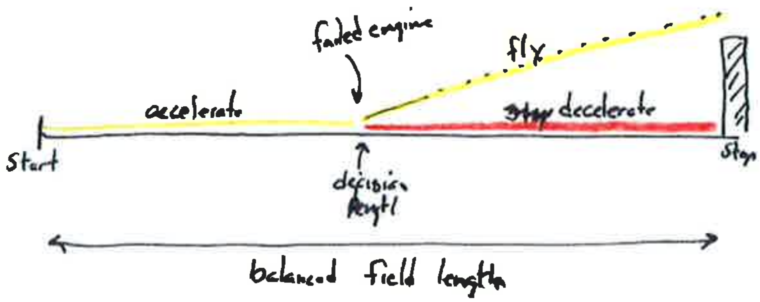
Balanced Field Length

Given an aircraft and an obstacle, the balanced field length is where

- The length necessary to accelerate and clear the obstacle (one engine failed)

equals

- The length necessary to accelerate to the decision speed and decelerate to a stop



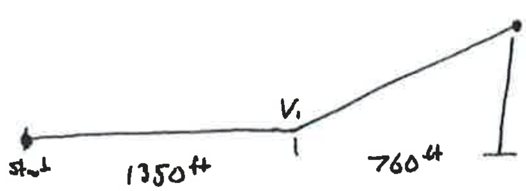
$\boxed{\text{The concept of BFL gives the pilot survivable options.}}$

The legal requirements vary depending the type and operation of the aircraft.

ie. ρ
μ_r
T
W
$\vdots$

Ex:
Estimate the BFL of a Mustang with a 50 foot obstacle with no failed engine

Fly



$1350 + 760 = 2100$ ft

Decelerate at V_1

$1350 + 1121 = 2471$ ft

with both engines operational, BFL = 2471

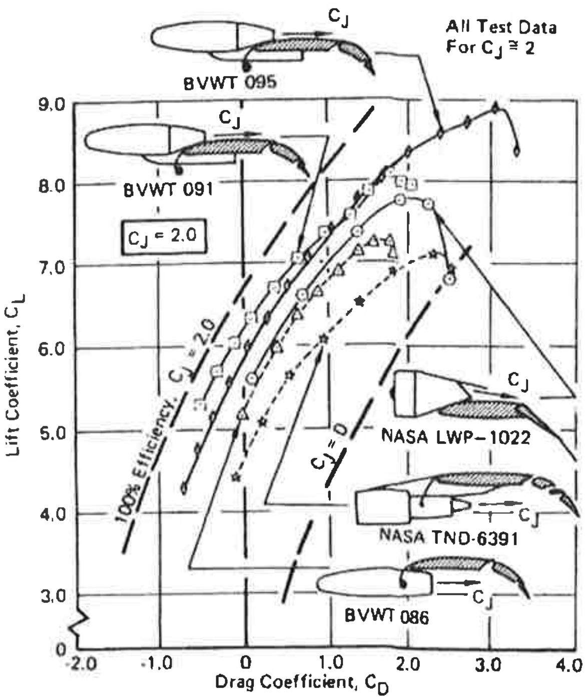
What other cases need to be evaluated?

YC14 STOL TO/Landing

$$S_{to} \approx \frac{1}{\rho}\left(\frac{W}{S}\right)\underbrace{\frac{1}{C_{L_{to}}}}\ \frac{1}{g}\ \frac{1}{K_T}$$

Upper surface blowing:

$C_{L_{max}} \approx 9$



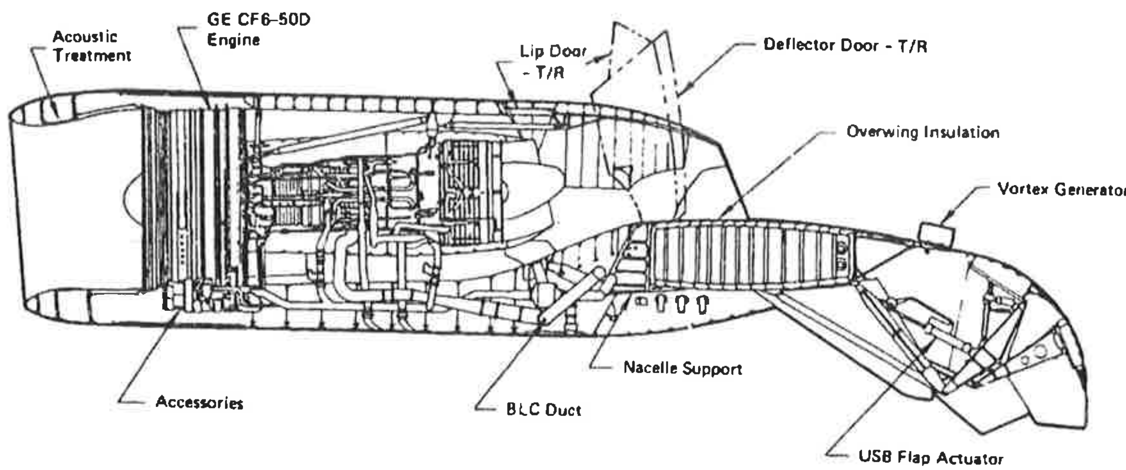


Fig. 21 Engine and nacelle cross section. Reprinted with permission from AIAA Preprint AIAA-74-972 © 1974.

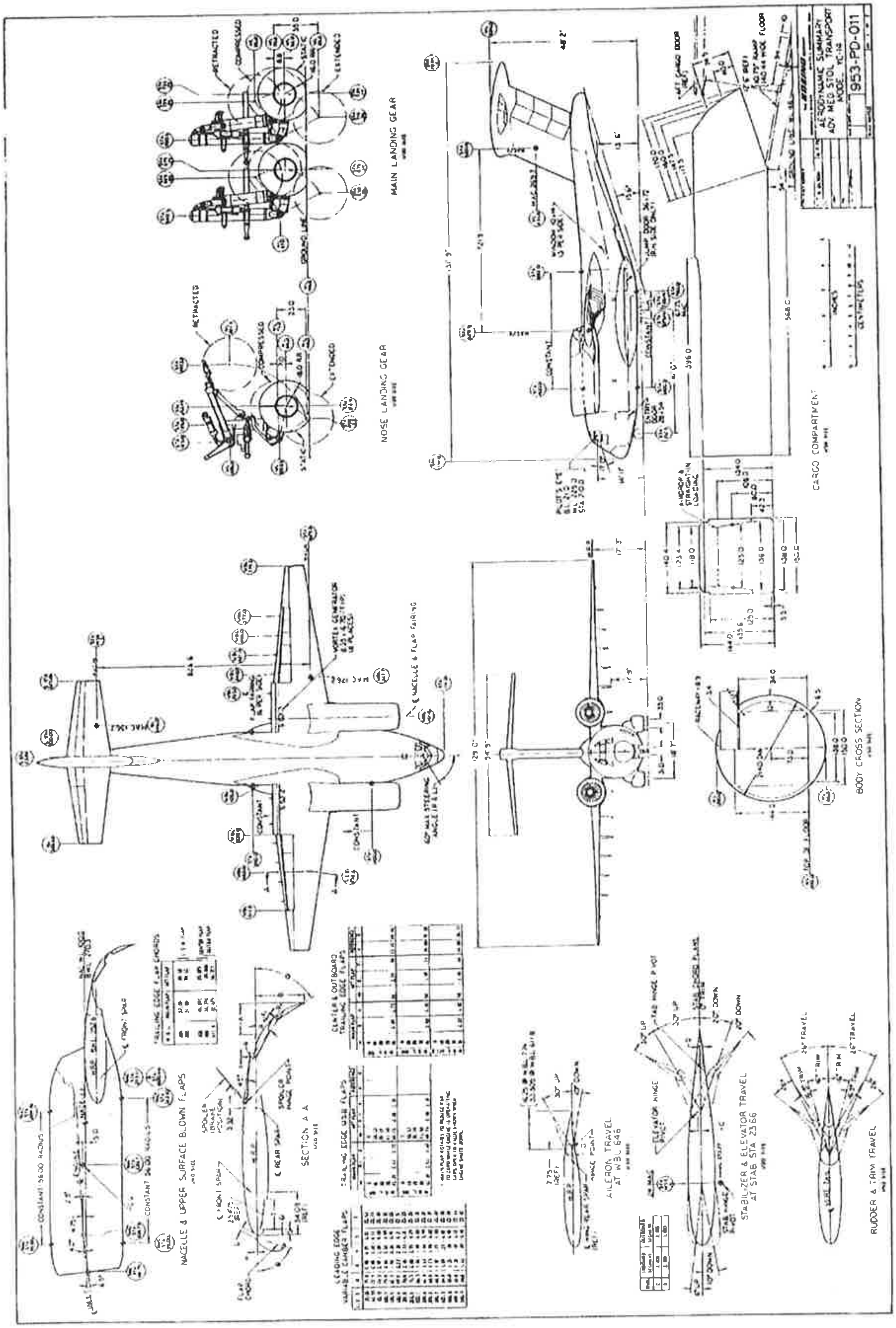
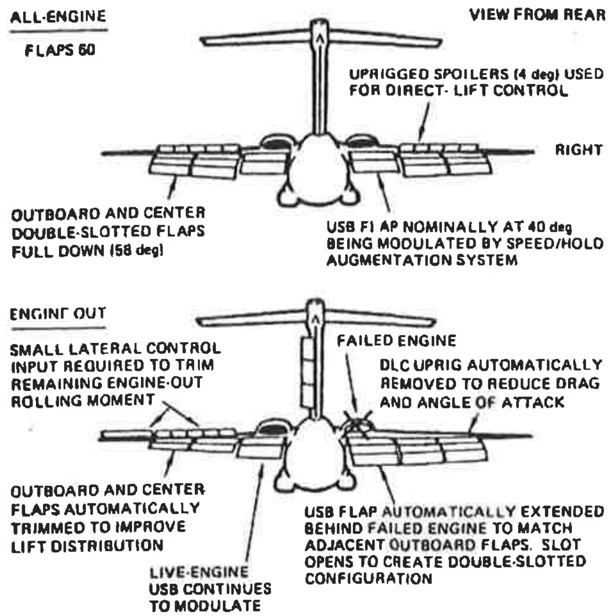


Fig. 25 YC-14 configuration details.

YC-14 Flight Control System.

What happens if an engine fails?



Approach at 80 kts "Like a big Cessna 182"

tiny.cc/aem368YC14

VJ101

In the 1960s, the German Airforce had a problem. The front line fighters required dedicated airfields with long runways. (Ex. F104 liftoff at $\approx$190 kts)

Just a few nuclear blasts could knock out their entire airforce. Even a few carefully placed conventional bombs could prevent T/O. (See: [redacted] Img 91).

Solutions:

- Disperse and Decentralize

↓

- VTOL

Q: How do you design a front line air superiority fighter with F104 performance in a VTOL configuration?

A: F-35 is still having deployment issues using modern materials. (present day)

A: F104 "Zell" ZLL (Zero length launch) .... Landing?

A: Tailsitters

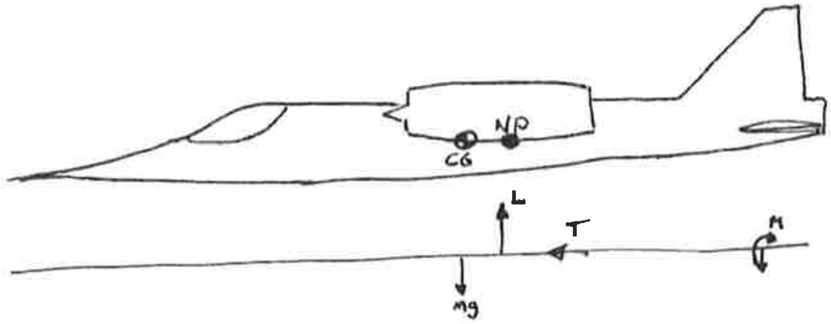
problem:
- Sears-Haack says "long and slender" for high Mach aircraft.
- Landing stability of tailsitters says "wide and short".

A: Vectored Thrust

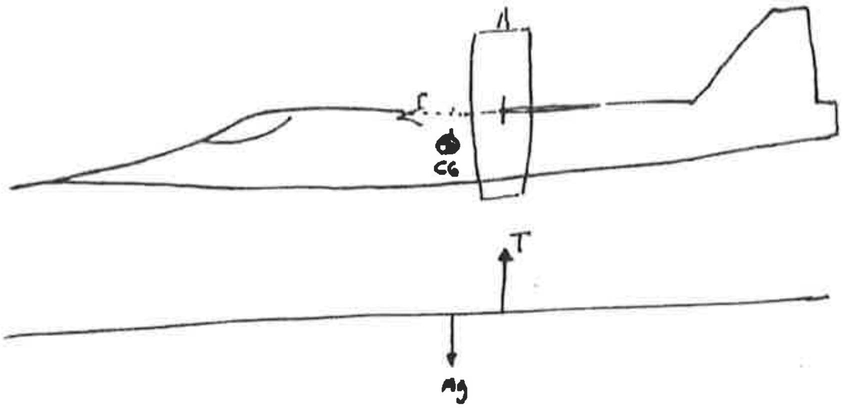
VJ 101: tilt the cruise engines

VAK 191B: Nozzle vectored thrust (eg. Harrier)

Forces and Moments (Conceptual VTOL system)

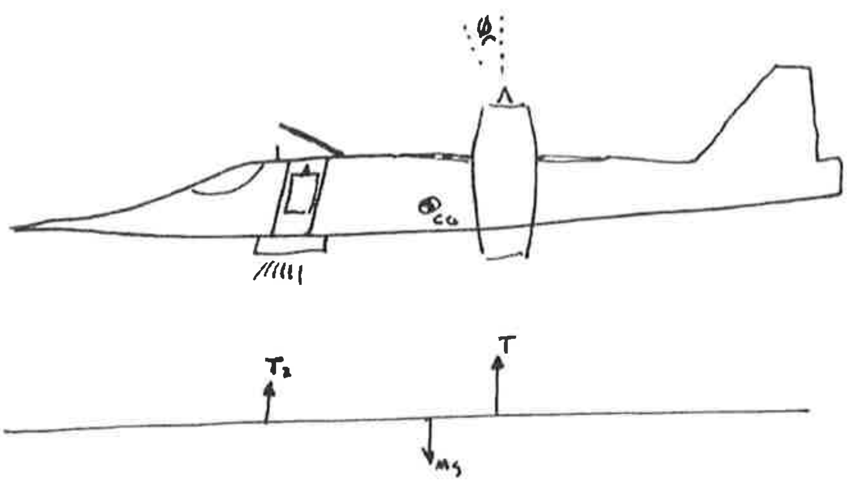


Stable

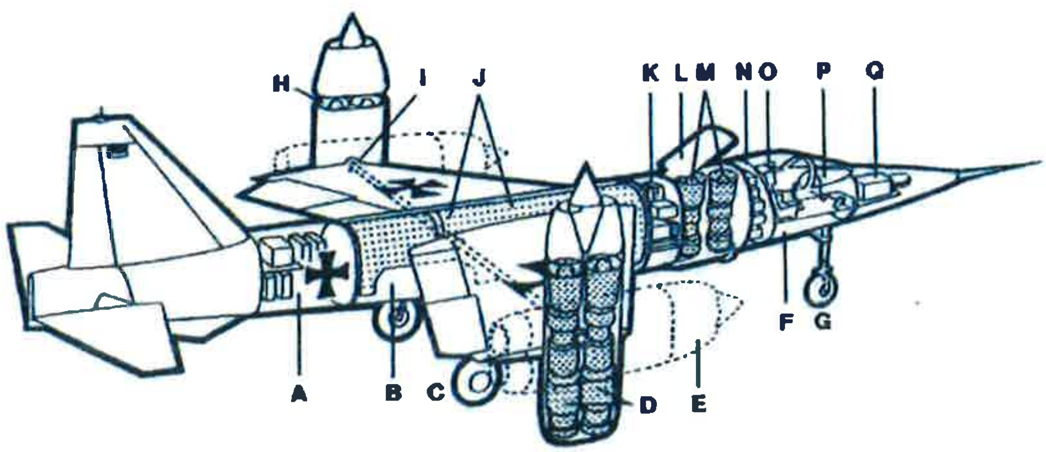


not balanced

- We clearly need an additional moment (Nose Up). Ahead of the CG would be most efficient. Add lift engines ahead of CG.



- pitch control with lift engines
- roll control with wingtip (cruise) engines
- yaw control with differential tilt angle of wingtip cruise engines

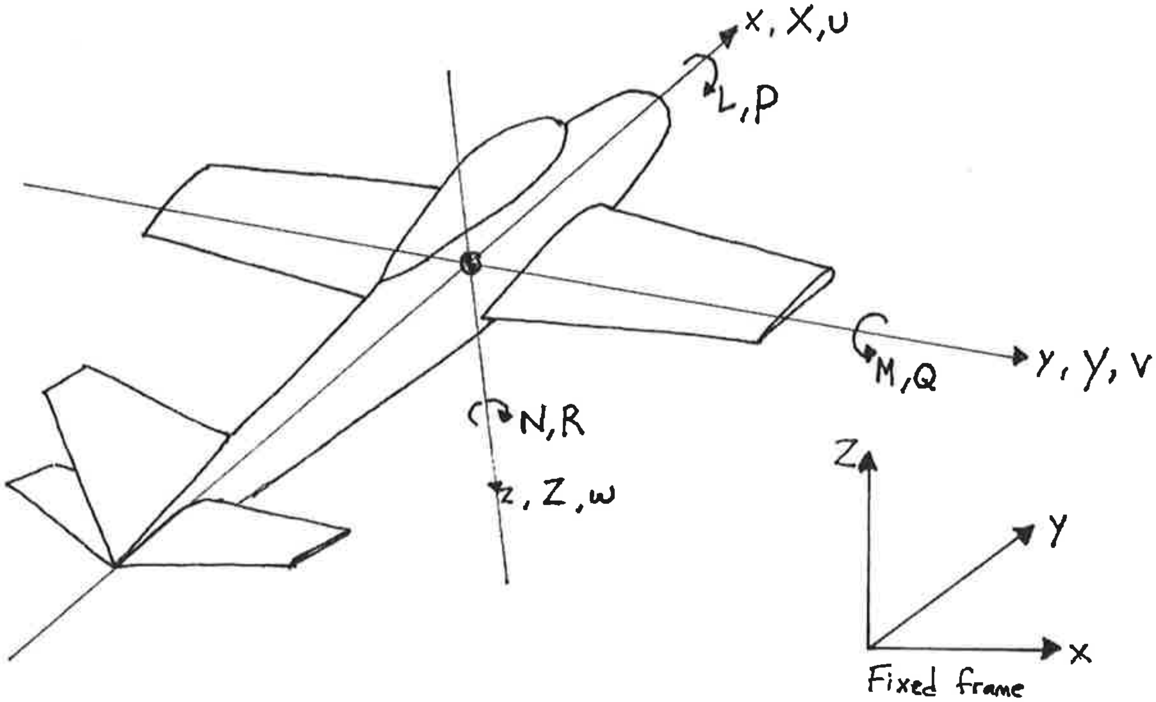


A. Equipment bay
B. Main landing gear bay
C. Rearward retracting main wheels
D. Afterburning RB.145 turbojets in swivelling nacelles
E. Nacelle in forward position
F. Nosewheel bay
G. Rearward retracting nosewheel
H. Nose section of nacelle raised for V/STOL flight
I. Hollow shaft on which nacelles swivel
J. Two-cell fuselage tank
K. Avionics bay
L. Retractable air intake door for lift engines
M. RB.145 lift engines
N. Avionics bay
O. Pilot ejection seat
P. Instrument panel
Q. Nose radar installation (planned)

Fig. 10 Cutaway drawing of the afterburner-equipped VJ 101 X2.

Flight Dynamics

Aircraft Coordinate System



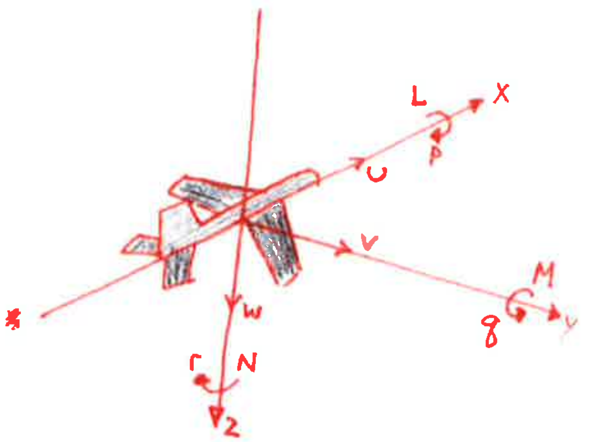
x, y, z	Aircraft "stability" frame location
X, Y, Z	Forces in stability frame
u, v, w	Velocity in stability frame
L, M, N	moment in stability frame
P, Q, R	Angular velocities (roll, pitch, yaw)
ϕ, θ, ψ	Euler angles (orientation)

Warning:

Outside of aerodynamics, the x axis is usually down the length of the a/c.

$$X_{loft} = -X_{aero\ stability}$$
$$Y_{loft} = Y_{aero\ stability}$$
$$Z_{loft} = -Z_{aero\ stability}$$

Nomenclature / Coordinates (Review)



Body fixed stability frame:

yaw
- Z points <u>down</u>
- N moment in z direction (cw +)
- r angular rate in z direction

roll
- X points forward
- L moment in x-direction (cw +)
- P angular rate in x direction

pitch
- y point off right starboard wing
- M moment in y direction (nose up +)
- q angular rate

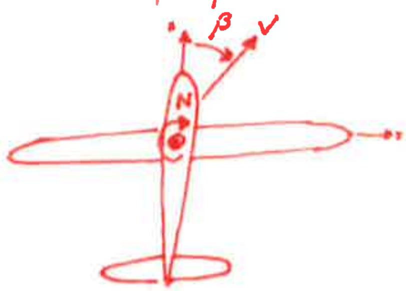
$$\beta = \operatorname{asin}\left(\frac{V}{|V|}\right)$$

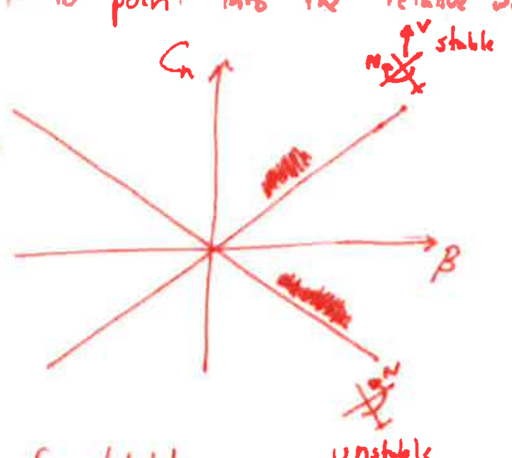
$$= \operatorname{asin}\left(\frac{V}{\sqrt{U^2+V^2+W^2}}\right)$$

$$\approx \frac{V}{U} \text{ in radians if } V \ll U$$

Static directional Stability

The ability of the aircraft to point into the relative wind.



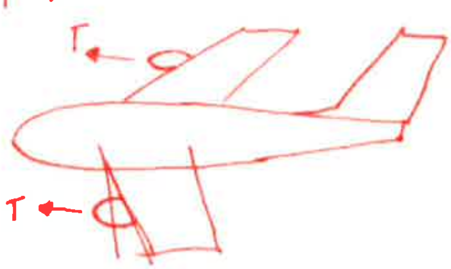


$$C_n = \frac{N}{q S b}$$

↑ notice wing span

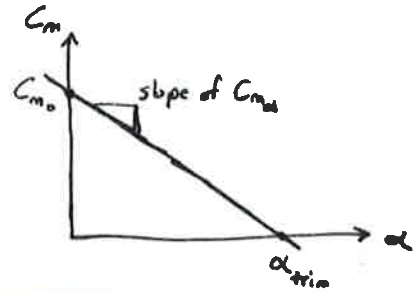
$$C_{n_\beta} \equiv \frac{d C_n}{d\beta} > 0 \text{ for stability}$$

Major players



- Vertical Stabilizer
- Fuselage
- Offset thrust
- Wing (small to negligible at $\alpha \ll \alpha_{stall}$)
  exception adverse yaw

Wing - Tail Static Stability



Moment about CG

$$C_{m_0} = \underbrace{C_{m_{ac_w}} + C_{L_{0_w}}\left(\frac{x_{cg}}{\bar{c}} - \frac{x_{ac}}{\bar{c}}\right)}_{\text{wing}} + \underbrace{\eta V_H C_{L_{\alpha_t}}\left(\epsilon + i_w - i_t\right)}_{\text{tail}}$$

$$C_{m_\alpha} = \underbrace{C_{L_{\alpha_w}}\left(\frac{x_{cg}}{\bar{c}} - \frac{x_{ac}}{\bar{c}}\right)}_{\text{wing}} \underbrace{- \eta V_H C_{L_{\alpha_t}}\left(1 - \frac{d\epsilon}{d\alpha}\right)}_{\text{tail}}$$

with $V_H = \frac{l_t}{\bar{c}}\frac{S_t}{S_w}$

Trim α

$y = ax + b \Rightarrow C_m = C_{m_\alpha}\alpha + C_{m_0}$

$\Downarrow$ $\qquad\qquad\qquad\qquad$ $\Downarrow$

$0 = ax + b$ $\qquad\qquad$ $0 = C_{m_\alpha}\alpha_{trim} + C_{m_0}$

$\Downarrow$ $\qquad\qquad\qquad\qquad$ $\Downarrow$

$x = -\frac{b}{a}$

$$\boxed{\alpha_{trim} = -\frac{C_{m_0}}{C_{m_\alpha}}}$$

$$\alpha_{trim} = -\frac{C_{m_{ac_w}} + C_{L_{0_w}}\left(\frac{x_{cg}}{\bar{c}} - \frac{x_{ac}}{\bar{c}}\right) + \eta V_H C_{L_{\alpha_t}}\left(\epsilon + i_w - i_t\right)}{C_{L_{\alpha_w}}\left(\frac{x_{cg}}{\bar{c}} - \frac{x_{ac}}{\bar{c}}\right) - \eta V_H C_{L_{\alpha_t}}\left(1 - \frac{d\epsilon}{d\alpha}\right)}$$

$+ \Rightarrow$ TED / LEU

$- \Rightarrow$ TEU / LED

Q: How does increasing i_t (the tail incidence angle) affect trim

A: "+" i_t (TED) <u>lowers</u> α_{trim}

Q: Why? Why does the wing influence the tail?

Why does the wing influence σ and $\frac{d\sigma}{d\beta}$

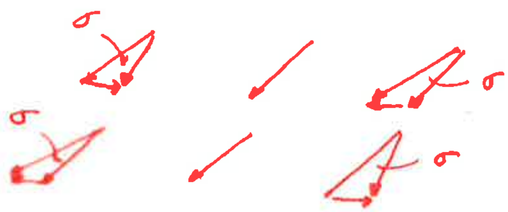
A: Aerodynamics of a wing generating lift

The shed vorticity is

$$\gamma(y) = -\frac{d\Gamma}{dy}$$

So the flow field (cross section) looks like

If we add the free stream to this, there is a sidewash velocity with an angle σ



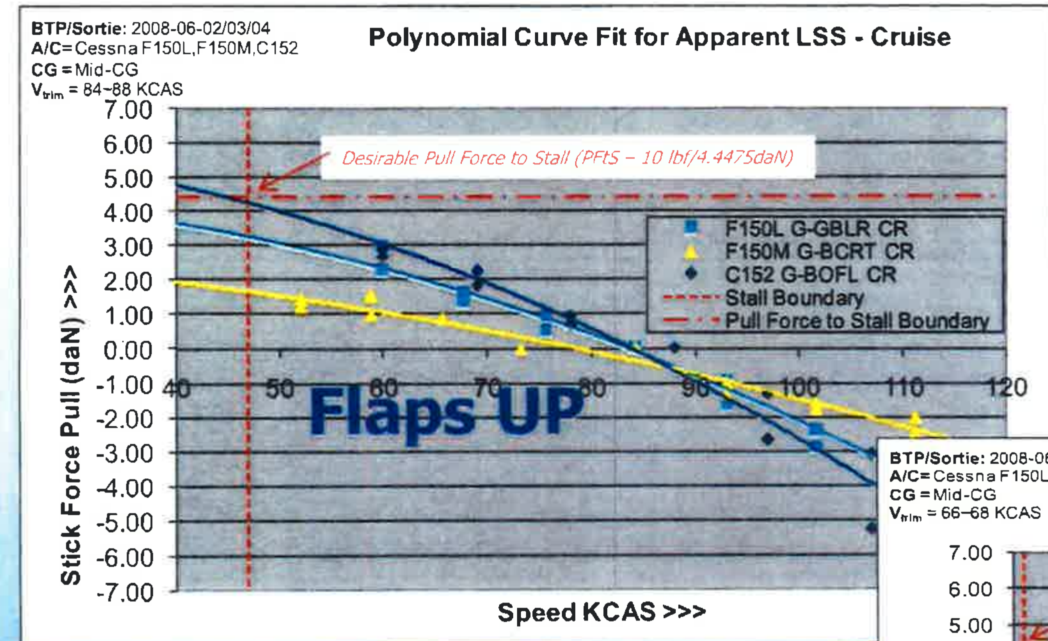
$$\boxed{\text{The wing's vertical location impacts } \sigma \quad 0.4\,\frac{z_w}{d} \text{ term}}$$

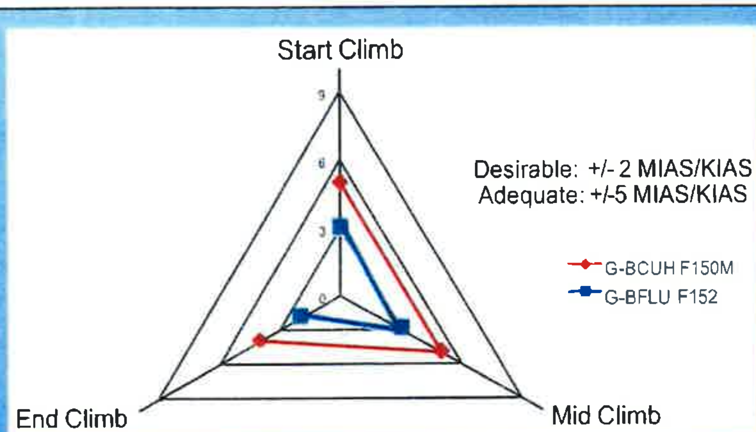
- Flow directions σ ~~outboard~~ above the wing turn the velocity vector inwards ~~[illegible]~~
- A sideslip angle β places the vertical in a different spanwise location (i.e. increasing β moves the tail into the right (starboard) wing's wake)
- σ increases going outboard above the starboard wing (decreases port above)

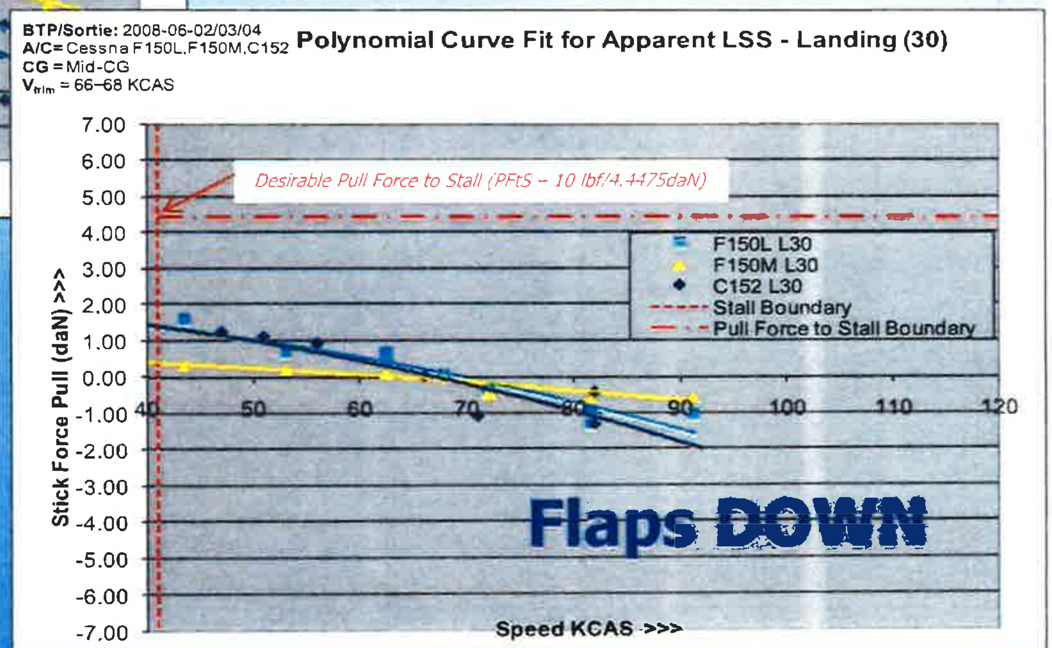
  Thus $\frac{d\sigma}{d\beta}$ is positive for a lower wing. (the ~~wing~~ vertical "sees" more of the upper wing wake)

Spot the difference...?

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

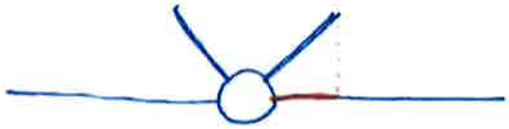


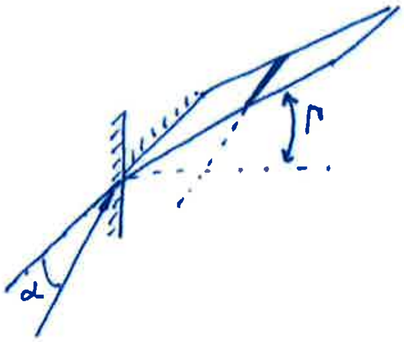




V-Tail Pitch Stability

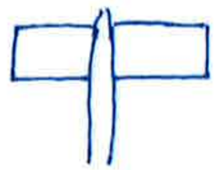
The proper size of a v-tail is NOT the projected area.



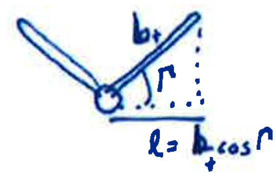


There are <u>two</u> terms from geometry:

1) projected area



Top view



2) Chordwise AOA

$$\alpha_{eff} = \alpha \cos\Gamma$$

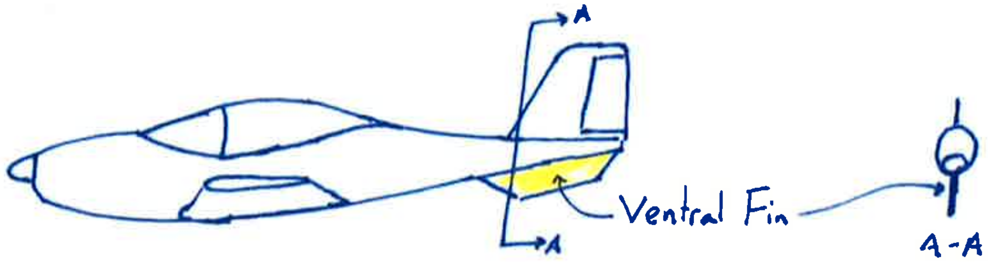
Thus

$$L = qSC_{L_\alpha}\alpha = q\underbrace{b_t \cos\Gamma\, \bar{c}}_{\text{projected Area}} \cdot C_{l_\alpha} \cdot \underbrace{\alpha \cos\Gamma}_{\text{effective AOA}}$$

$$\boxed{L = L_t \cos^2\Gamma}$$

If you use only the <u>projected</u> area, your tail will be <u>too small</u> by $\frac{1}{\cos\Gamma}$

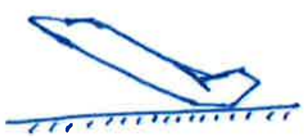
Ventral Fin



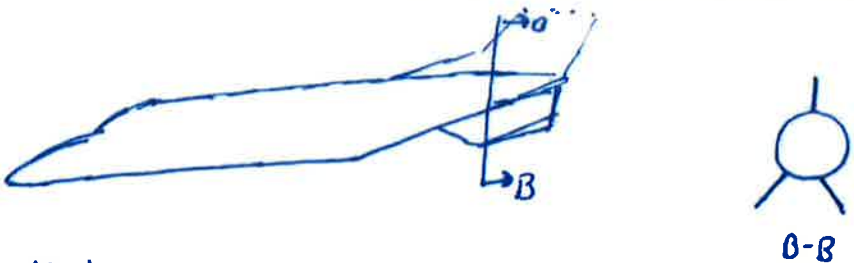
A vertical fin (or two) often seen on the underside of high performance aircraft. Usually the fin appears as a "plate" and is usually not controlled.

The ventral fin is particularly useful for maintaining directional stability at high AOA, when the vertical may be in stalled unsteady wakes. The ventral sticks down into undisturbed air.

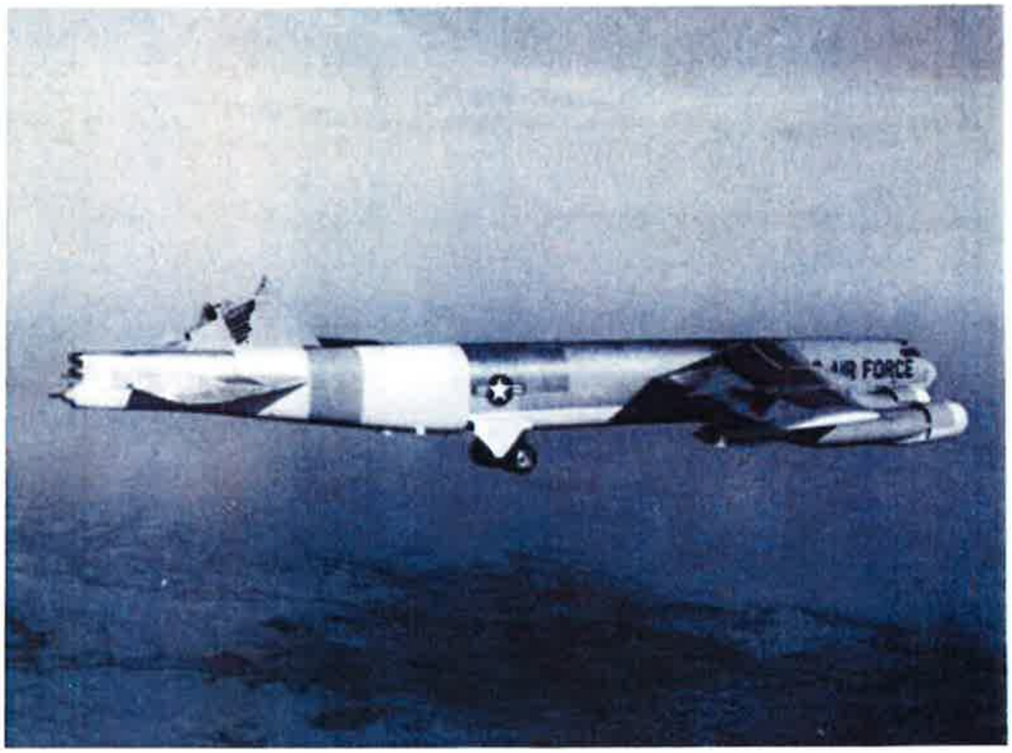
Ventral Fins are often limited by tail strike angles on takeoff + landing.
or even integrated into!



The Learjet 60 has an example of twin ventral fins ("Delta fins").



At high AOA, the twin ventrals provide both directional and long' stability.


AIR FORCE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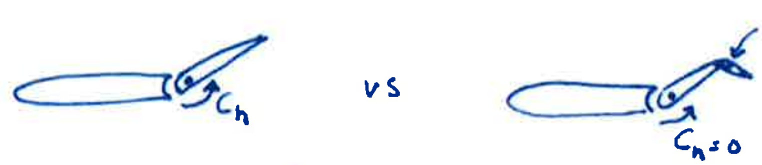
Servo Tabs

Q: How can a single pilot of a Boeing KC-135 provide enough force to control the 322 500 lbf aircraft? Only the rudder has a powered surface!

A: Servo Tabs.

We know that control surfaces are particularly good at generating a hinge moment.

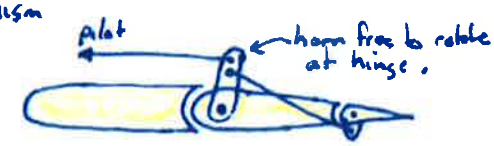
So, the strategy is to add a control tab to the control surface. The pilot moves the tab, which helps moves the surface.



but area of tab is much smaller than area of control surface

Think of servo tabs as amplifiers.

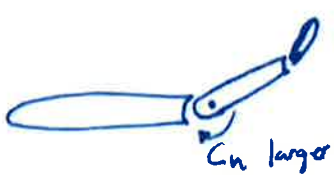
Mechanism



Anti-Servo tabs

Often, the control forces on small aircraft are too small.

An anti-servo tab is added to increase the effective hinge moment.

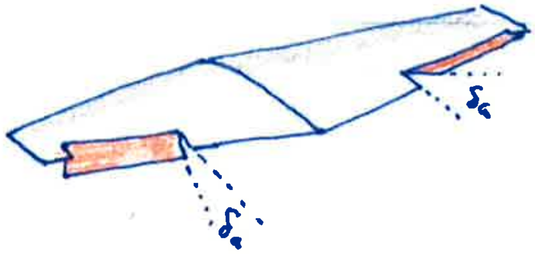


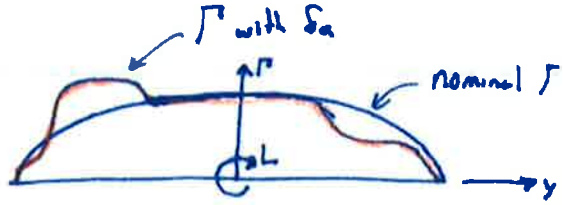
Mechanism

piper cherokee.

by adjusting this length, the tab acts as both an anti servo tab and a trim tab [illegible]!

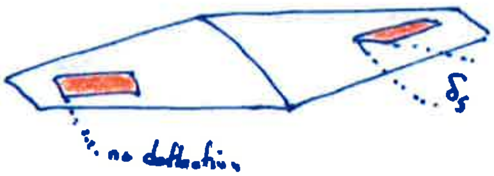
Ailerons





$$\Delta L_{moment} = \Delta L_{lift} \cdot y$$

Spoilers



Roll Moment:

$$dC_L = \frac{dL}{qSb} = \frac{(q C_\ell c)\, y\, dy}{qSb}$$

$$= \frac{C_\ell\, c\, y\, dy}{Sb}$$

Integrate

$$C_L = \int_{-b/2}^{b/2} \frac{C_\ell\, c\, y\, dy}{Sb}$$

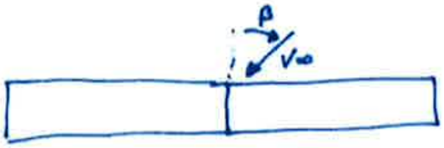
This is called strip theory (assume that C_ℓ depends only on α ... no Γ distribution).

Advantages?

- Control power (ailerons ✓ > spoilers X)
- Near Stall (ailerons $C_{\ell_{\delta_a}}$ ✓)
- C_ℓ distribution (ailerons X)
- Adverse yaw (ailerons X typically)
- Aeroelasticity (spoilers ✓)
- FCS rigging (ailerons ✓)

Dihedral

front view

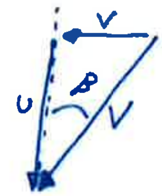


top view

Dihedral ~~gives~~ creates a roll moment given a side slip angle β.

Simplified Analysis

$$\beta = \operatorname{asin}\left(\frac{v}{|V|}\right) \approx \frac{v}{U} \quad \text{when } v \ll U$$



Thus $v \approx \beta V_\infty$

panel has upwash from β and Γ

$v_n \uparrow$

panel has downwash from β and Γ

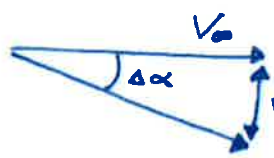
$v_n \downarrow$

In the wing panel frame, there is a normal component of the side velocity v.

$$v_n = v \sin\Gamma \approx v\Gamma$$

$$\approx \beta V_\infty \Gamma$$

Now that the normal component is known, compare to V_∞



$v_n = \beta V_\infty \Gamma$

$$\tan(\Delta\alpha) = \frac{v_n}{V_\infty} = \frac{\beta V_\infty \Gamma}{V_\infty}$$

For small angles, $\tan(\Delta\alpha) \approx \Delta\alpha = \beta\Gamma$

The local change in AOA is the product of sideslip angle and the dihedral angle

Dihedral Effect from Fuselage

The wing position contributes to dihedral effect

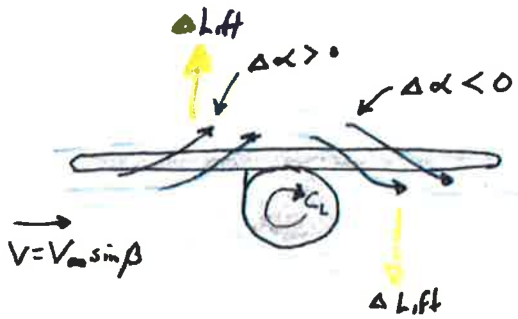
front view

No fuselage, no C_{ℓ_β}

front view

fuselage, no C_{ℓ_β}

perhaps C_{Y_β} (sideforce)

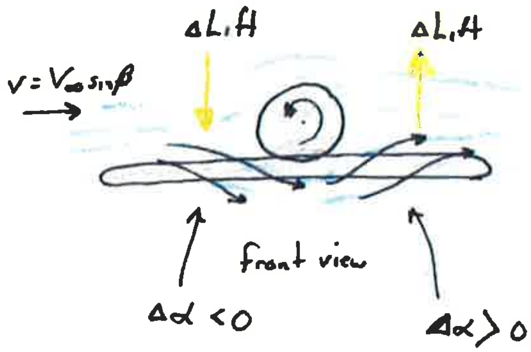


front view

fuselage + high wing

In this case, β from left (front view) gives a roll moment in negative x direction

$$C_{\ell_{\beta_{\text{high wing + fuselage}}}} < 0$$



fuselage + low wing

$$C_{\ell_{\beta_{\text{low wing}}}} > 0$$

McCormick says:

$$\Delta C_{\ell_{\beta_{\text{high}}}} \approx -0.00016 \ {}^{1}/_{\text{deg}}$$

$$\Delta C_{\ell_{\beta_{\text{low}}}} \approx 0.00016 \ {}^{1}/_{\text{deg}}$$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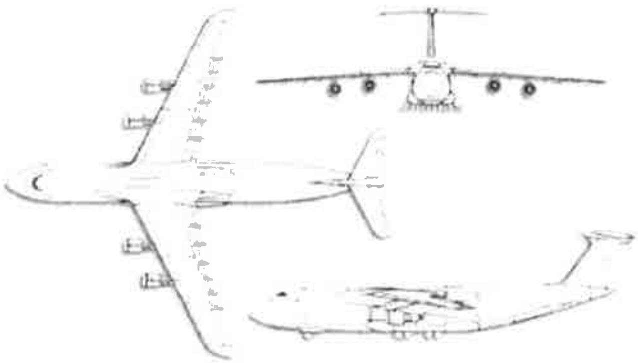
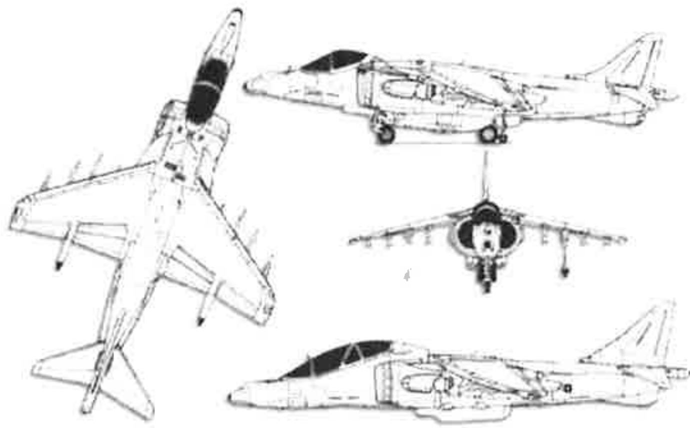
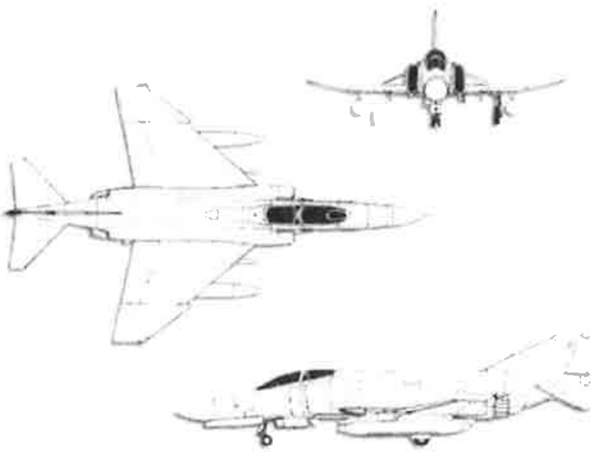
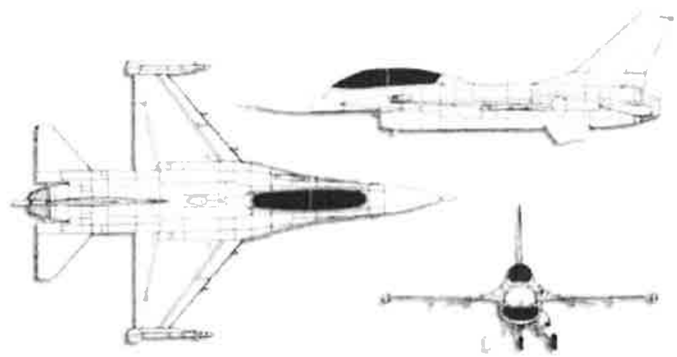
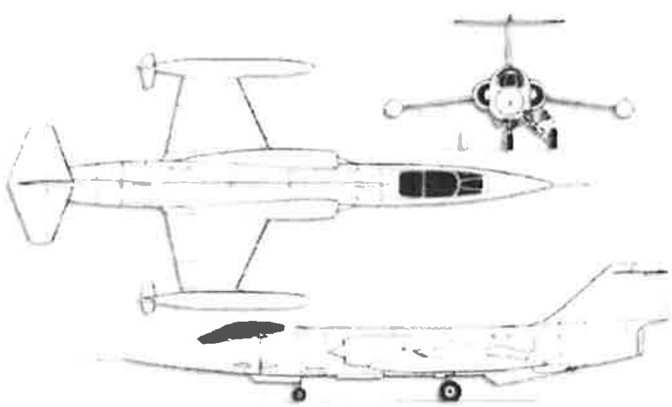
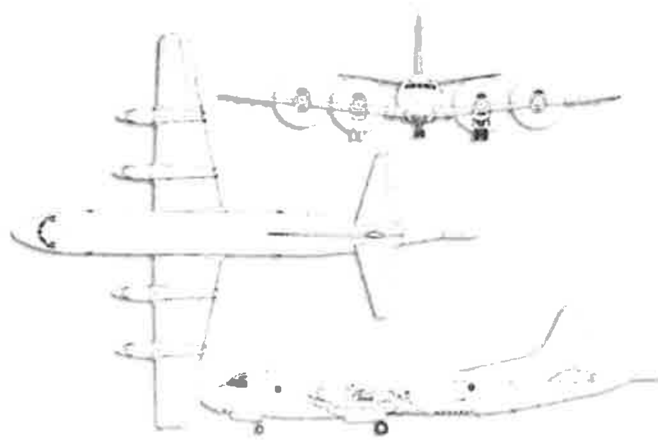
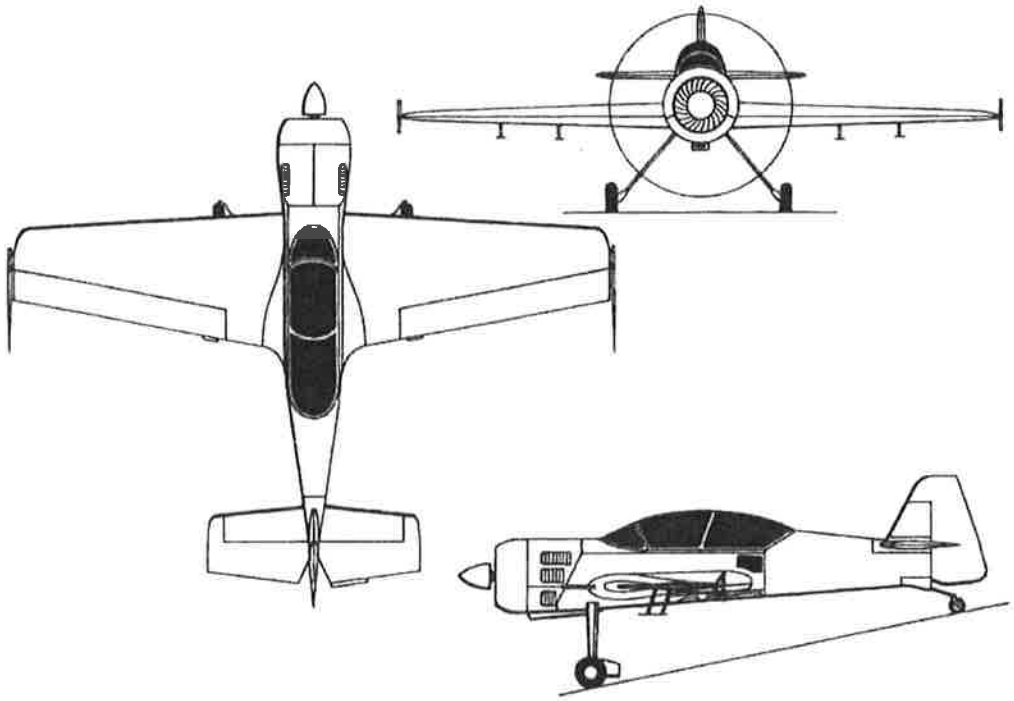
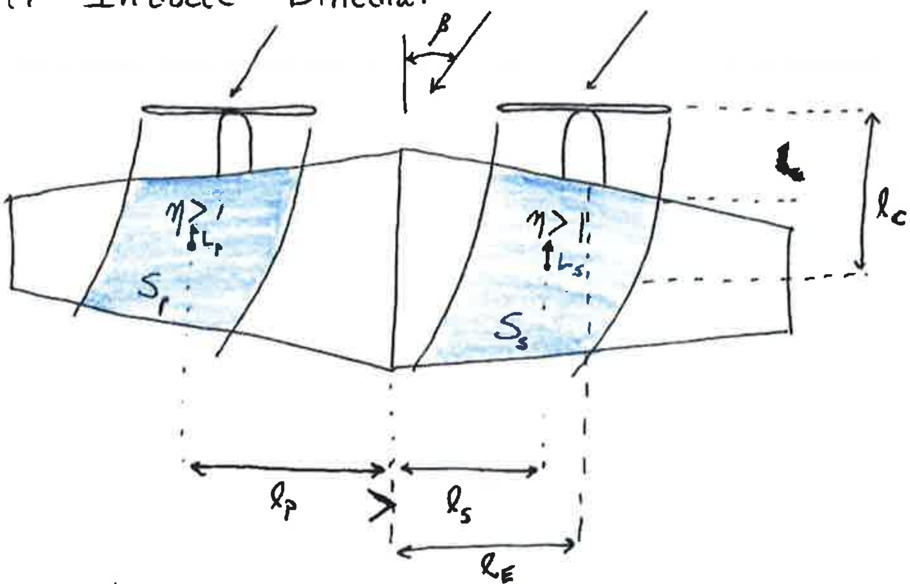









Power Induced Dihedral



$$L_{moment} = L_p l_p - L_s l_s$$

For a $\lambda \approx 1$, $S_p \approx S_s$ $\Rightarrow$ $L_{moment} = \eta q S_p C_{l_\alpha} \alpha_{eff}\, l_p - \eta q S_s C_{l_\alpha} \alpha_{eff}\, l_s$

$$= \eta q S_{sp} C_{l_\alpha} \alpha_{eff} (l_p - l_s)$$

$$C_L = \frac{L}{qSb} = \eta \frac{S_{sp}}{S} \left(\frac{l_p - l_s}{b} \right) C_{l_\alpha} \alpha_{eff}$$

Also, $l_p \approx l_E + \tan(\beta)\, l_c \approx l_E + \beta l_c$

$$l_s \approx l_E - \tan\beta\, l_c \approx l_E - \beta l_c$$

$$C_L \approx \eta \frac{S_{sp}}{S} \left(\frac{l_E + \beta l_c - l_E + \beta l_c}{b} \right) C_{l_\alpha} \alpha_{eff}$$

$$\approx \eta \frac{S_{sp}}{S} \frac{2\beta l_c}{b} C_{l_\alpha} \alpha_{eff}$$

$$\boxed{C_{L_{\beta_{power}}} \approx \underbrace{\eta \left(\frac{S_{sp}}{S} \right)}_{\text{power swept area}} \underbrace{\left(\frac{2}{b} l_c \right)}_{\text{extended power disk}} C_{l_\alpha} \underbrace{\alpha_{eff}}_{\text{AOA}}}$$

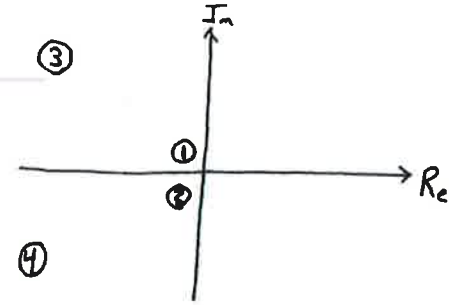
positive sign!! rolls into β

At high power settings and low speeds, high performance aircraft show <u>anhedral</u>. The Martin 202 prototype <u>had</u> to be fixed (add dihedral) to fix this issue (at a <u>great</u> sacrifice of the employees).

The longitudinal linearized equations gives 4 eigenvalues, but of two sets of conjugate pairs.

$$\lambda_{1,2} = -\text{small} \pm i \cdot \text{small} \Big\} \text{ low damping, low frequency "phugoid"}$$

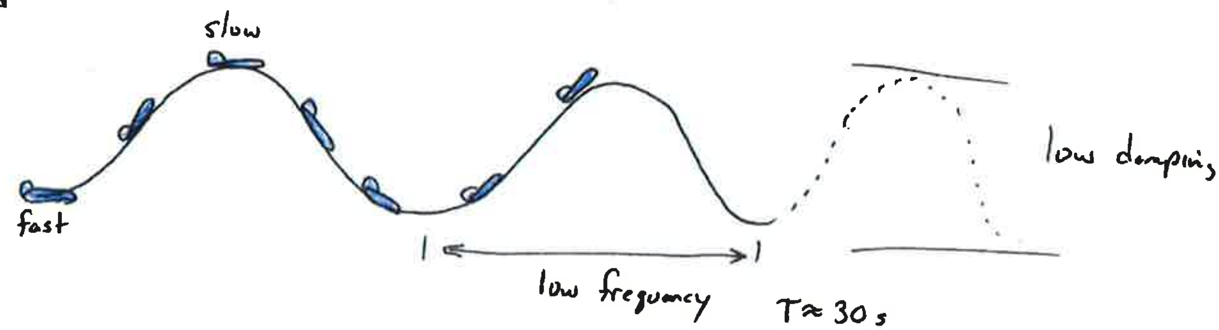
$$\lambda_{3,4} = -\text{large} \pm i \cdot \text{large} \Big\} \text{ high damping, high frequency "short period"}$$



Looking at the eigenvectors shows that:

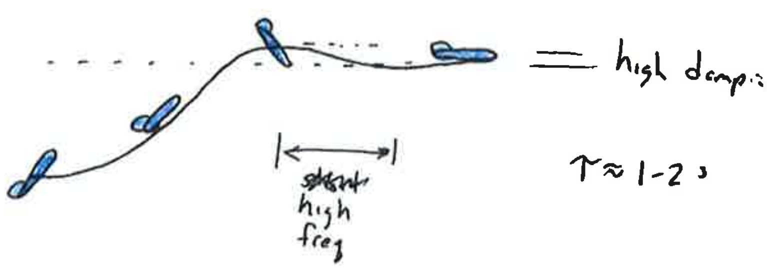
- The phugoid contains mostly Δu and $\Delta\theta$
- The short period contains mostly $\Delta\alpha$ and $\Delta\theta$ $\left(\Delta\alpha \approx \frac{\Delta w}{U_0}\right)$

Phugoid



Short Period

$\boxed{\text{Exaggerated scale}}$



very little velocity change

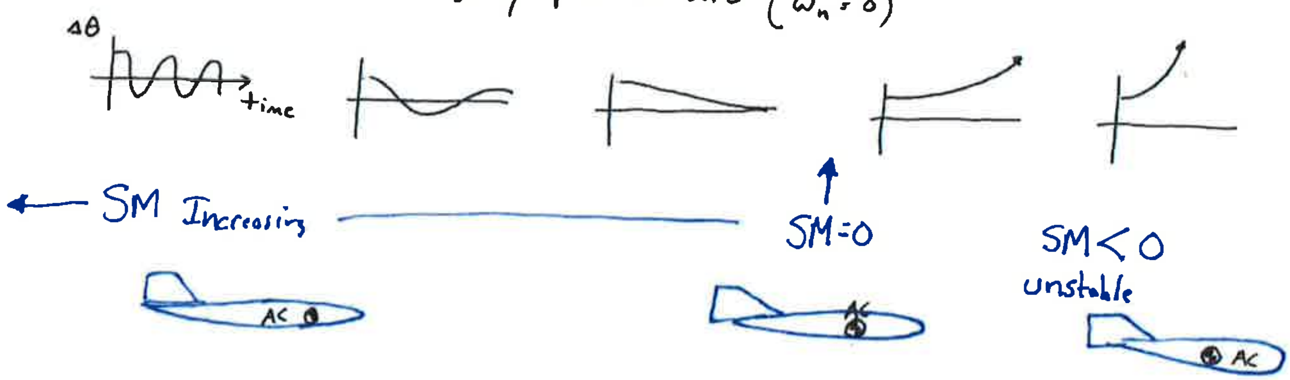
Summary

- The dynamics of longitudinal motion (pitching motion, velocity) indicates two behaviors: ① Phugoid and ② Short Period

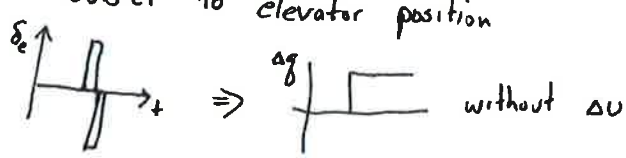
- Phugoid scales with $\omega = \frac{g}{U_0}\sqrt{2}$ and $\zeta = \frac{1}{\sqrt{2}}\frac{1}{L/D}$

- Short Period scales with Tail size and Static margin

- At a static margin of zero, the phugoid damping ($\lambda = \eta \pm i\omega$) goes through zero. The imaginary part is zero ($\omega_n = 0$)



- The short period is often felt in turbulence when Δw changes due to atmospheric disturbances. We can also excite S.P through Δq with a doublet to elevator position



- Excessively stable aircraft (SM too large) are overly damped in rough air. The ride quality is poor.

Dutch Roll

Visualization

tiny.cc/AEM368-DutchRoll-1

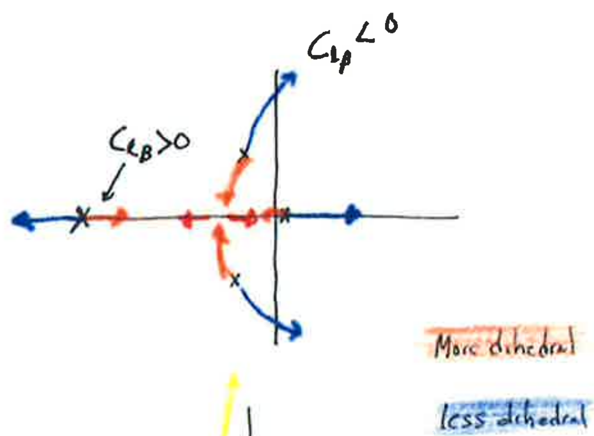
tiny.cc/AEM368-DR2

tiny.cc/AEM368-DR3

tiny.cc/AEM368-DR4

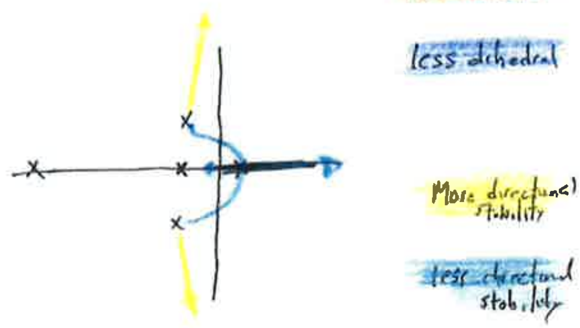
This is a combination of the lateral modes with a phase shift.

Design Parameters on lateral modes



a) Increase Dihedral C_{l_β}
- Improves Spiral mode ✓
- Degrades Dutch Roll ✗

b) Increase Directional stability C_{n_β}
- Improves Dutch Roll ✓
- Degrades Spiral ✗



c) Increase Yaw damping C_{n_r}
- Improves both ✓ ✓

But we usually can't aerodynamically obtain C_{n_r} without C_{n_β}!

2 typical solutions

1) Vertical below cg

2) Yaw damper stability feedback (AEM 468)