

Aeronautics Design Project



Flight stability

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Content of the course



- Introduction
- Control surfaces
- Geometry of airplane
- Static stability
- Longitudinal stability
- Lateral stability
- Dynamic stability

Complete procedures in text books:

- D.P. Raymer, *“Aircraft Design : A conceptual Approach”*
- E. Torenbeek, *“Synthesis of Subsonic Airplane Design”*
- R. D. Finck. *“USAF Stability and Control DATCOM”*

Stability and Control



Stability = ability to **keep** the aircraft in the air in the chosen flight attitude and to counteract disturbances

Control = ability to **change** the flight direction and attitude of the aircraft

Both issues are :

- Not design criteria (aircraft are designed for performance)
- Investigated in the preliminary design process

Stability and control refer to “Flight Mechanics” (AERO0003-1)

Flight mechanics ensure to:

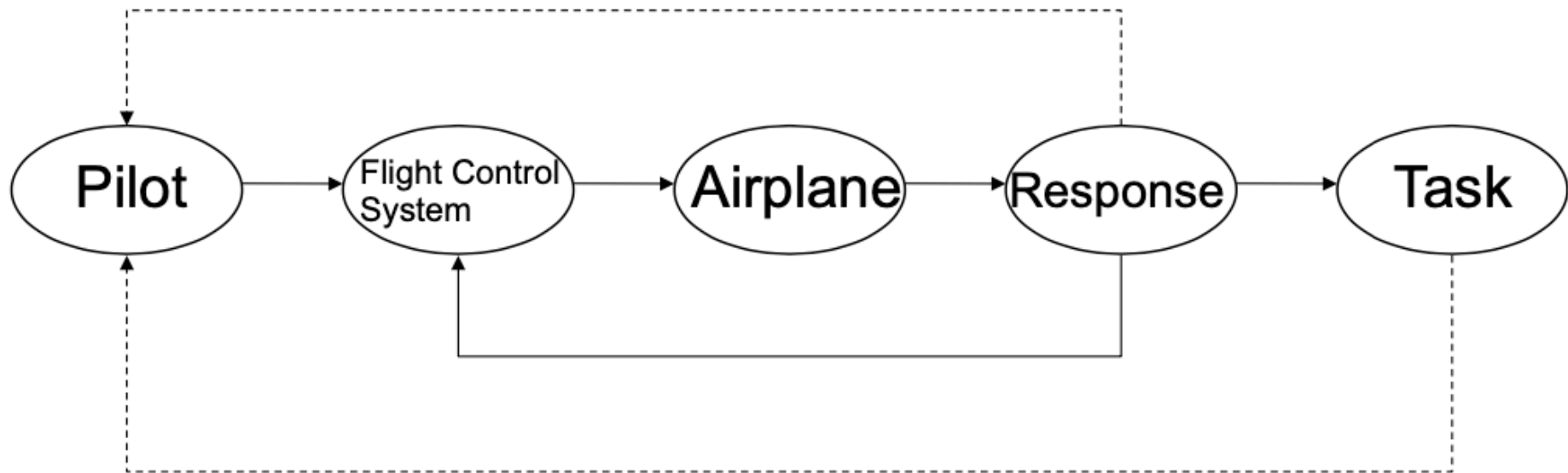
- Design an airplane able to accomplish efficiently a mission
- Make the task of the pilot easier (good handling in flight)
- Avoid unwanted/unexpected phenomena

Aircraft controls



Pilot :

- Controls only the Flight Control System (FCS)
- Tailors the inputs on the FCS by observing aircraft's response
- Always keeping an eye on the task at hand



Aircraft degrees of freedom



6 DOFs around the center of gravity (c.g.)

3 displacements:

x = horizontal motion

y = side motion

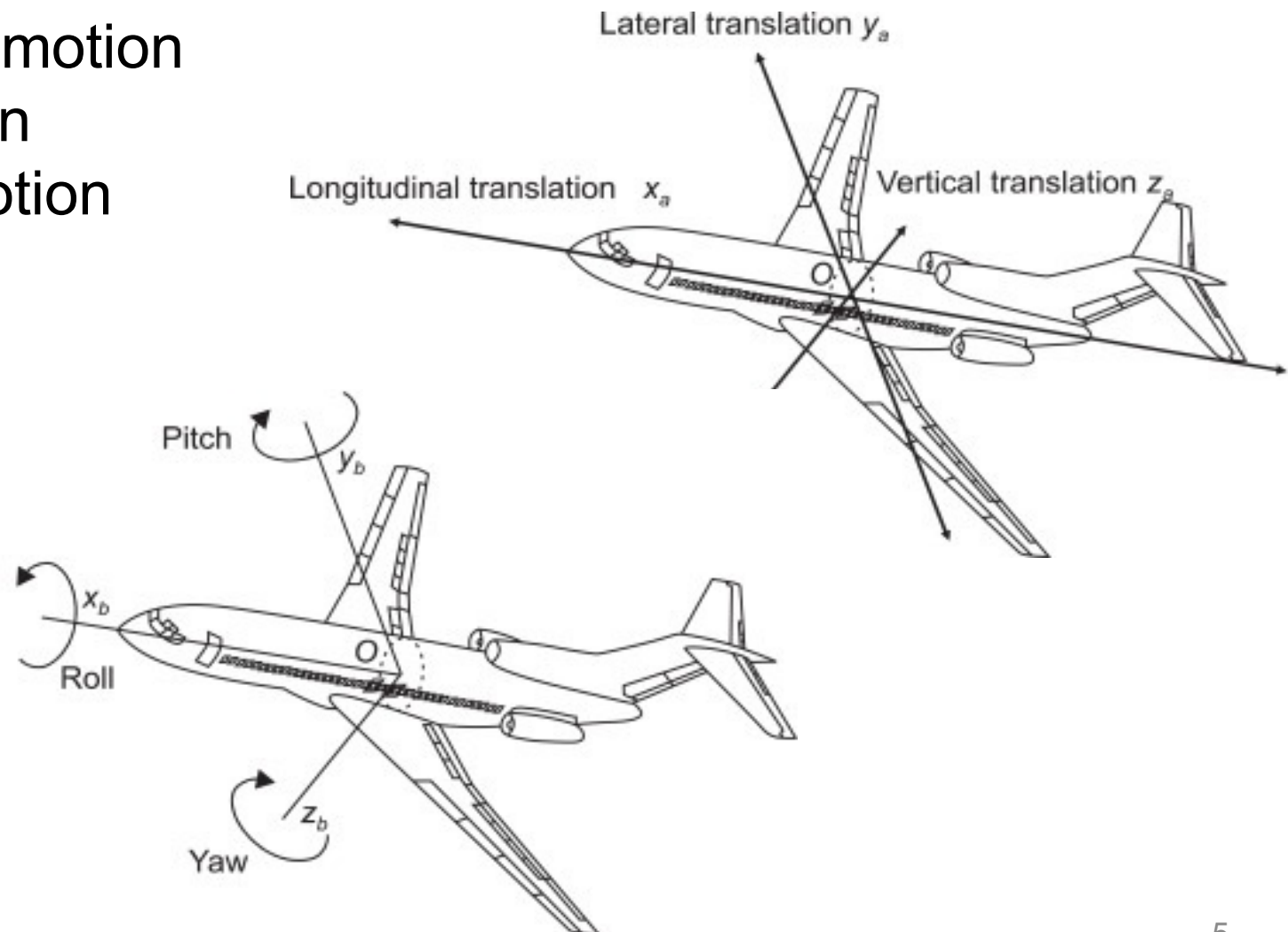
z = vertical motion

3 rotations:

x = roll

y = pitch

z = yaw



Control surfaces



Aircraft is controlled via control surfaces and power:

- Ailerons (used in pairs to control roll)
- Elevators (on the tail to control pitch)
- Rudder (on the fin to control yaw)
- Throttle (adjust the thrust of engine(s))

Rudder



Elevator

Aileron



Rudderon
(rudder+aileron)



Elevon
(elevator+aileron)



Throttle

Other control surfaces



Flaps

Airbreak

Spoilers



Vectored thrust

Combination of control surfaces



Decelerons



Spoileron



Flaperon



Wing geometry



- Standard mean chord (smc) $\rightarrow$ pure geometric quantity

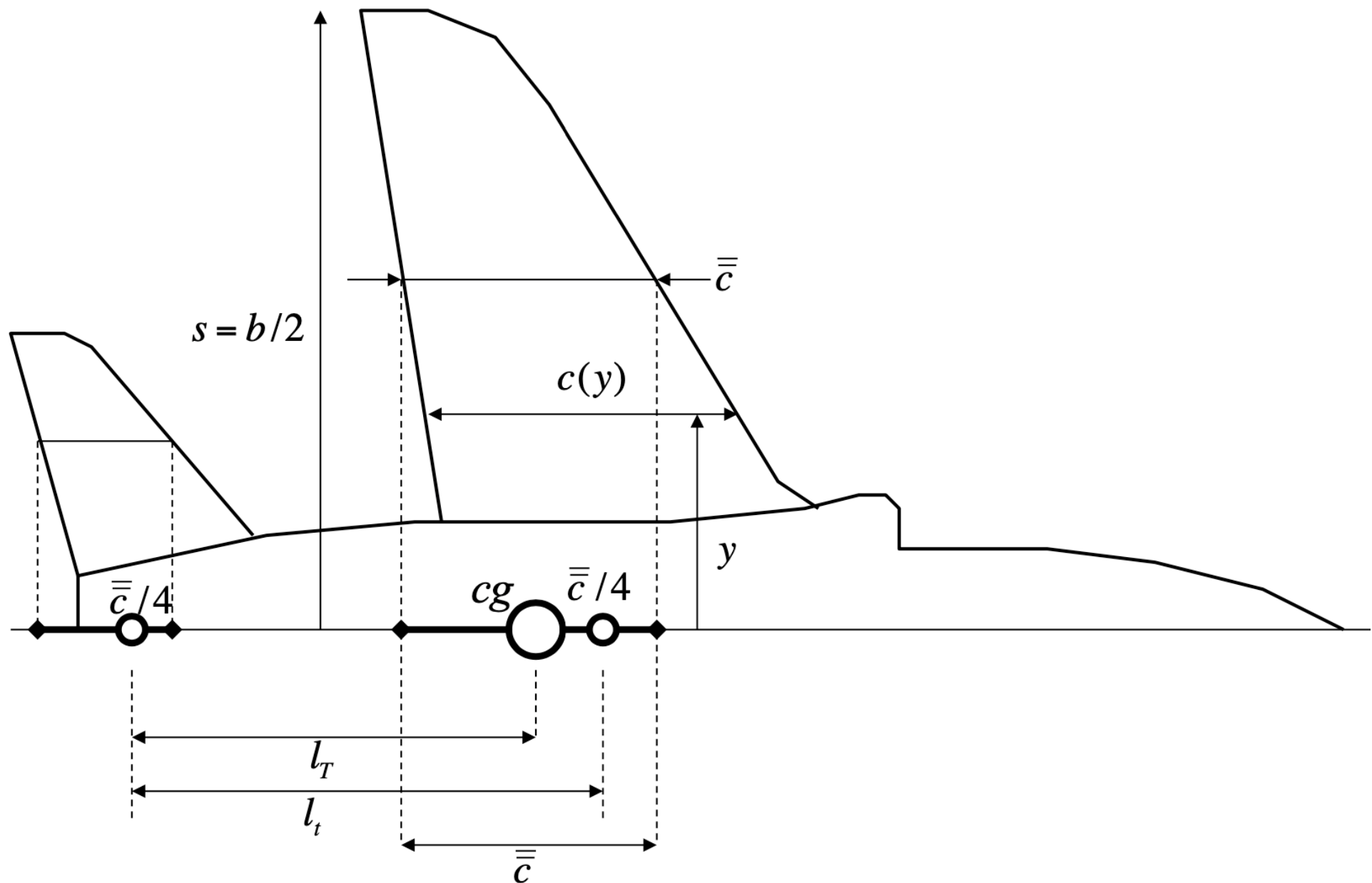
$$\bar{c} = \int_{-s}^s c(y) dy / \int_{-s}^s dy$$

- Mean aerodynamic chord (mac) $\rightarrow$ aerodynamic meaning

$$\bar{\bar{c}} = \int_{-s}^s c^2(y) dy / \int_{-s}^s c(y) dy$$

- Span of the wing = $b \rightarrow s = b/2$ (semi-span)
- Wing area : $S = b\bar{c}$
- Aspect Ratio : $AR = b^2/S$

Airplane geometry

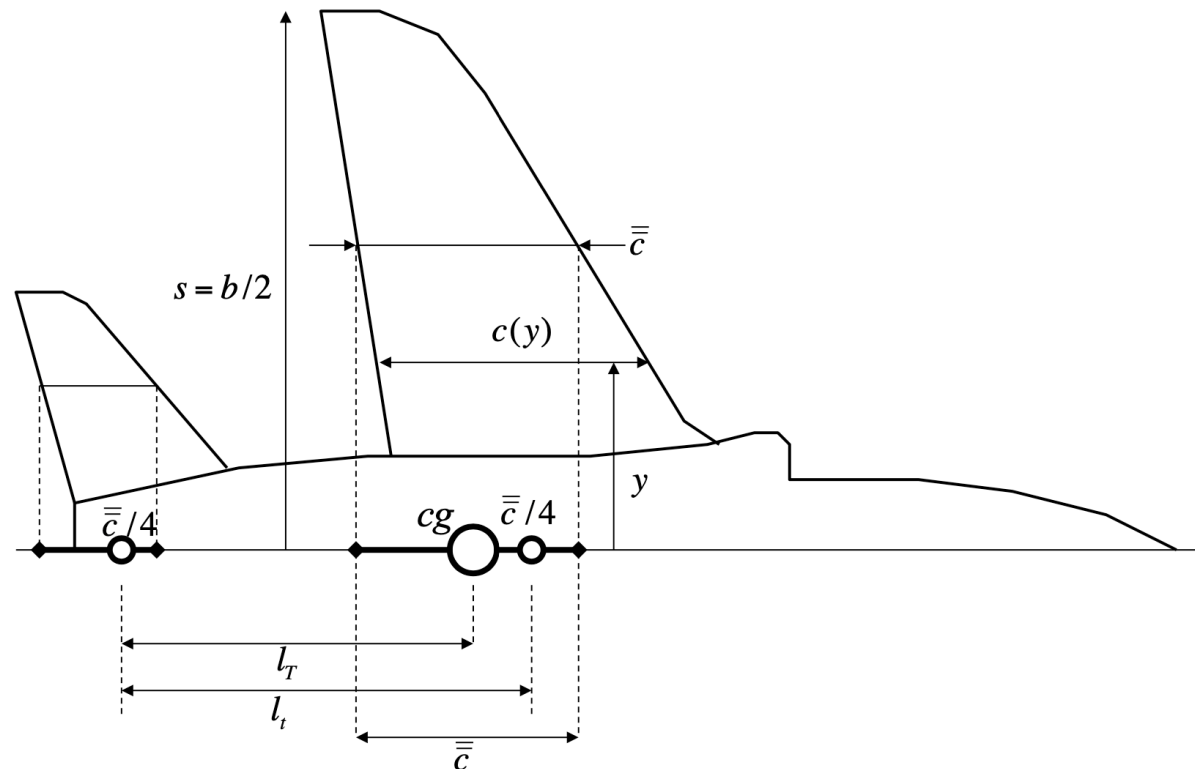




Tail geometry

- Tailplane area : S_T
- Tailplane moment arm : l_T
- Tailplane volume ratio ($\bar{V}_T$) is a measure of the aerodynamic effectiveness of the tail

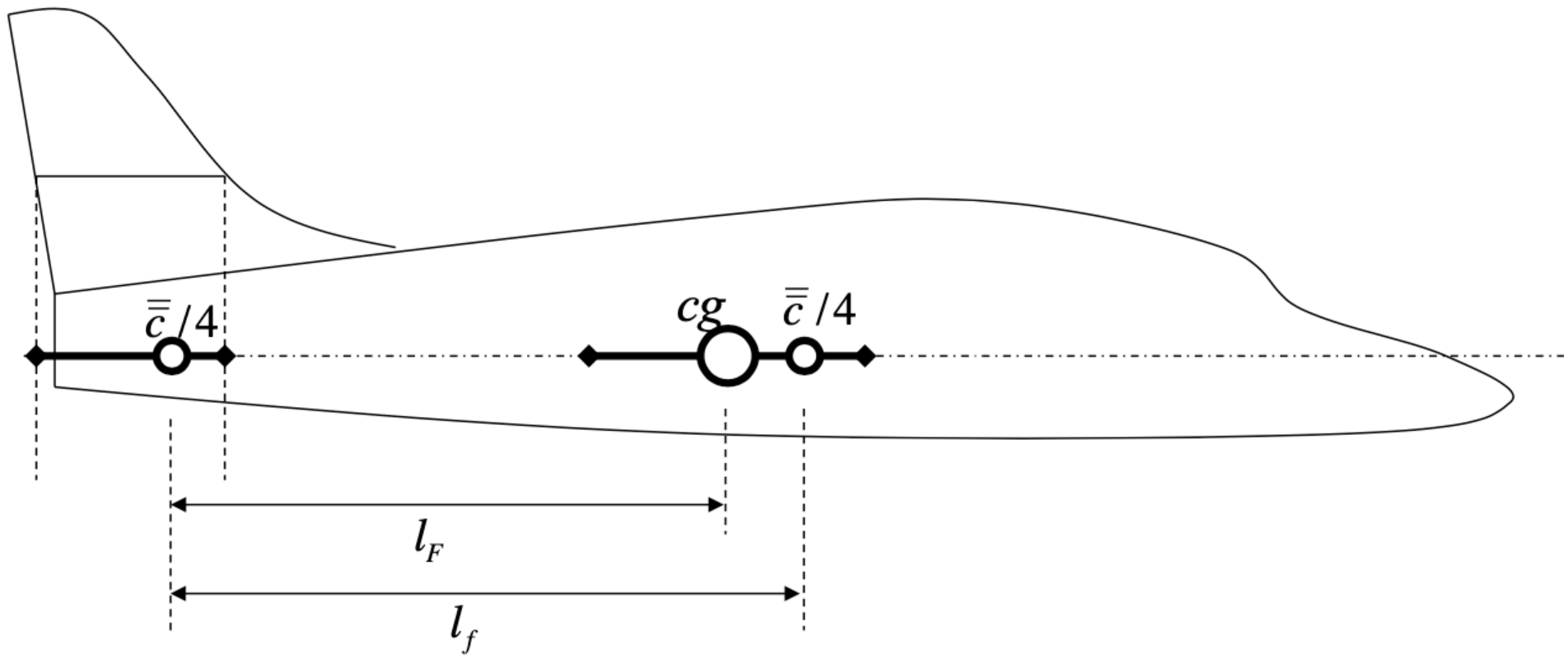
$$\bar{V}_T = \frac{S_T l_T}{S \bar{c}}$$



Fin geometry



- Fin moment arm : l_F
- Fin volume ratio / effectiveness : $\bar{V}_F = \frac{S_F l_F}{S \bar{\bar{c}}}$



Reference points



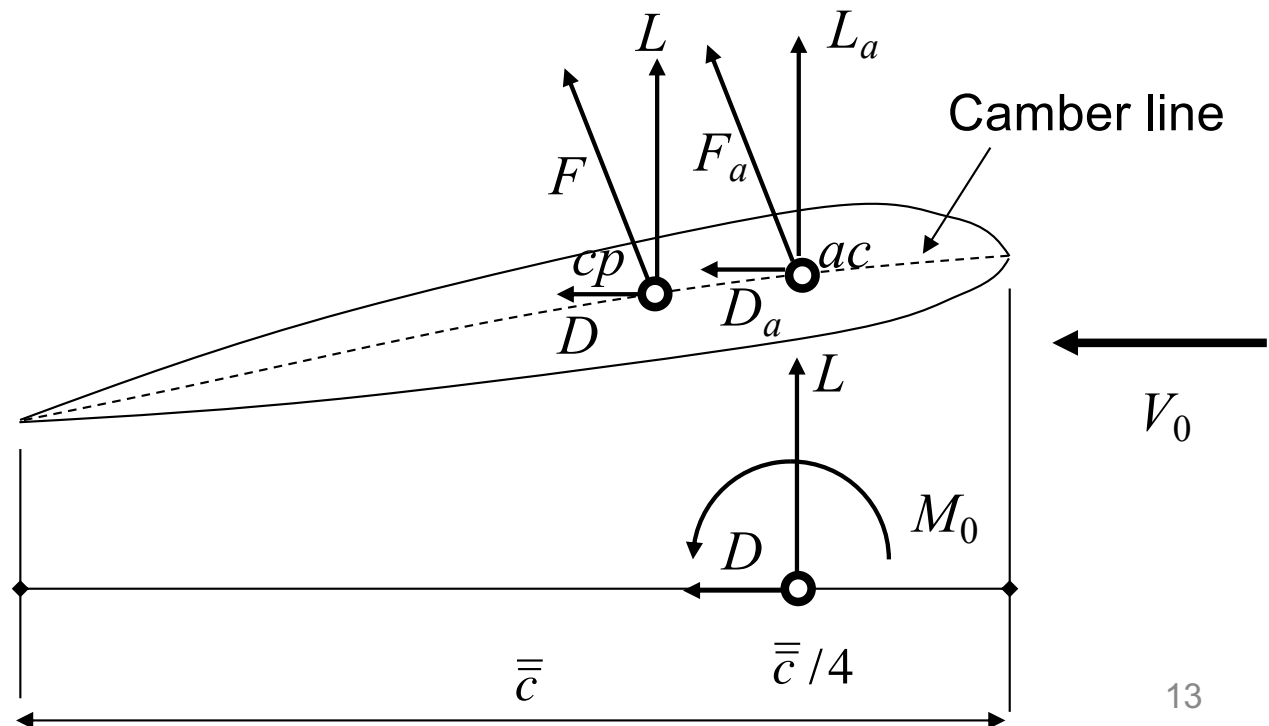
Centre of pressure (*cp*)

Position of the aerodynamic resultant force, where no aerodynamic moment applies

Aerodynamic centre (*ac*)

Point at which aerodynamic force due to angle of attack, F_a , acts. Aerodynamic moment, M_o , is independent of angle of attack

Located around $\frac{\bar{c}}{4}$



Static stability



Stability conditions:

- **At fixed flight conditions :**

→ *All forces and moments around c.g. are balanced*

= TRIM position (adjusted using trim tabs)

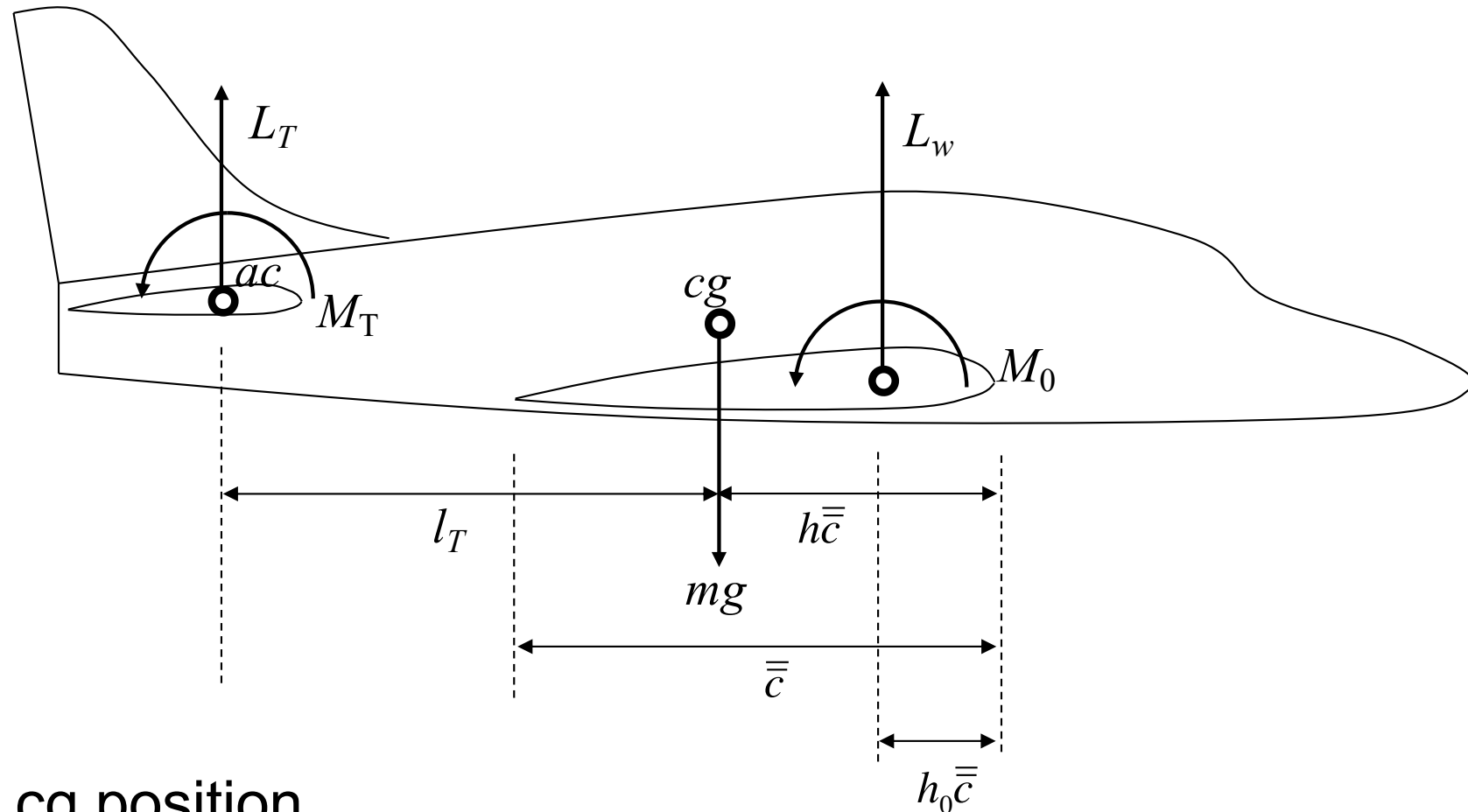
- **After any small perturbation in flight attitude**

→ *Aircraft returns to its equilibrium position*

Aircraft static stability must be ensured along:

- Longitudinal axis
- Lateral axis

Longitudinal stability



h = cg position

h_0 = position of the aerodynamic center of the wing

l_T = (dimensional) position of the aerodynamic center of the tail

Pitching moment equation



Assuming:

- Steady level flight
- Thrust balances drag and both pass by the c.g.

Vertical balance:

$$L_w + L_T - mg = 0$$

Pitching moment (around c.g.):

$$M = M_0 + L_w(h - h_0)\bar{\bar{c}} - L_T l_T + M_T = 0$$

(positive nose-up)

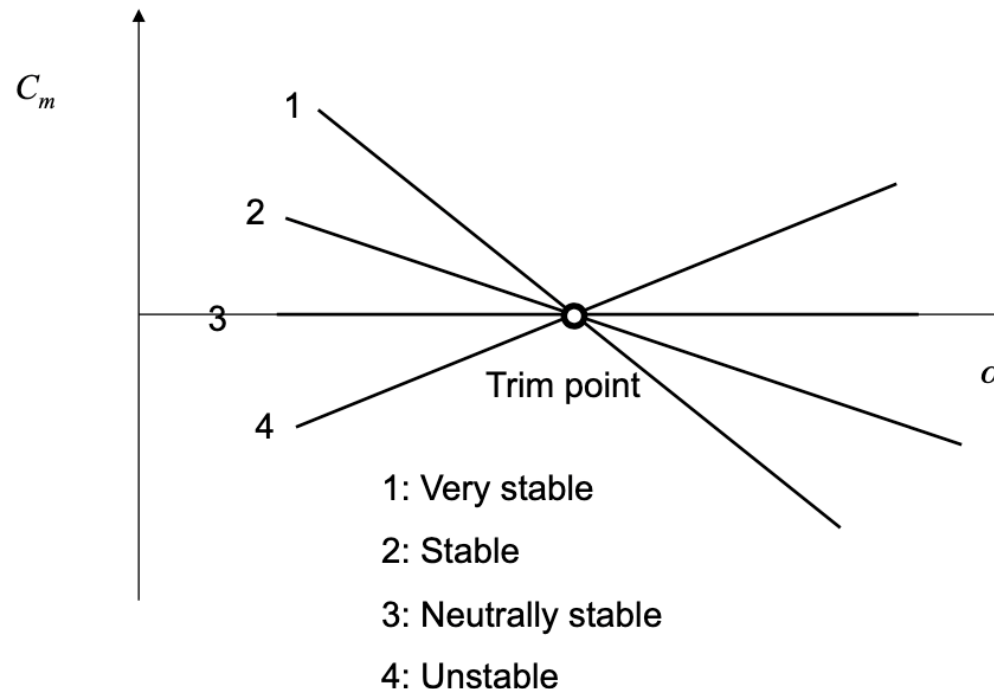
Pitching stability



Equilibrium point can be **stable**, **unstable** or **neutrally stable**

Stable equilibrium point is characterized by

$$M = 0 \quad \text{and} \quad \frac{dM}{d\alpha} < 0 \quad \text{or} \quad C_m = 0 \quad \text{and} \quad \frac{dC_m}{d\alpha} < 0$$



A more general condition, to get rid of compressibility effects, is

$$C_m = 0 \quad \text{and} \quad \frac{dC_m}{dC_L} < 0$$

Pitching stability



Pitching moment equation

$$M = M_0 + L_w(h - h_0)\bar{c} - L_T l_T + M_T = 0$$

Assuming a symmetric tailplane so that $M_T = 0$

The equation can be re-written

$$C_m = C_{m_0} + C_{L_w}(h - h_0) - C_{L_T}\bar{V}_T = 0$$

where

$$C_m = \frac{M}{\frac{1}{2}\rho V_0^2 S \bar{c}} \quad C_{L_w} = \frac{L_w}{\frac{1}{2}\rho V_0^2 S} \quad C_{L_T} = \frac{L_T}{\frac{1}{2}\rho V_0^2 S_T}$$



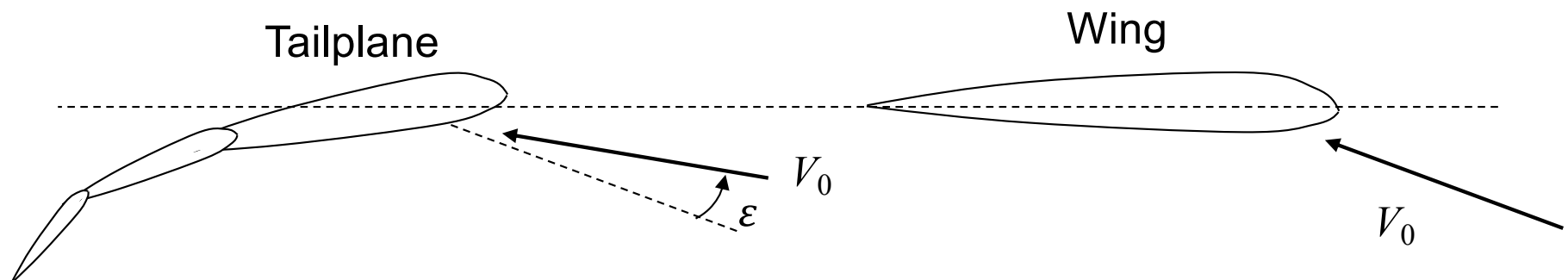
Pitching stability

Static stability if $\frac{dC_m}{dC_L} < 0$ or approximately $\frac{dC_m}{dC_{L_w}} < 0$

$$C_m = C_{m_0} + C_{L_w}(h - h_0) - C_{L_T}\bar{V}_T$$

$$\rightarrow \frac{dC_m}{dC_{L_w}} = \underbrace{\frac{dC_{m_0}}{dC_{L_w}}}_{=0} + \underbrace{\frac{dC_{L_w}}{dC_{L_w}}}_{=1}(h - h_0) - \underbrace{\frac{dC_{L_T}}{dC_{L_w}}}_{\text{must be calculated}}\bar{V}_T$$

Tailplane is impacted by the downwash effect from the wing
→ downwash angle ε



Wing-tail flow

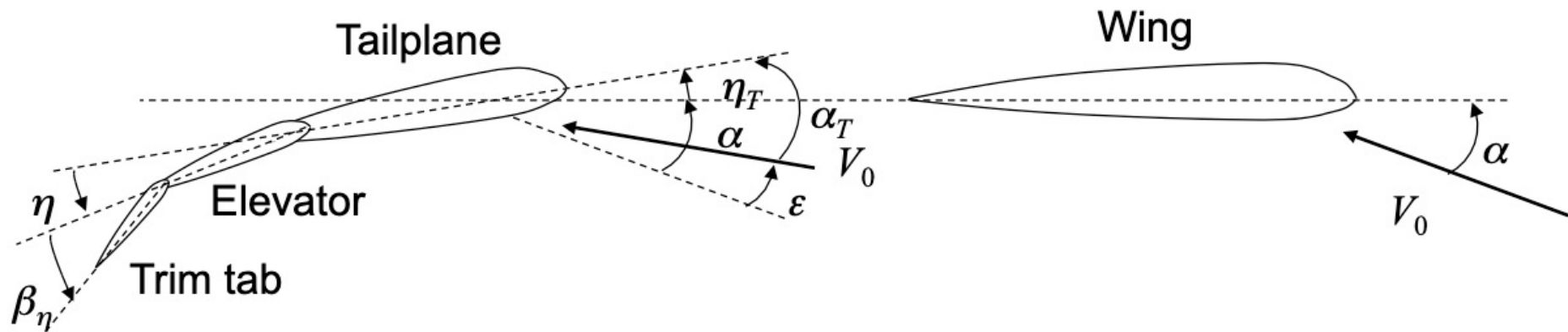


Total angle of attack of the tailplane

$$\alpha_T = \alpha - \varepsilon + \eta_T$$

Lift coefficient of the tailplane

$$C_{L_T} = a_0 + a_1 \alpha_T + a_2 \eta + a_3 \beta_\eta$$



Wing-tail flow



Downwash on the tailplane, for small disturbances

→ ε is a linear function of wing incidence α

$$\rightarrow \varepsilon = \frac{d\varepsilon}{d\alpha} \alpha$$

The lift coefficient of the wing is also a linear function of α

$$C_{L_w} = a \alpha \rightarrow \alpha = C_{L_w} / a$$

$$\text{So that, } \alpha_T = \alpha - \varepsilon + \eta_T = \frac{C_{L_w}}{a} \left(1 - \frac{d\varepsilon}{d\alpha} \right) + \eta_T$$

Wing-tail flow



Lift coefficient of the tailplane becomes

$$C_{L_T} = a_0 + a_1 \frac{C_{L_W}}{a} \left(1 - \frac{d\varepsilon}{d\alpha} \right) + a_1 \eta_T + a_2 \eta + a_3 \beta_\eta$$

The derivation over C_{L_W} yields

$$\frac{dC_{L_T}}{dC_{L_W}} = \frac{a_1}{a} \left(1 - \frac{d\varepsilon}{d\alpha} \right) + a_2 \frac{d\eta}{dC_{L_W}} + a_3 \frac{d\beta_\eta}{dC_{L_W}} \quad (\text{since } \eta_T \text{ is constant})$$

The derivative of the pitching moment coefficient becomes:

$$\frac{dC_m}{dC_{L_W}} = (h - h_0) - \left[\frac{a_1}{a} \left(1 - \frac{d\varepsilon}{d\alpha} \right) + a_2 \frac{d\eta}{dC_{L_W}} + a_3 \frac{d\beta_\eta}{dC_{L_W}} \right] \bar{V}_T$$

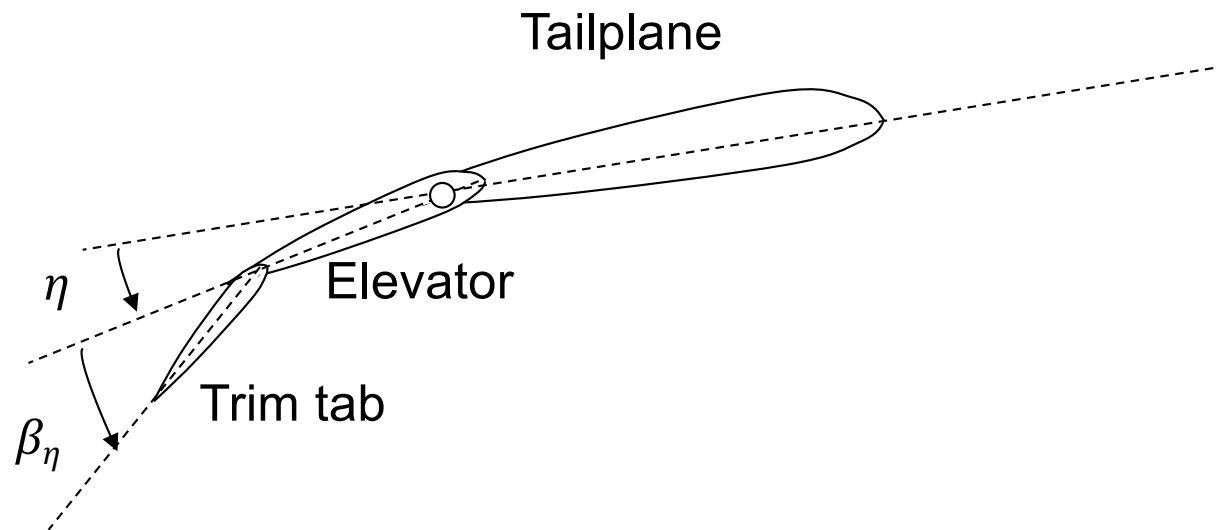
Controls



Two manners to control the aircraft:

- fixed/locked controls $\rightarrow \eta$ and β_η are constant
- free controls $\rightarrow \eta$ and β_η are variable

$\rightarrow$ the pitching stability $\frac{dC_m}{dC_{L_w}} < 0$ will take different forms



Fixed controls



Assuming a trimmed aircraft

What is the effect of a small perturbation (e.g. gust) ?

The pitching moment equation becomes

$$\frac{dC_m}{dC_{L_w}} = (h - h_0) - \left[\frac{a_1}{a} \left(1 - \frac{d\varepsilon}{d\alpha} \right) + a_2 \frac{d\eta}{dC_{L_w}} + a_3 \frac{d\beta_\eta}{dC_{L_w}} \right] \bar{V}_T$$

$$\rightarrow \frac{dC_m}{dC_{L_w}} = (h - h_0) - \bar{V}_T \frac{a_1}{a} \left(1 - \frac{d\varepsilon}{d\alpha} \right)$$

Controls fixed **stability margin**, $K_n = -\frac{dC_m}{dC_{L_w}} = h_n - h$

where h_n is the controls fixed **neutral point**, $h_n = h_0 + \bar{V}_T \frac{a_1}{a} \left(1 - \frac{d\varepsilon}{d\alpha} \right)$

Fixed controls

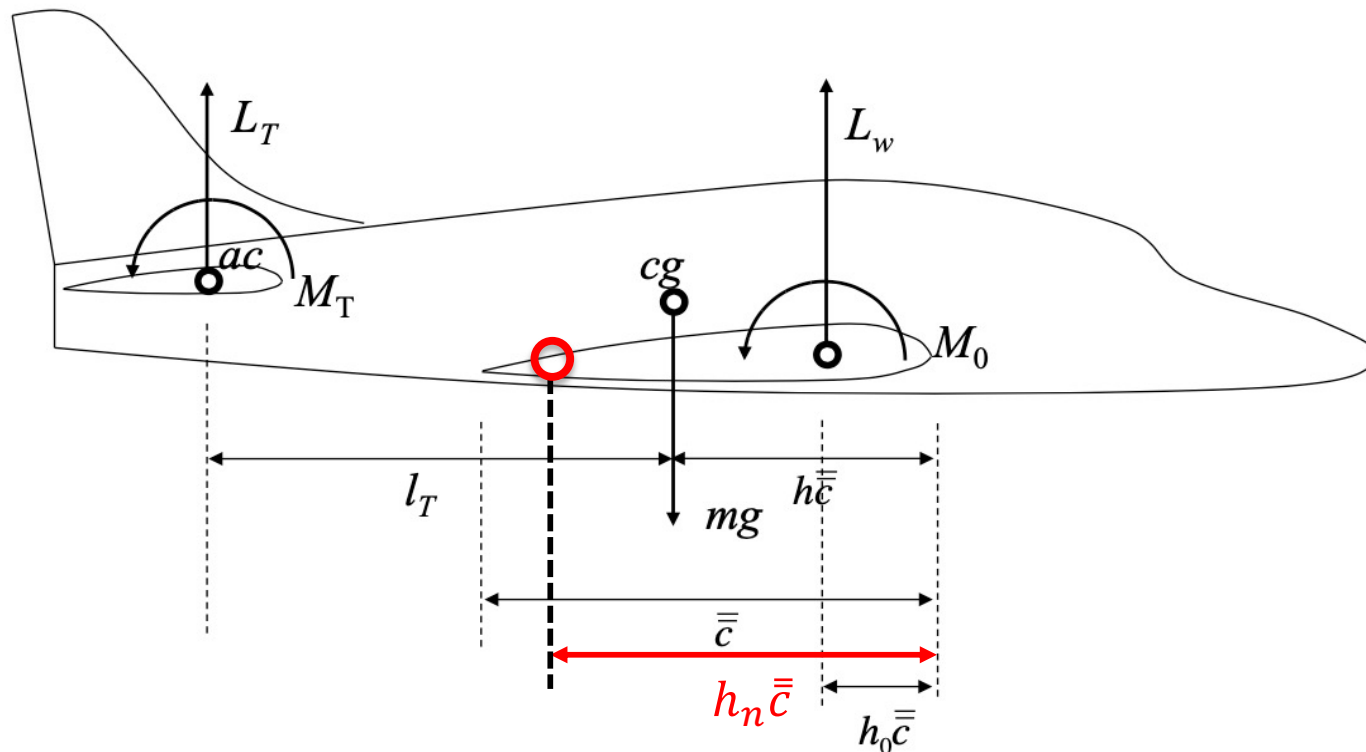


A **stable** aircraft has a **positive stability margin** : $K_n > 0$

(the more positive, the more stable)

Stable aircraft if the cg position is ahead of the neutral point

$$\rightarrow h_n - h > 0$$



Fixed controls



Certification authorities specify that

$$K_n \geq 0.05 \text{ at all times}$$

Too much stability can be a bad thing !

Stability margin may change if:

- fuel is burned
- payload is released
(missiles, bombs, external fuel tanks, paratroopers, ..)

Free controls

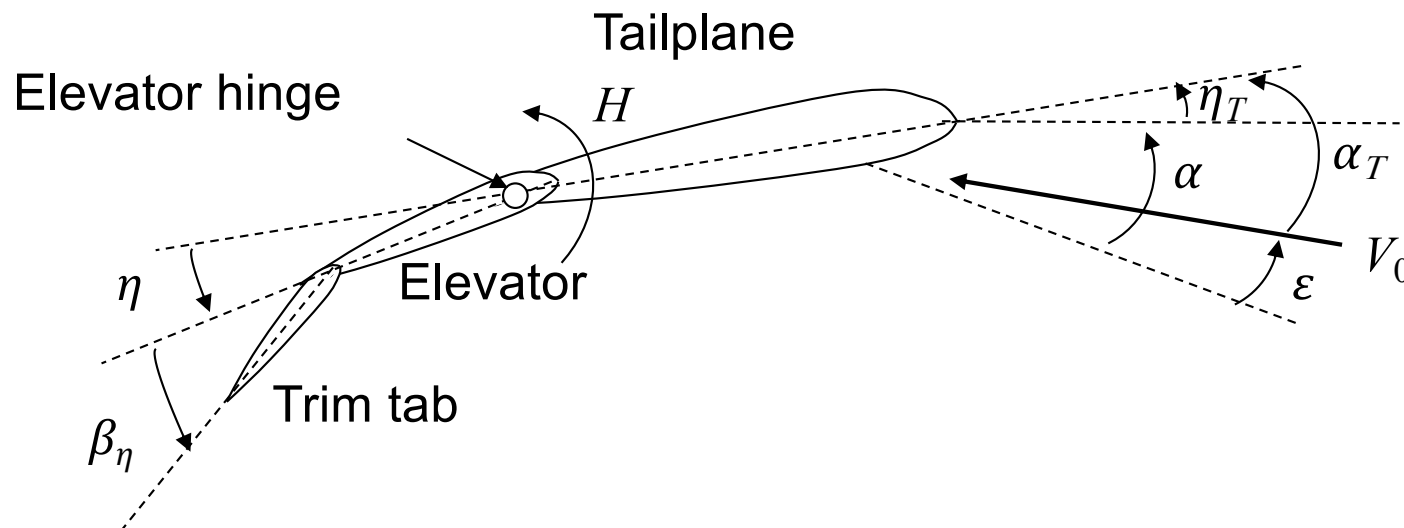


Pilots **don't want to hold the controls** throughout the flight

→ Trim tab (β_η) can be adjusted such that if the elevator float freely, it will at an angle (η) corresponding to the desired trim condition

= hands-off trim condition

(pilot does not need to adjust the elevator)





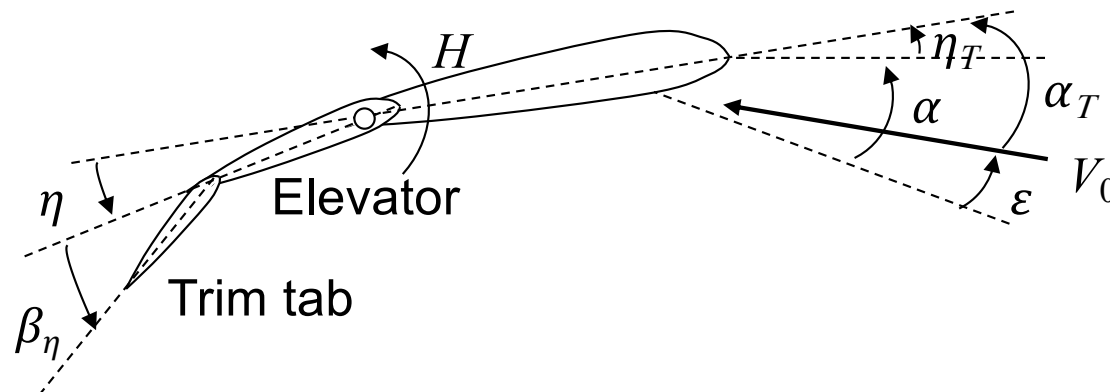
At hands-off trim condition

The derivative of the pitching moment with the wing lift is

$$\frac{dC_m}{dC_{L_w}} = (h - h_0) - \bar{V}_T \frac{a_1}{a} \left(1 - \frac{d\varepsilon}{d\alpha}\right) \left(1 - \frac{a_2 b_1}{a_1 b_2}\right)$$

where b_1 , b_2 and b_3 define the elevator hinge moment

$$C_H = b_1 \alpha_T + b_2 \eta + b_3 \beta_\eta$$



Free controls



Controls **free** stability margin, $K'_n = -\frac{dC_m}{dC_{L_w}} = h'_n - h$
where h'_n is the controls **free** neutral point,

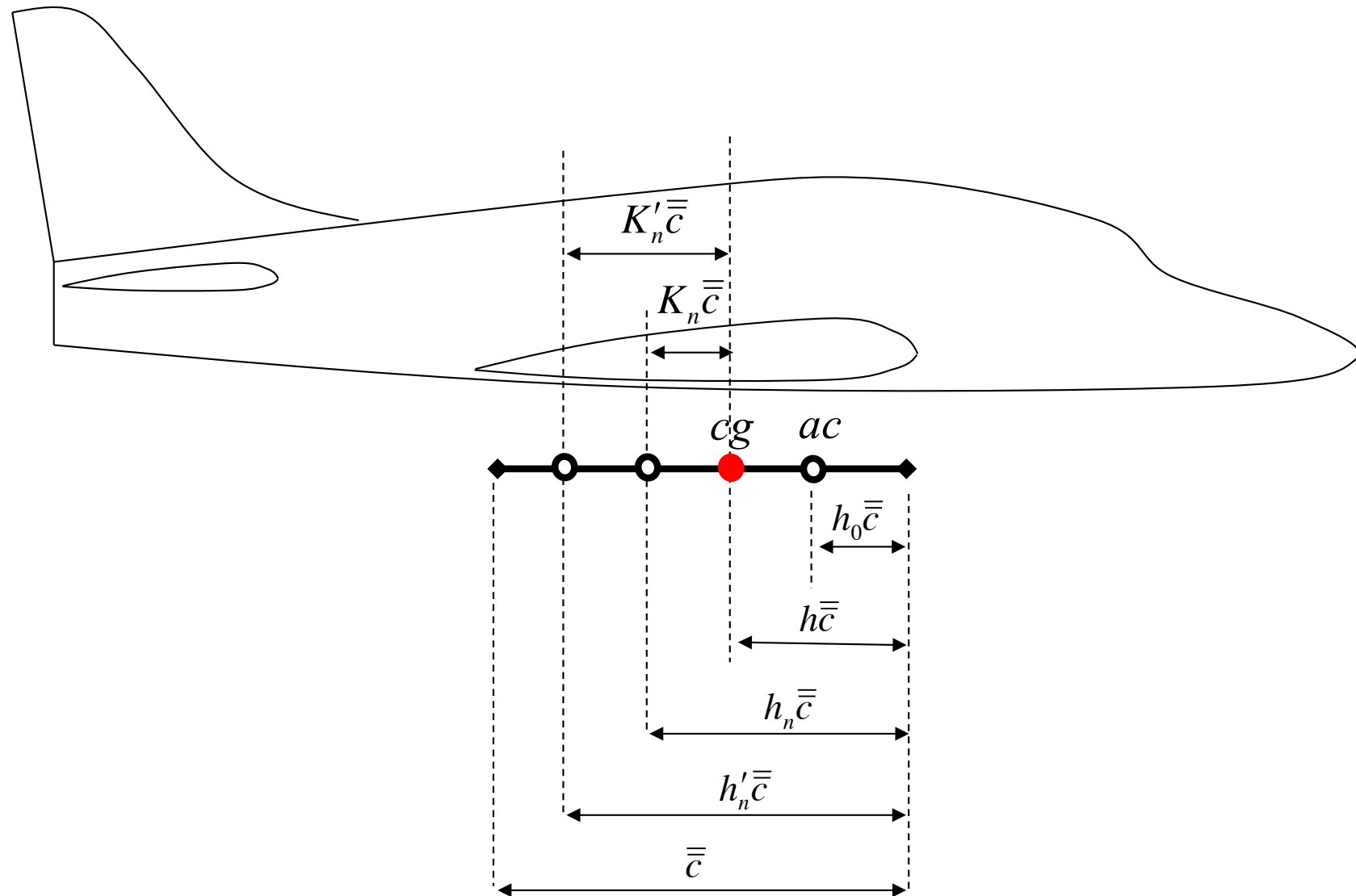
$$h'_n = h_0 + \bar{V}_T \frac{a_1}{a} \left(1 - \frac{d\varepsilon}{d\alpha}\right) \left(1 - \frac{a_2 b_1}{a_1 b_2}\right)$$

or

$$h'_n = h_n - \bar{V}_T \frac{a_2 b_1}{a b_2} \left(1 - \frac{d\varepsilon}{d\alpha}\right)$$

- As for fixed controls, $K'_n \geq 0 \rightarrow$ stable aircraft
- Centre of gravity must be ahead of the controls free neutral point to have a stable aircraft
- Usually, $h'_n > h_n$
- An aircraft that is stable with controls fixed is usually also stable controls free

Summary of longitudinal stability

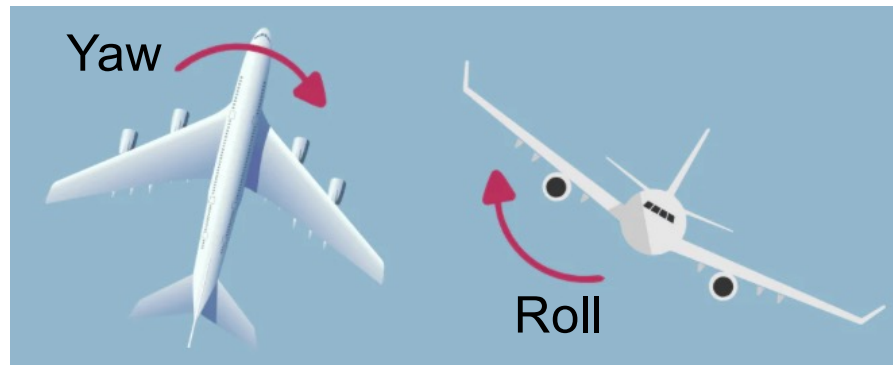


Lateral stability



Lateral flight → flow unsymmetric (unlike longitudinal flight)

Two types of lateral motion : Roll & Yaw



Roll and yaw are **always coupled**, because:

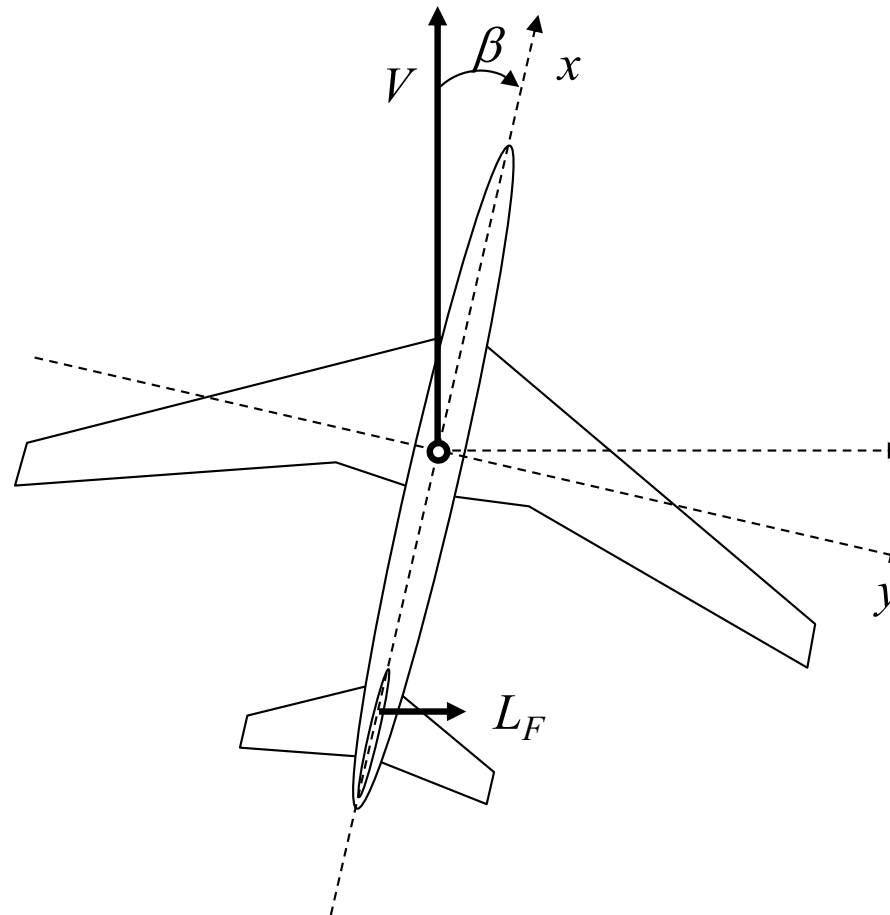
- Rolling produces sideslip
- Ailerons cause adverse yaw
- Dihedral
- Wingtip vortices
- Sweepback of wings
- Fin

Yaw stability



Yaw angle β induces a dissymmetry on the aircraft

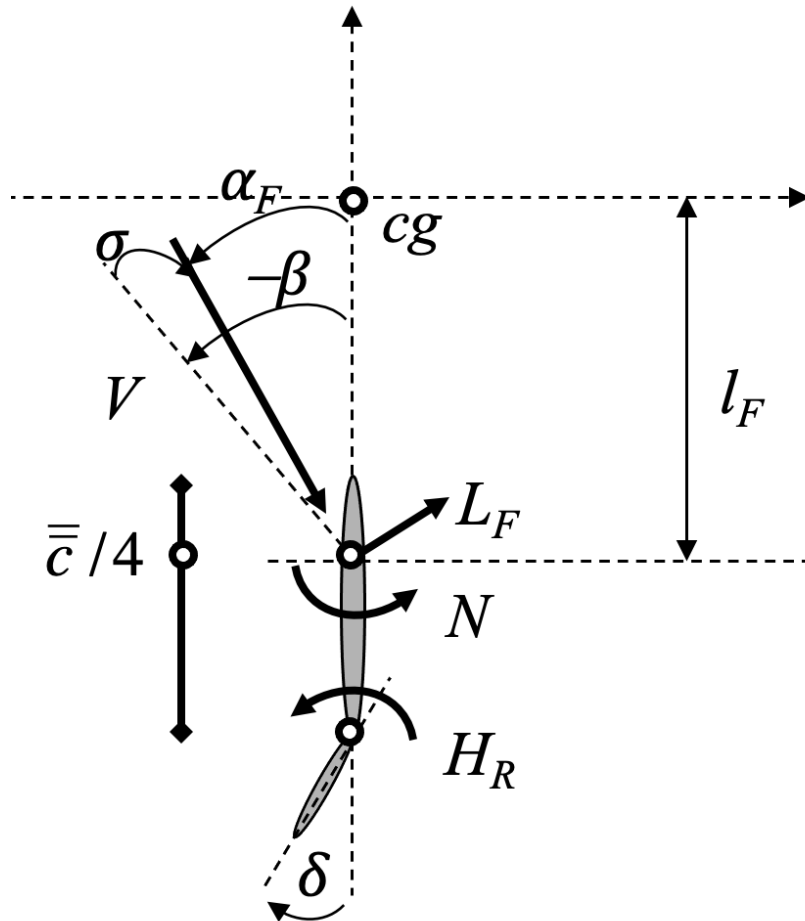
Lift on the vertical stabilizer (L_F) producing a stabilizing moment around z axis



Yaw stability



Lift coefficient of the fin : $C_{L_F} = c_1 \alpha_F + c_2 \delta = c_1 (-\beta + \sigma) + c_2 \delta$



σ = the sidewash velocity

local windspeed component induced by the effect of the fuselage, wing and possibly propellers.

Moment coefficient of the fin: $C_n = C_{L_F} \bar{V}_F$, where $\bar{V}_F = \frac{S_F l_F}{S \bar{c}}$

Yaw stability



Stability condition for yaw : $\frac{dC_n}{d\beta} < 0$

which is equivalent to $\bar{V}_F \left(-c_1 + c_1 \frac{d\sigma}{d\beta} + c_2 \frac{d\delta}{d\beta} \right) < 0$

In this case, it makes no sense to differentiate the yawing moment by the lift since the two are independent

The sidewash factor $d\sigma/d\beta$, is very difficult to estimate

Three main contributions to the sidewash:

- **Fuselage**, which acts as a lifting body when at a yaw angle
- **Wing**, around which the flow is asymmetric. The resulting sidewash is more pronounced for low AR sweptback wings
- **Propeller**, which creates more asymmetric in its wake

Yaw control



Yaw angle must be zero during most flight conditions to minimize drag

Rudder deflection used to control yaw

Rudder power = rate of change of fin moment
with rudder angle

$$= \frac{dC_n}{d\delta} = \bar{V}_F \frac{dC_{LF}}{d\delta} = c_2 \bar{V}_F$$

This quantity must be large enough to maintain zero yaw in the most extreme flight conditions.

Yaw control



During cruise, aircraft tend to turn towards the wind to minimize drag.

The objective is to achieve zero yaw.

At take-off and landing, this is not possible :

The aircraft must be aligned with the runway even in presence of a very strong sidewind.

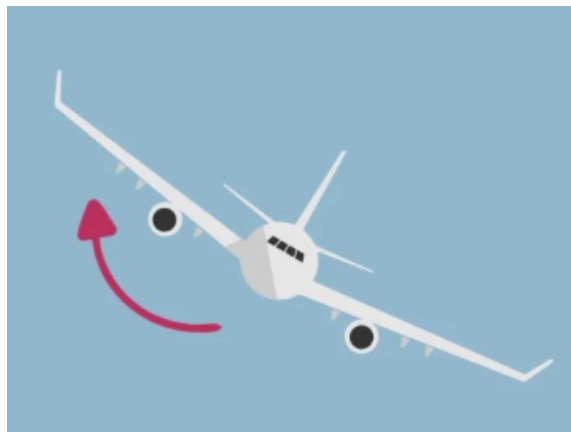
→ Rudder must be able to provide enough moment to keep the aircraft aligned with the runway

Roll stability



- Φ rotation around longitudinal axis of the aircraft
- **No active** mechanism for roll control

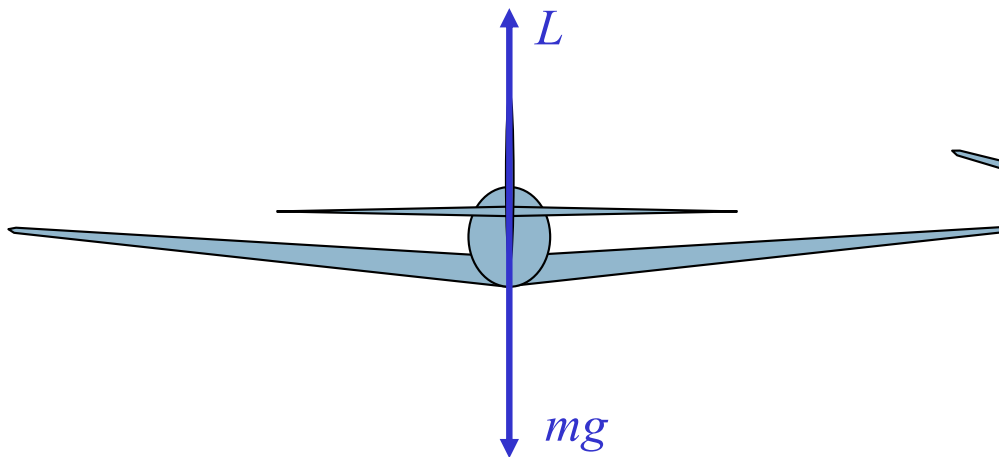
(such as tailplane/elevator for longitudinal stability or rudder for yaw stability)



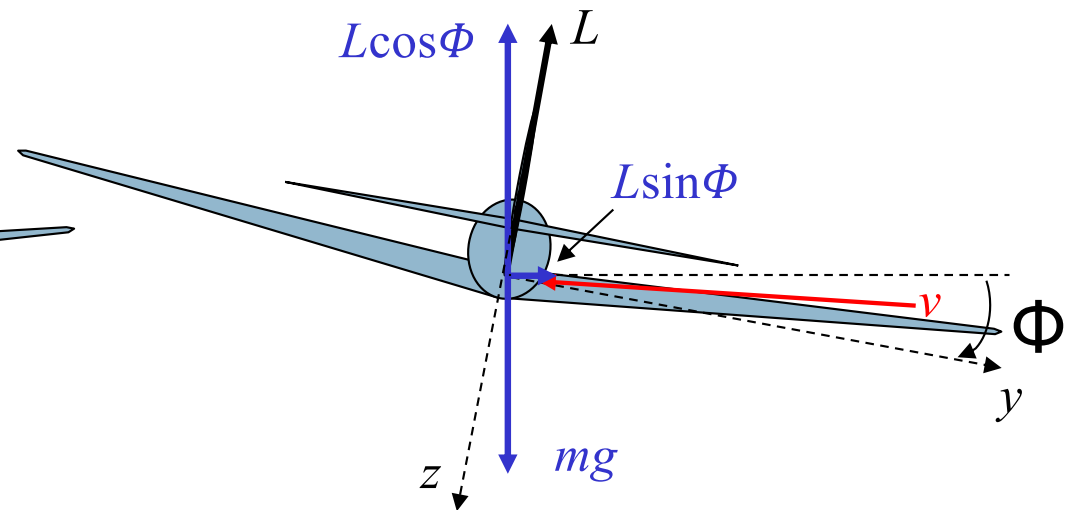
Several effects:

- Wing: Dihedral / Anhedral, Sweep
- Fuselage
- Vertical fin

Roll motion



Steady level flight: $L=mg$



At roll angle Φ

→ Lift is still perpendicular to wings and equal to mg

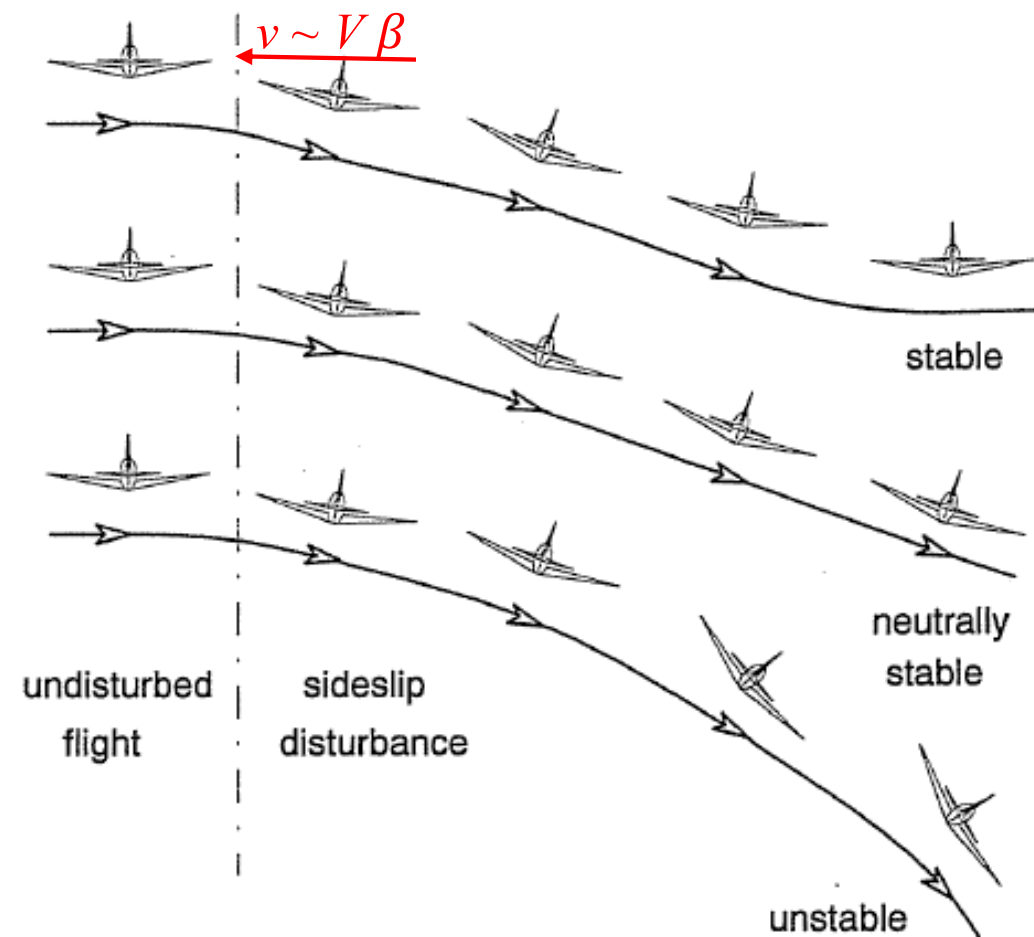
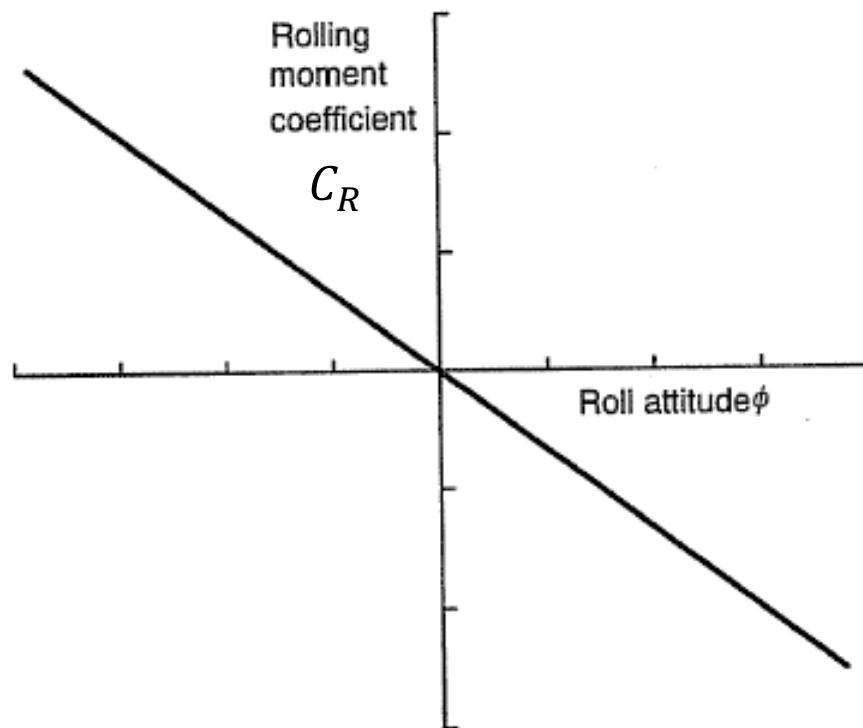
But along :

- Vertical axis : $L\cos\Phi < mg$ → Aircraft moves down
- Horizontal axis : $L\sin\Phi$ → Aircraft drifts (sideslip β)

Roll stability



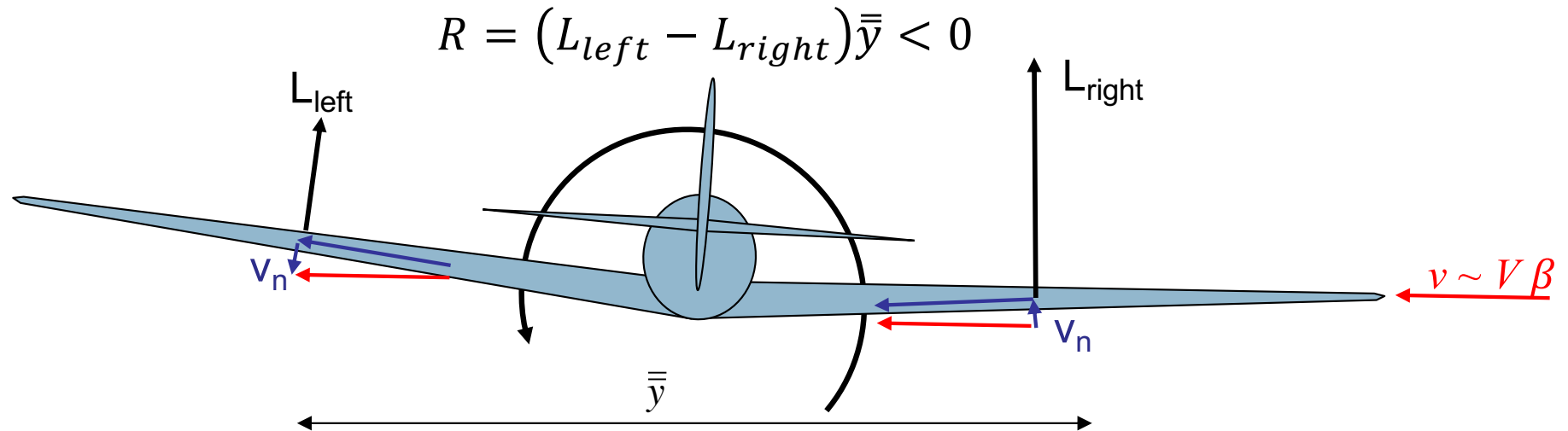
For a stable aircraft in roll $= \frac{dC_R}{d\phi} < 0$



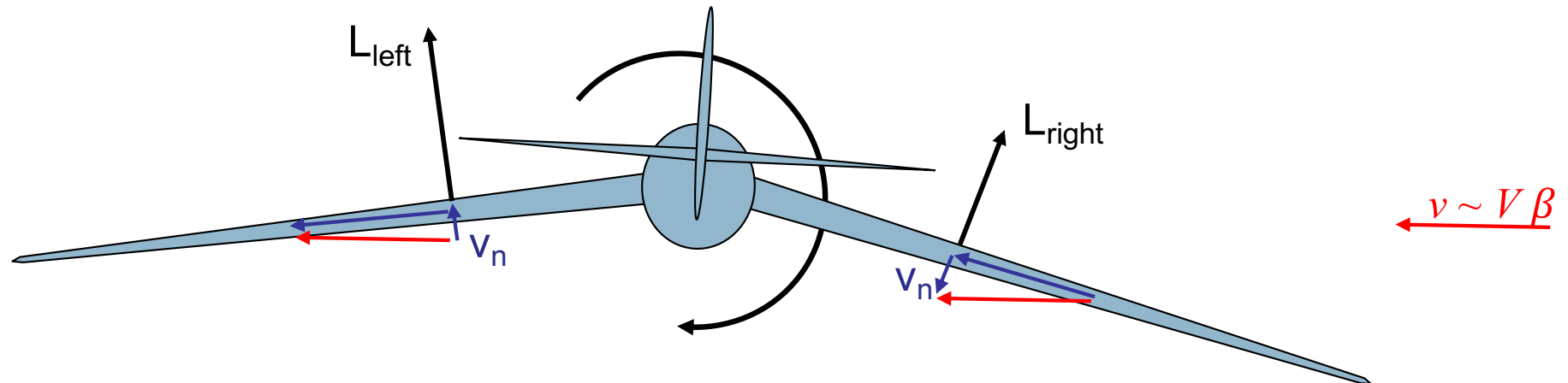
Dihedral / Anhedral



DIHEDRAL → Restoring/stabilising moment R



ANHEDRAL → Destabilising moment R $(L_{left} - L_{right})\bar{y} > 0$



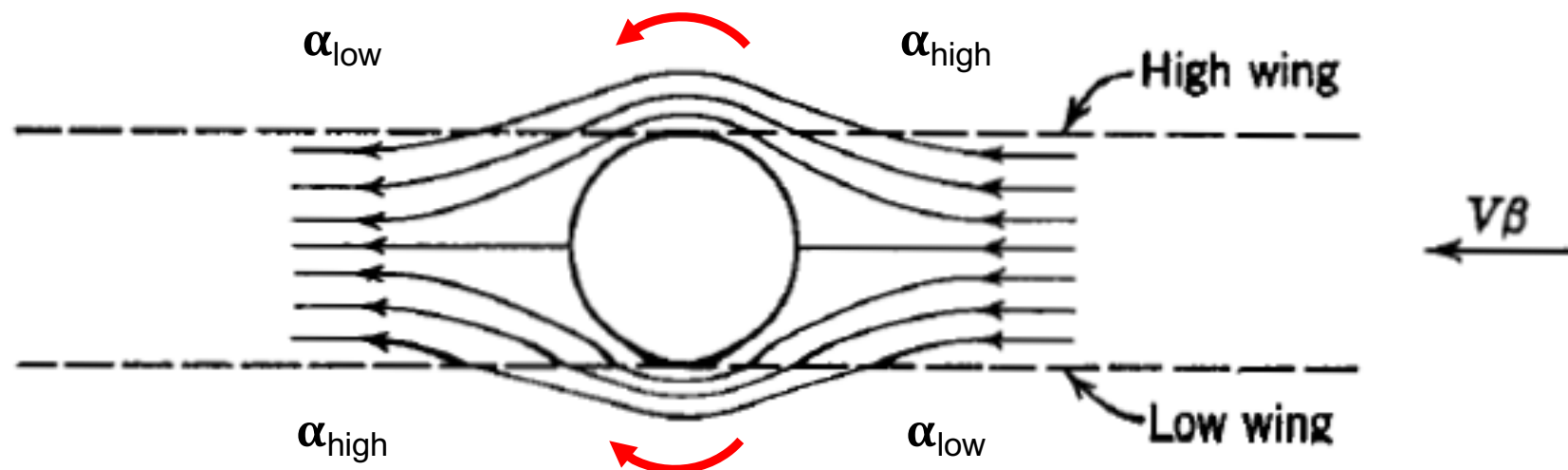
Effect of fuselage



Fuselage strongly impacts the flow around the wing

→ The position of the wing on the fuselage has a major effect:

- High wing → stabilising moment
- Low wing → destabilising moment

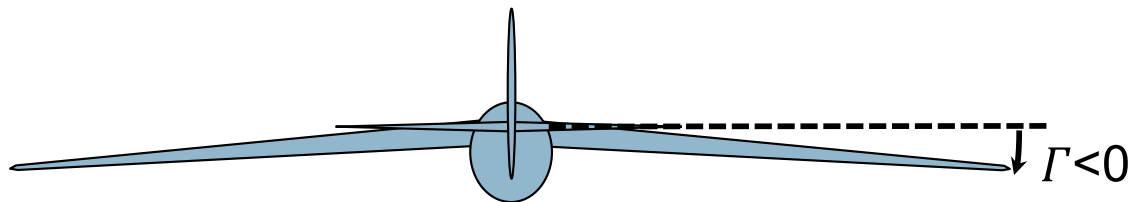


Wing position and dihedral



High wing aircraft: very stable (too much)

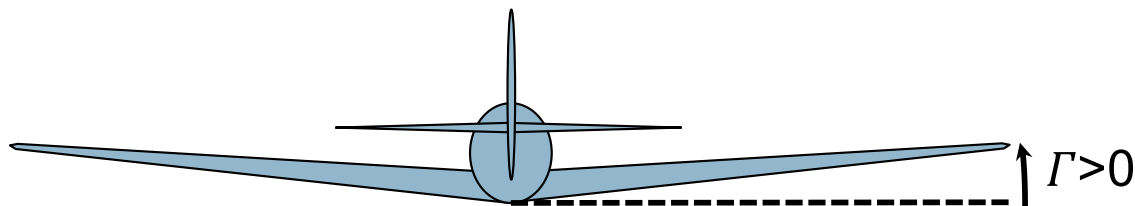
→ **Negative** dihedral (anhedral) can be used to **reduce stability** and **increase manoeuvrability**



Harrier

For **Low** wing aircraft: less stable

→ **Positive** dihedral can be used to **increase stability**



A350

Sweepback effect on roll



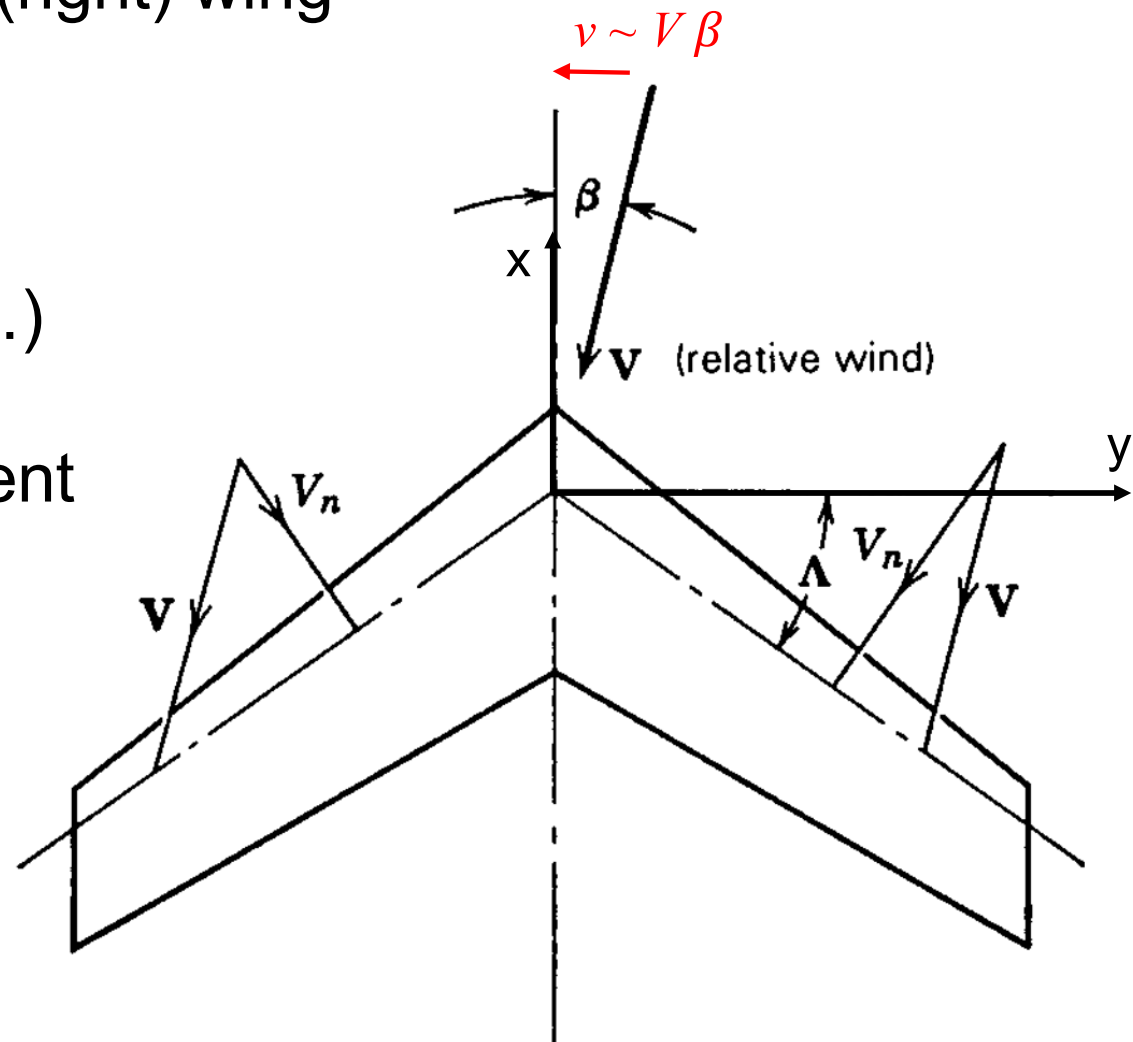
Normal velocity component (v_n) different on both sides :

Larger on the windward (right) wing
than leeward (left) wing

→ Larger lift (and drag ...)

→ Negative rolling moment

→ Stabilizing effect

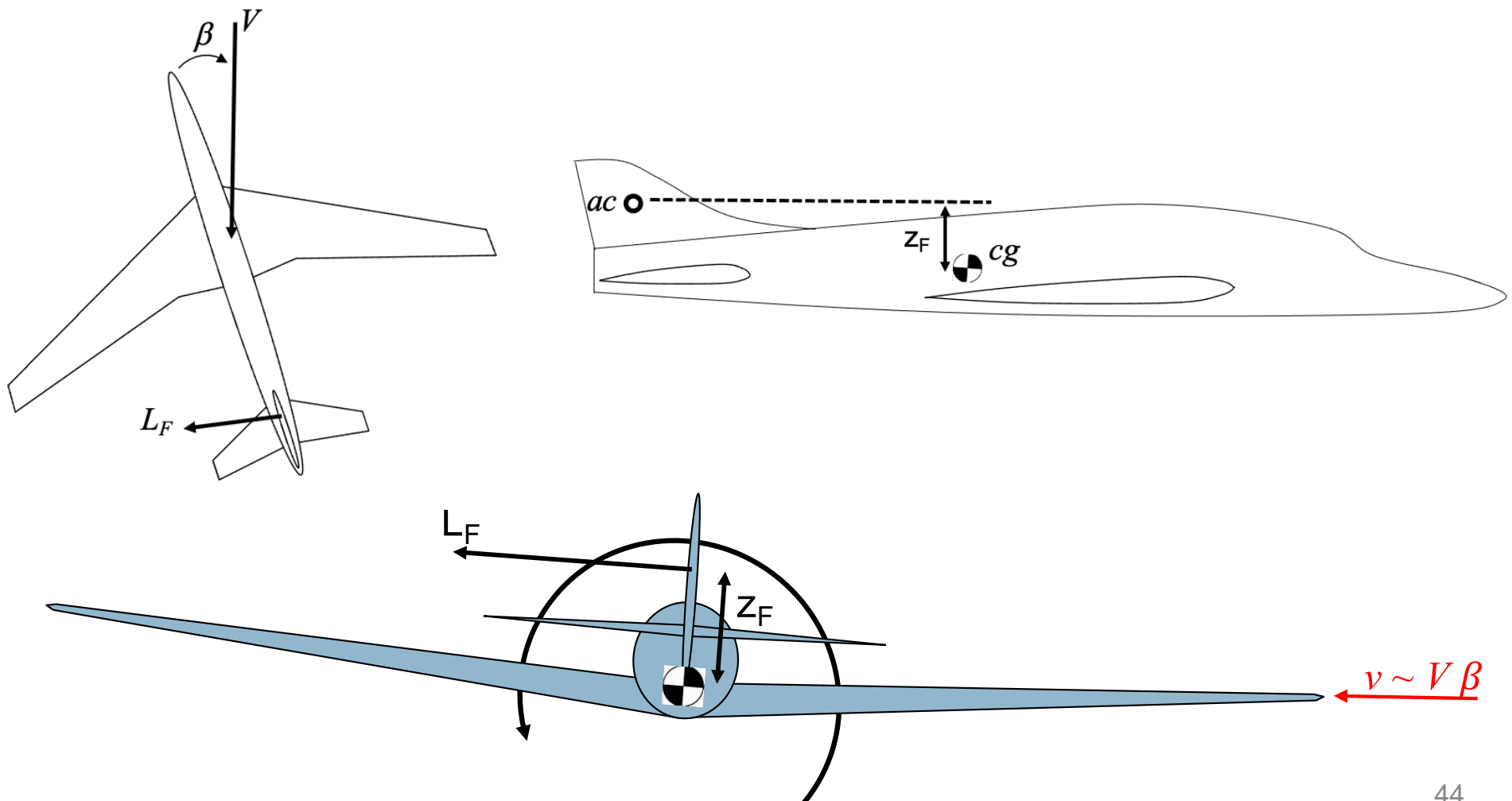


Vertical tail effect on roll

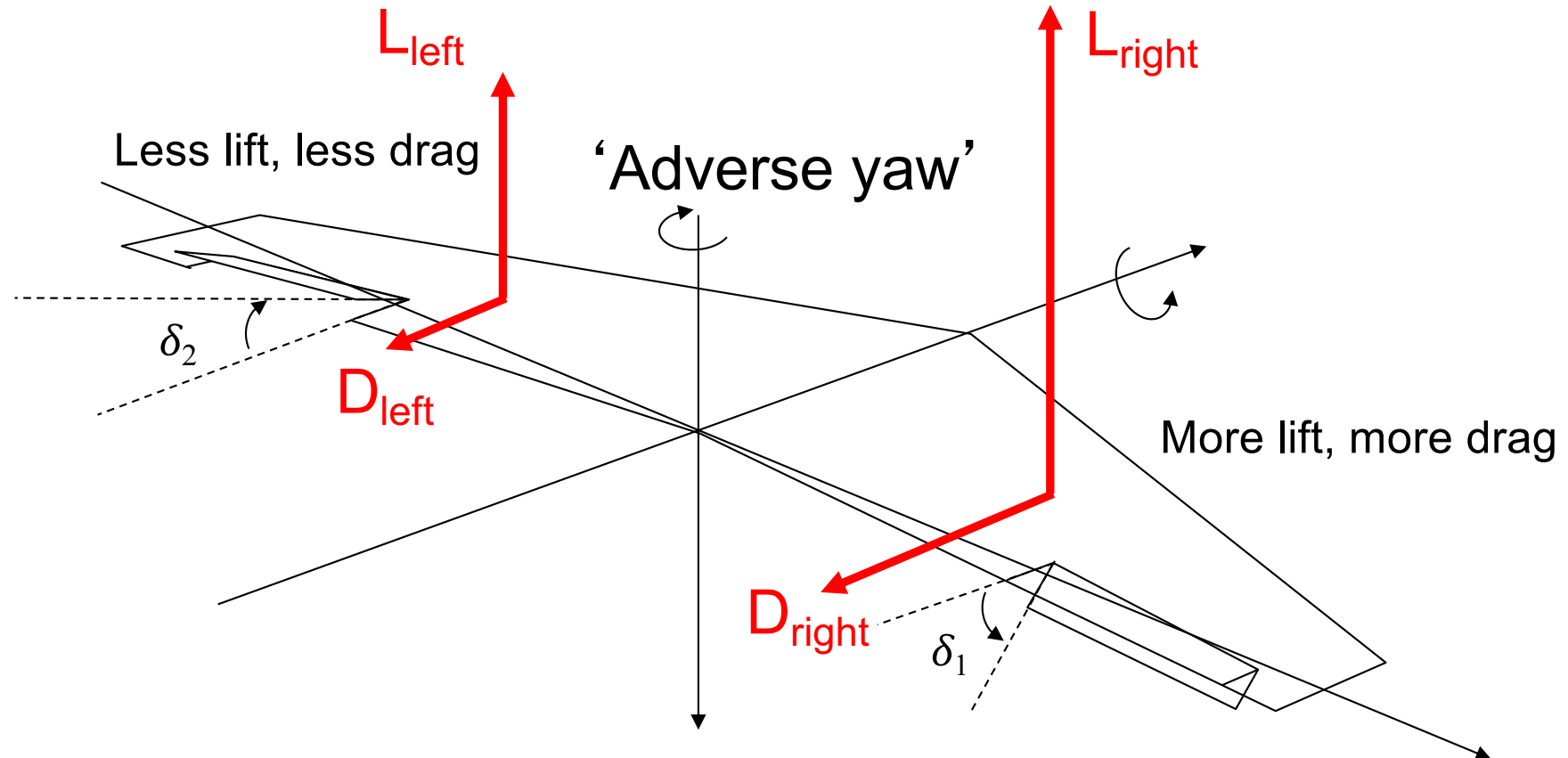


Sideslip (β) creates lift (L_F) on the fin

→ Rolling moment around the cg : $R = z_F L_F < 0$



Roll control via ailerons



On the up/right wing: Increasing the lift increases drag

On the down/left wing: Less lift and drag decrease

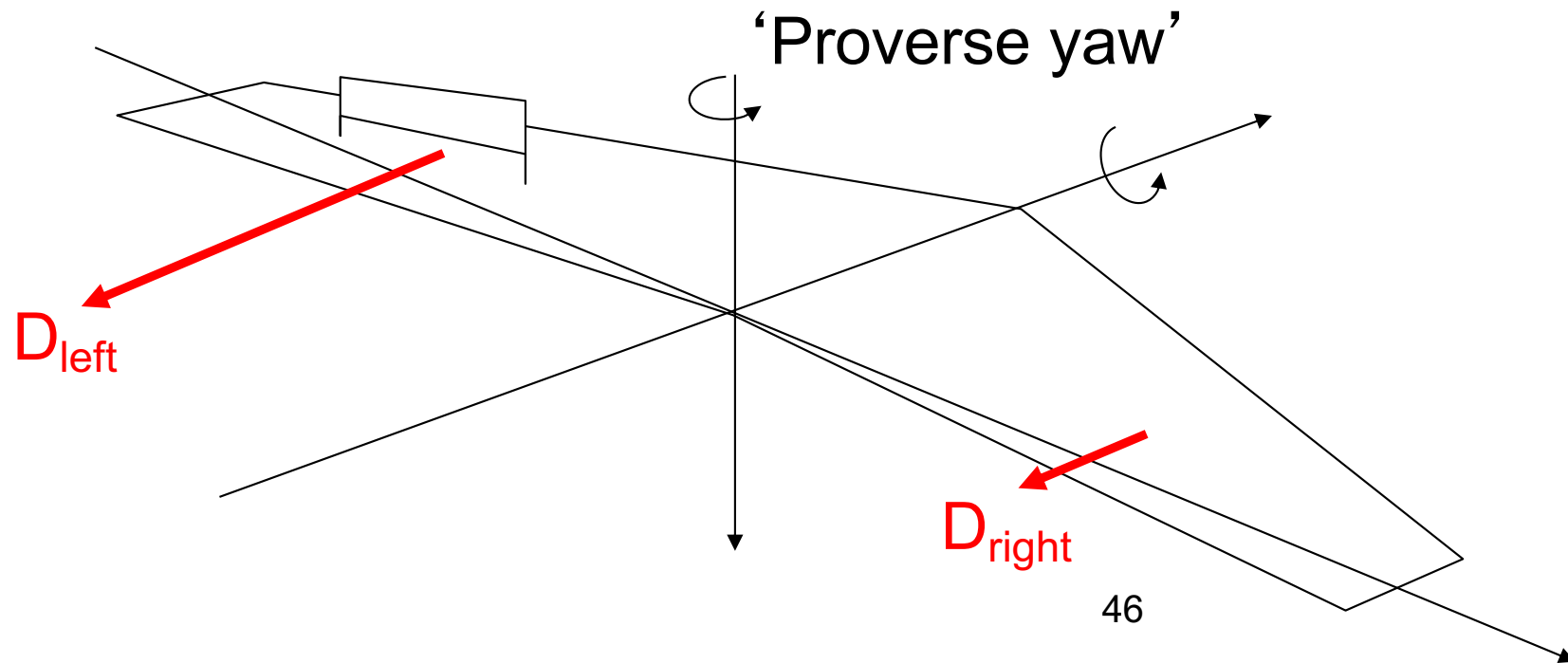
→ **Adverse yaw**

When rolling/turning left, there is a yaw moment to the right

Roll control via spoilers



Deforming a spoiler on the wing towards which we want to turn



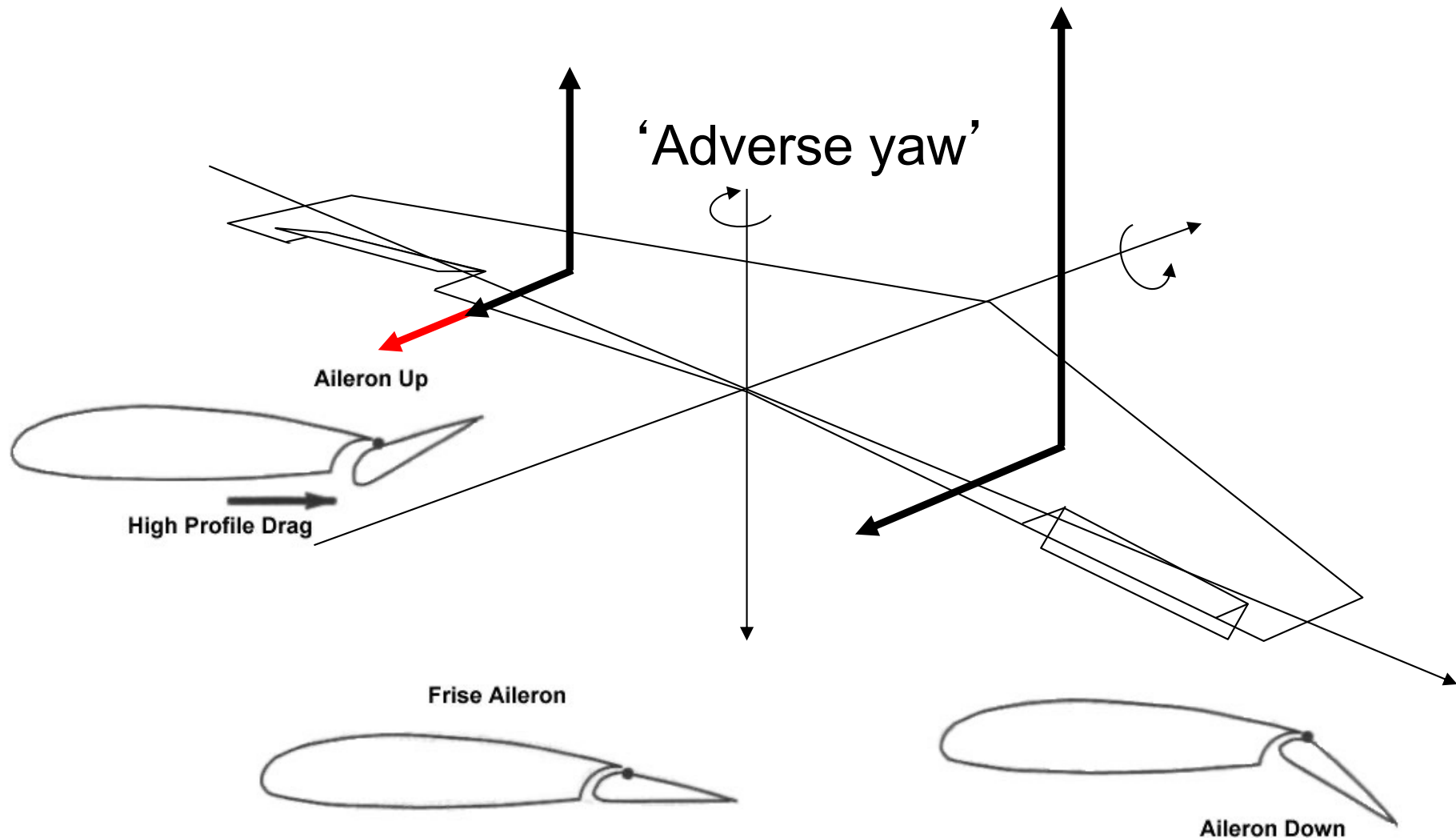
The spoiler decreases lift and increases drag.

→ Resulting yawing moment in the same direction as the roll
(Proverse yaw)

Frise ailerons



Increase (profile) drag on the down/left wing to counteract the large drag of the other wing

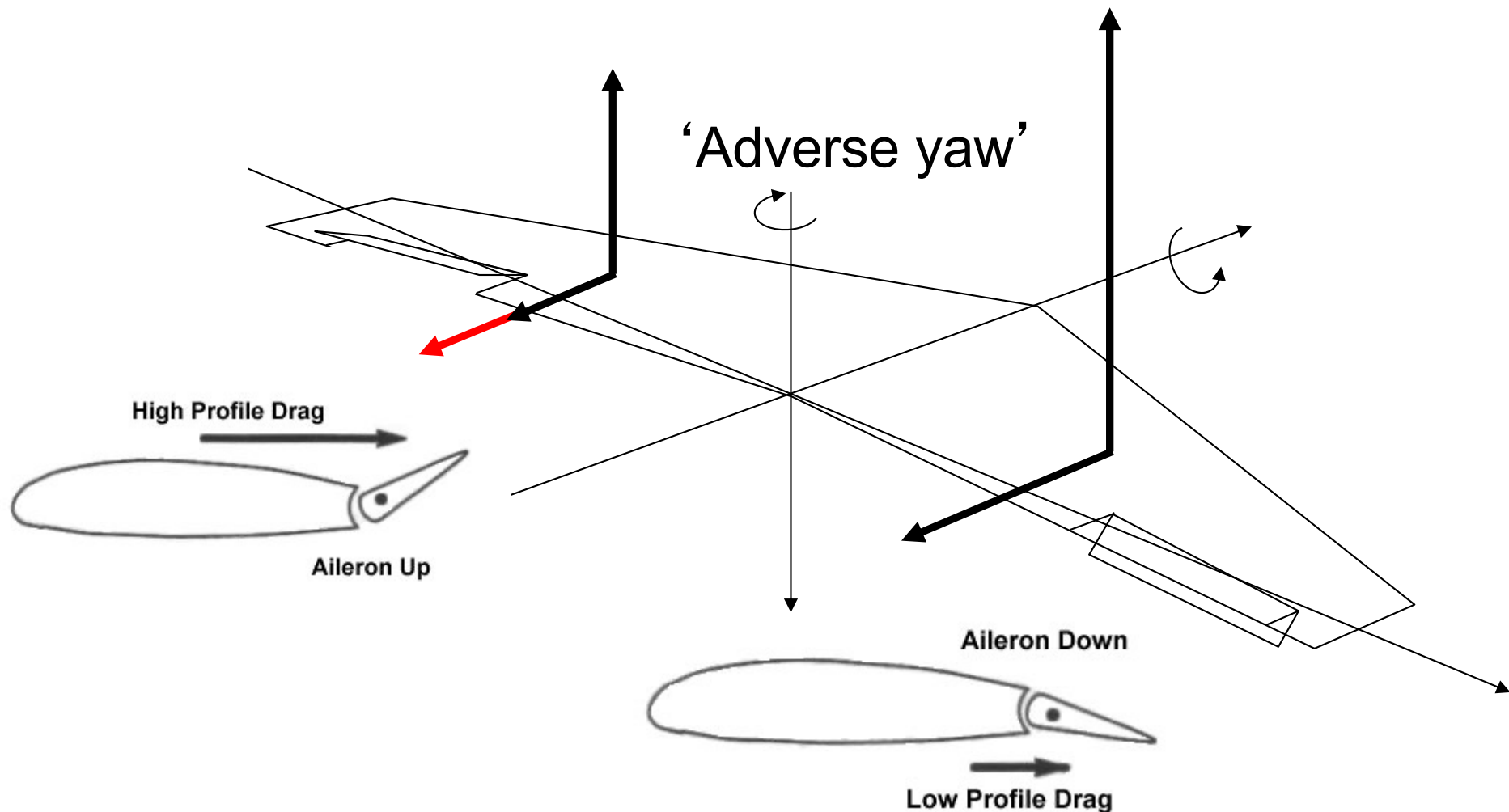


Differential aileron deflection



The roll rate of the aircraft depends on the mean aileron deflection angle.

The individual deflections δ_1 and δ_2 do not have to be equal



Summary on control surfaces:



Elevators: contribute to pitch stability and control pitch angle

Rudder: contribute to yaw stability and control yaw angle

Ailerons: do not contribute to stability but control roll rate, not roll angle.

Elevons = ailerons that can also move up or down in unison

Flaperons = ailerons moving downwards only (like flaps)

Spoilerons = ailerons that can also move upwards only (like spoilers)

→ contribute to stability and control both pitch and roll

Dynamic stability



A flying aircraft has several **modes of vibration**:

- **Longitudinal modes:**
 - Short period oscillation
 - Long period oscillation (Phugoid)
 - **Lateral modes**
 - Spiral mode
 - Roll subsidence
 - Dutch roll
- These dynamic modes must be considered

Short period oscillations



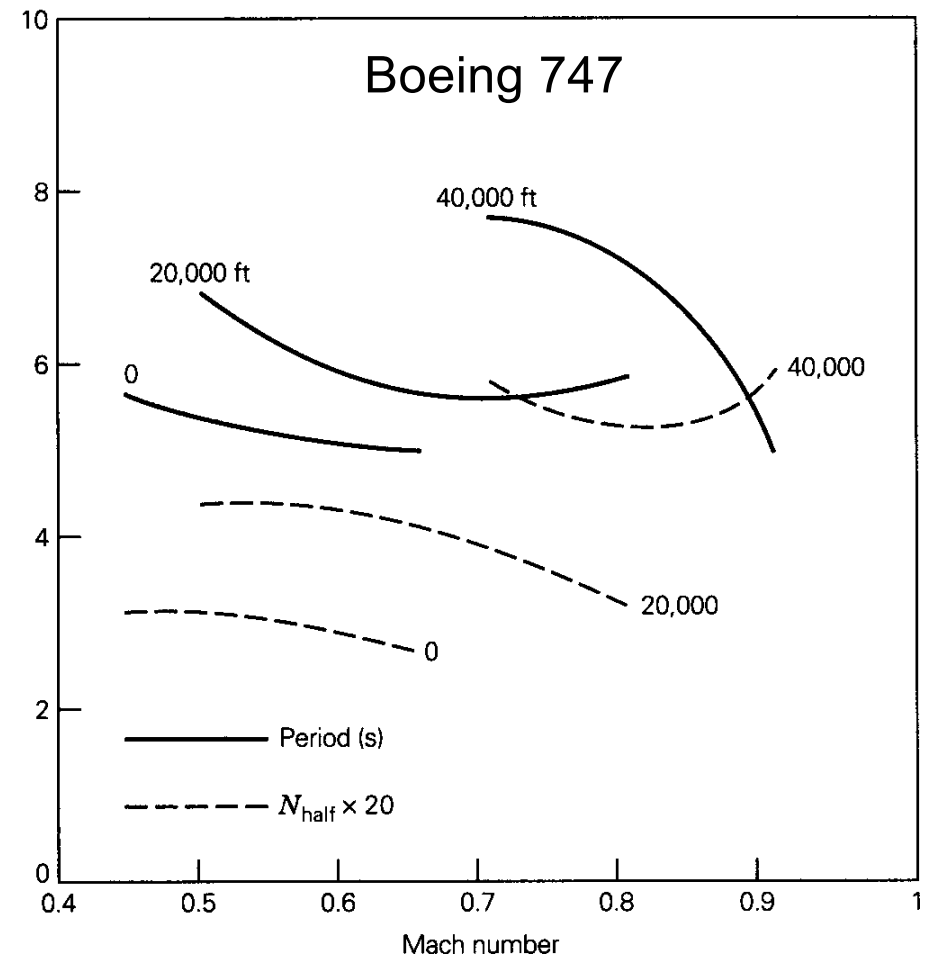
Driven by angle of attack

Take place because of abrupt input changes

Speed changes are negligible

Period of oscillation decreases
with airspeed/Mach

N_{half} = number of periods until
the amplitude is halved



Phugoid



= Long period oscillations in longitudinal direction

Angle of attack is constant: aircraft climbs and descends in an oscillatory manner

Low damping of the motion

Phugoid period:

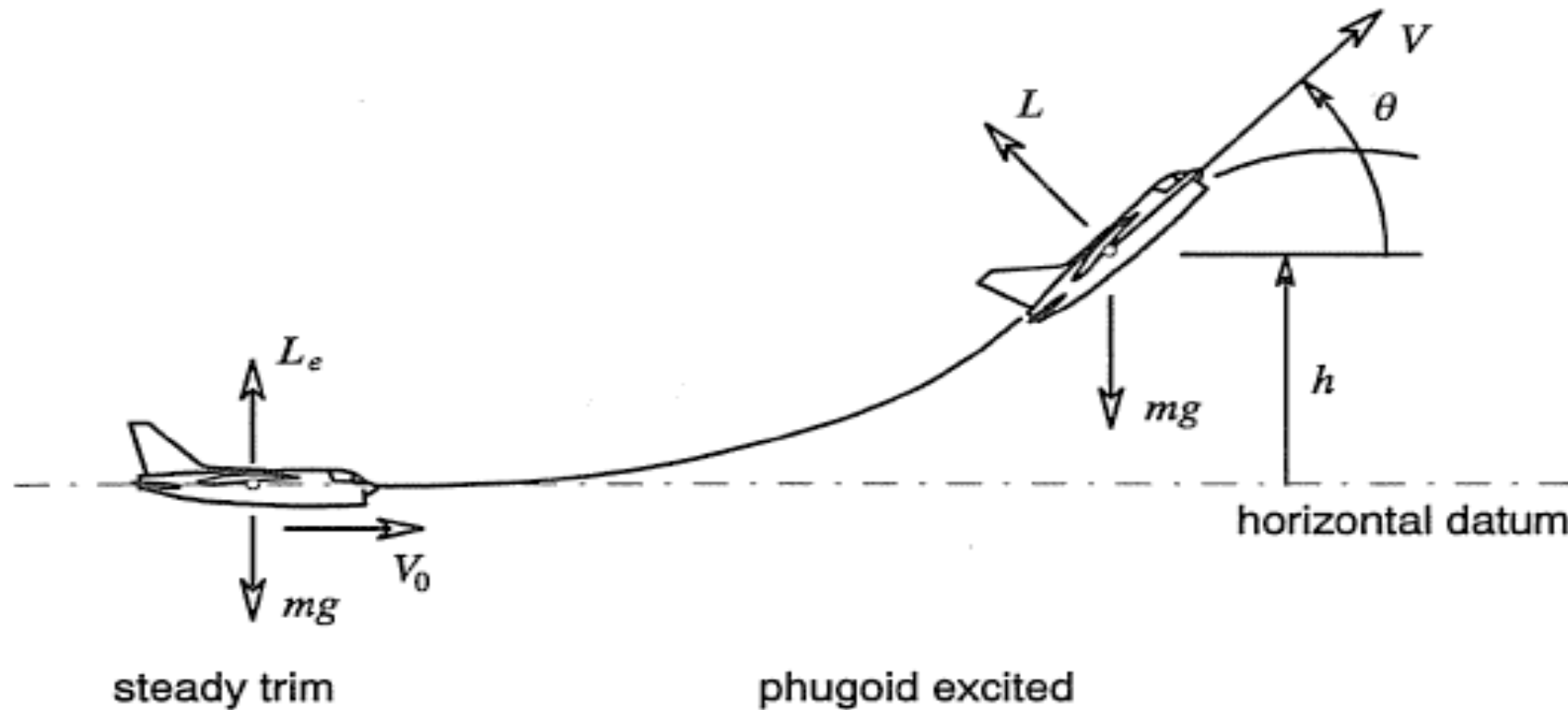
- microlight aircraft: 15-25s
- light aircraft: overs 30s
- jet aircraft: minutes

Neutralized by re-trimming the aircraft in a new flight configuration

Phugoid videos



Phugoid estimates



Lanchester approximations of damping and frequency:

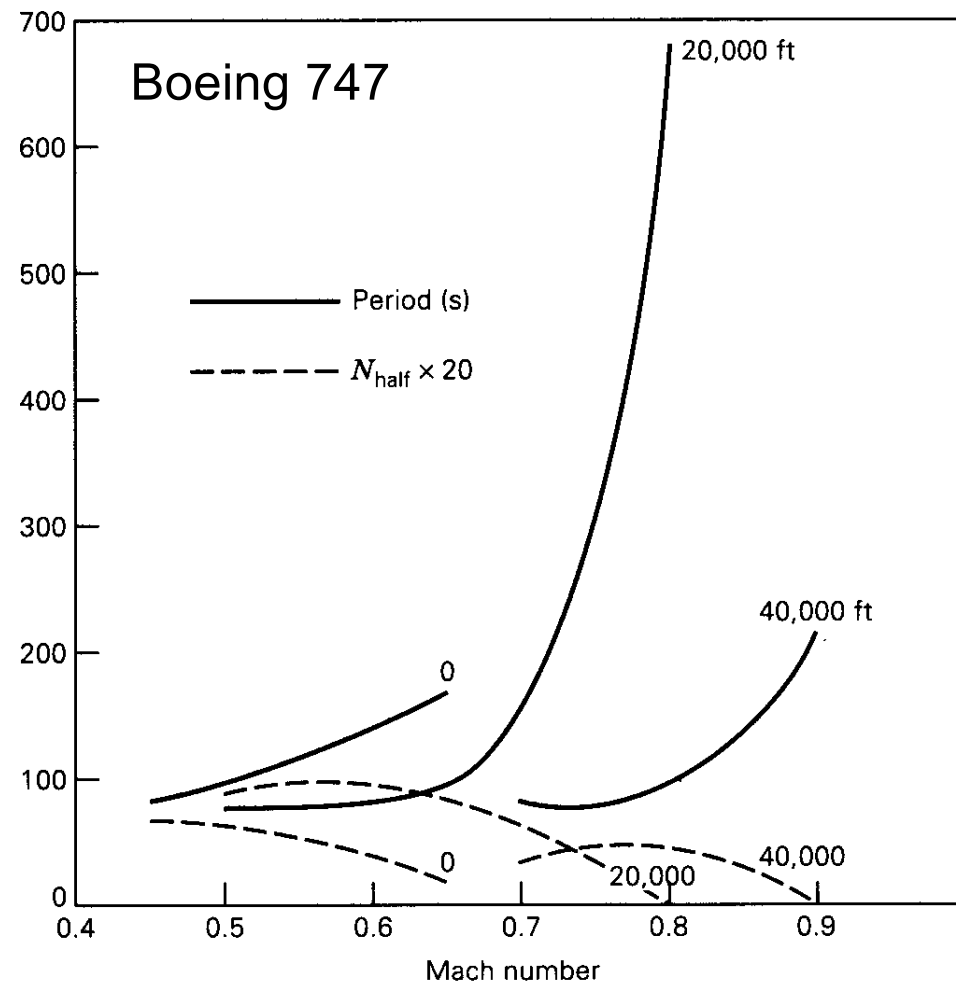
$$\zeta_p \omega_p = \frac{g C_D}{C_L V_0}, \quad \omega_p = \frac{g \sqrt{2}}{V_0}$$

Phugoid



Phugoid period increases with airspeed

Phugoid damping slightly increase with airspeed



Spiral mode



= yaw movement with a little of roll

Spiral mode can be stable or unstable

Non-oscillatory mode with large time constant

Typical half-life of spiral mode ~ a minute

Spiral movement is usually stopped by a corrective control input

Spiral mode video



Roll subsidence



An impulse aileron → initiate aircraft rolling

In general the aircraft will stop rolling with time
(i.e. the roll rate becomes zero after sufficient time)

The aircraft will find itself at a roll angle which depends on how fast the roll rate tends to zero.

This phenomenon is called roll subsidence.

Roll subsidence video



Dutch roll



Dutch roll = Roll-Yaw coupling

Roll stability is stronger than Yaw stability because wing is bigger than the fin

Name from ice skating in Holland

Solution :

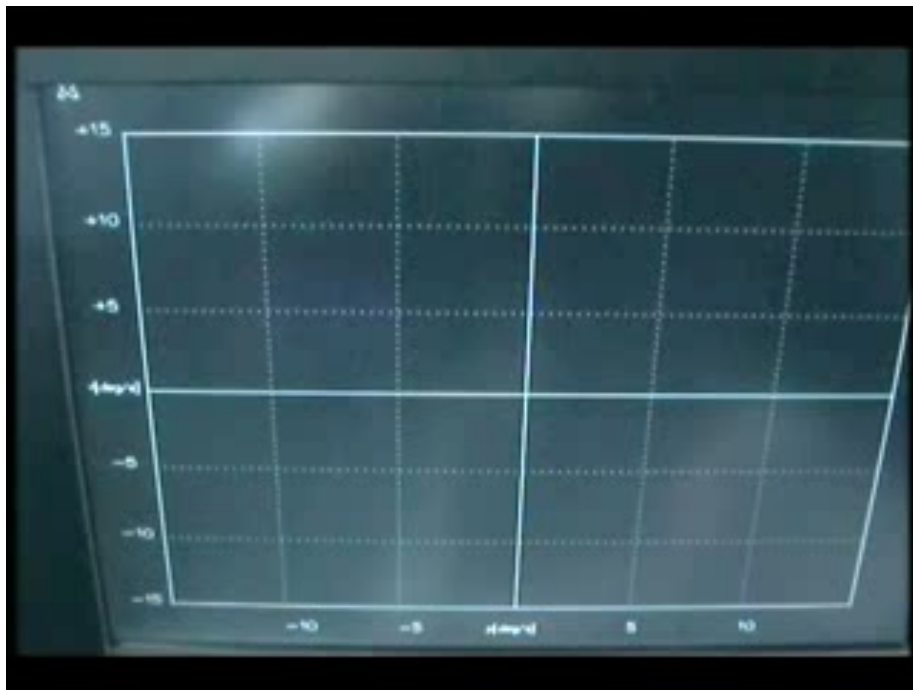
Yaw damper = computer connected to the rudder to mitigate yaw motion

Dutch roll dangerous ? No, but : there were incidence due to improper reaction by pilots (putting rudder input too late)

Dutch roll video



Dutch roll video



Lateral modes of Boeing 747



Altitude	Mach	Spiral	Dutch Roll	
		Half-life	Period	N_{half}
0	0.45	35.7	5.98	0.87
0	0.65	34.1	4.54	0.71
20,000	0.5	76.7	7.3	1.58
20,000	0.65	64.2	5.89	1.33
20,000	0.8	67.3	4.82	1.12
40,000	0.7	-296	7.99	1.93
40,000	0.8	94.9	6.64	3.15
40,000	0.9	-89.2	6.19	1.18

Summary



Stability = ability to **keep** the aircraft in the air in the chosen flight attitude

→ Aircraft returns to the equilibrium position after a perturbation.

→ Stability must be ensured but not oversized !

Control = ability to **change** the flight direction and attitude of the aircraft

Static stability in :

- Pitch → Longitudinal
- Yaw/Roll → Lateral

Dynamic stability to avoid dangerous phenomena