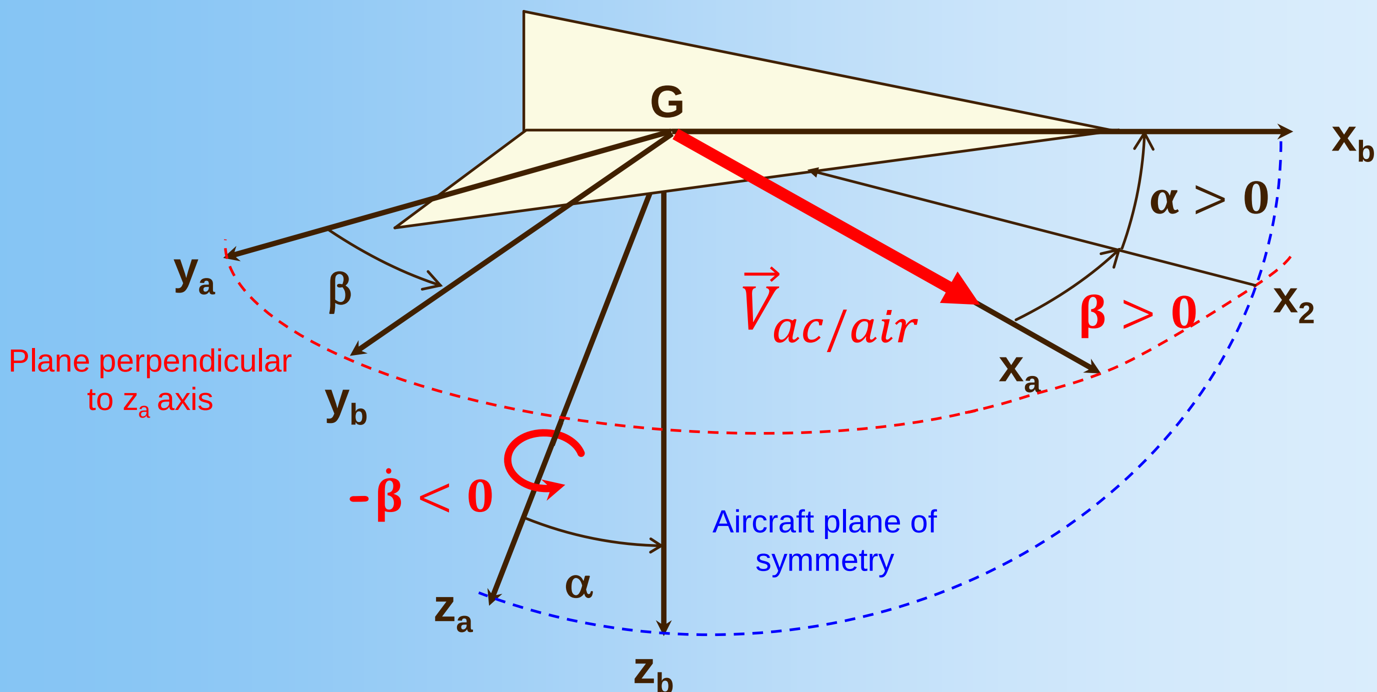
Lateral Aerodynamics

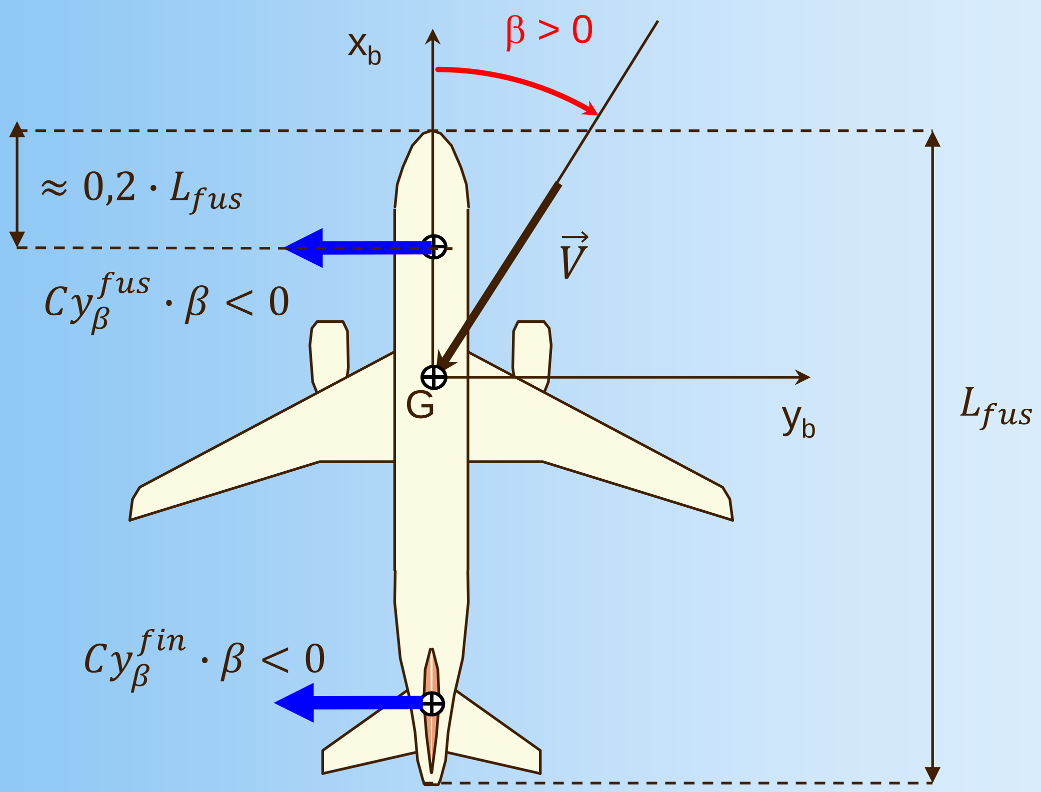
Airbus Beluga

Definition of the Side Slip angle β

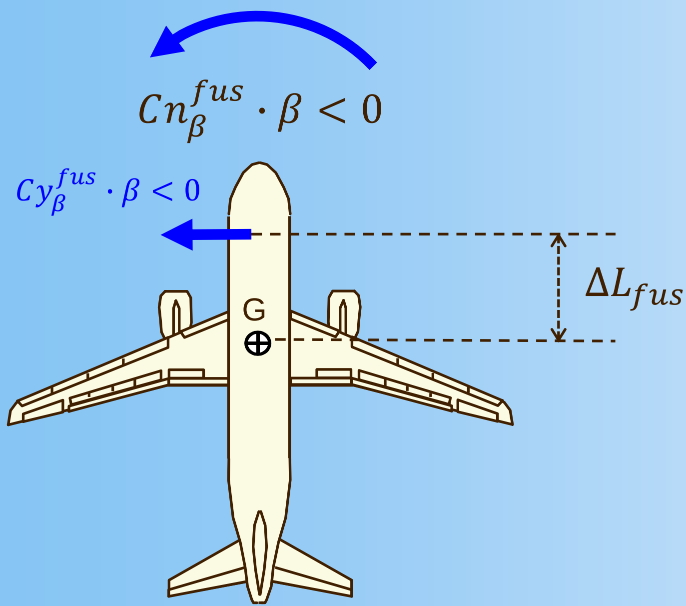


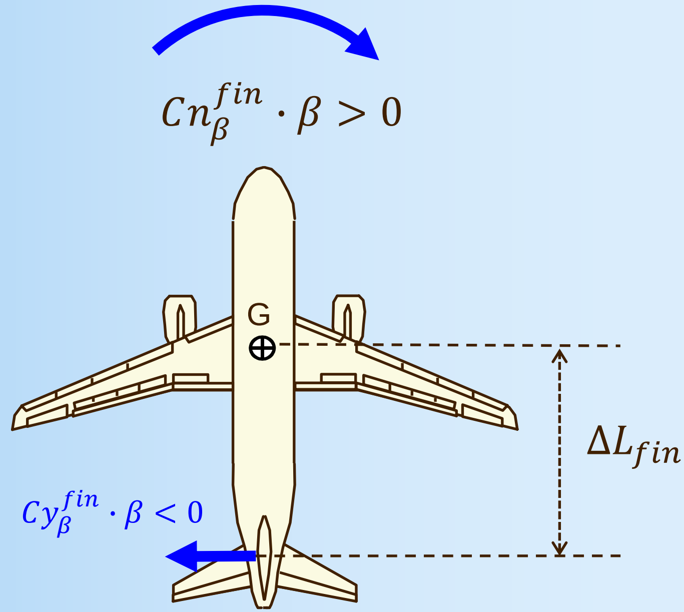
The Side slip angle β is the angle between $\vec{V}_{ac/air}$
and the Aircraft plane of symmetry
Positive when the flow field is coming from the right

Fuselage + Fin effect : Cy_β / Cn_β



Fuselage + Fin effect : Cy_β / Cn_β





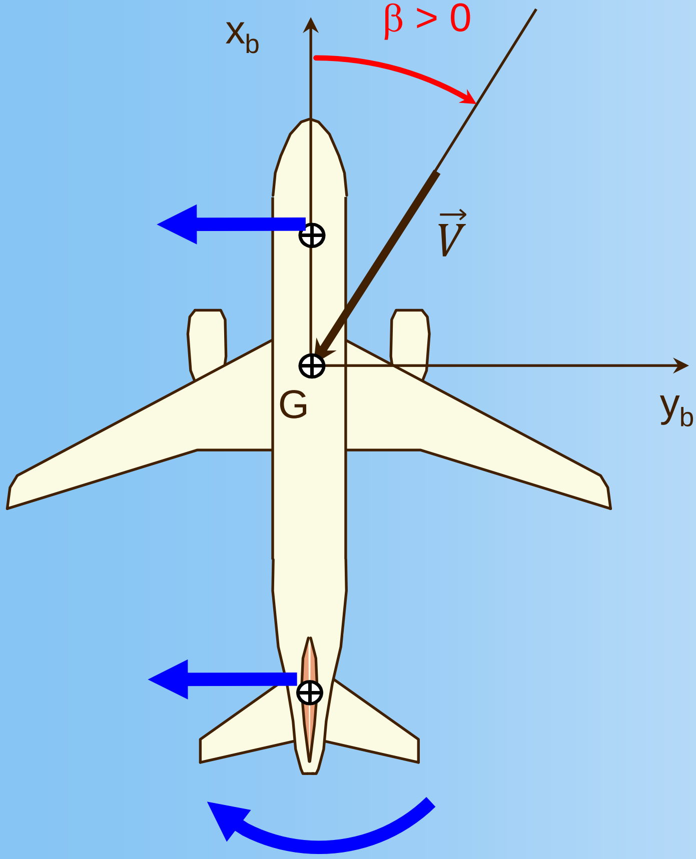
$$SL \cdot Cn_\beta^{fus} \cdot \beta = S \cdot Cy_\beta^{fus} \cdot \beta \cdot \Delta L_{fus}$$

$$SL \cdot Cn_\beta^{fin} \cdot \beta = -S \cdot Cy_\beta^{fin} \cdot \beta \cdot \Delta L_{fin}$$

$$\Rightarrow \quad Cn_\beta^{fus} = Cy_\beta^{fus} \cdot \frac{\Delta L_{fus}}{L} < 0$$

$$\Rightarrow \quad Cn_\beta^{fin} = -Cy_\beta^{fin} \cdot \frac{\Delta L_{fin}}{L} > 0$$

Aircraft lateral Stability

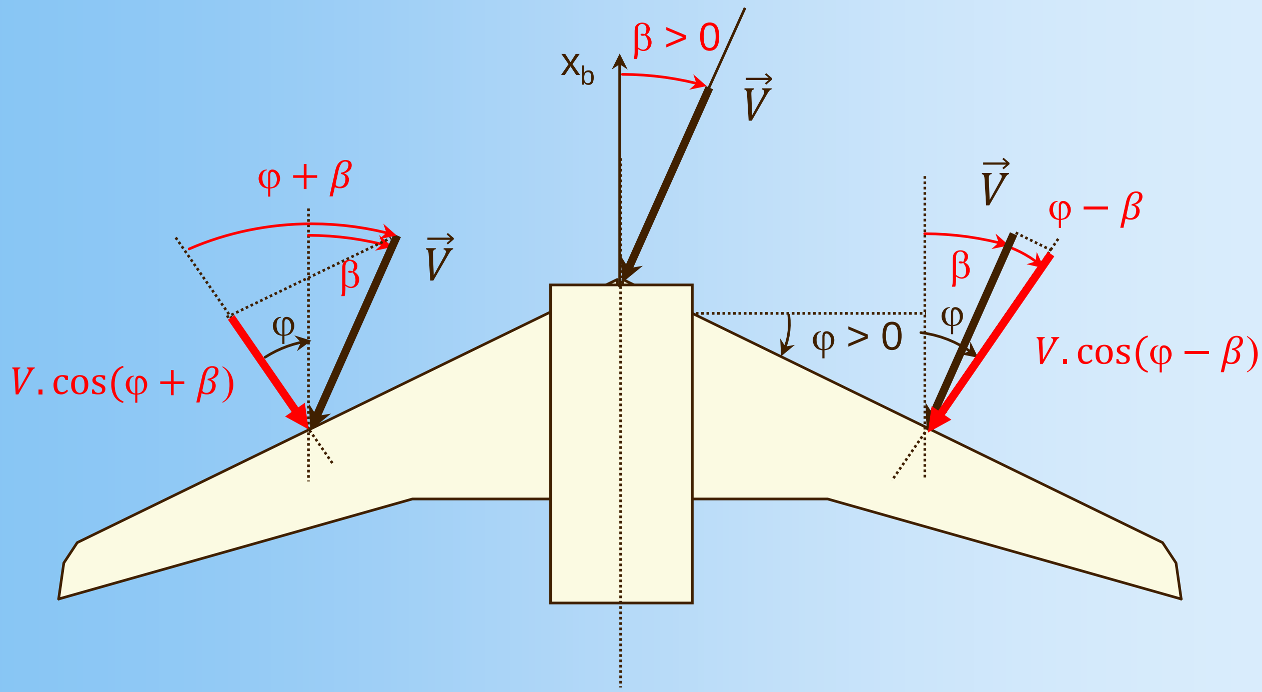


The positive $Cn_\beta^{a/c}$ reduces the side slip
the aircraft is stable

The aircraft is laterally stable
if $Cn_\beta^{a/c} > 0$

$$Cn_\beta^{a/c} = Cn_\beta^{fus} + Cn_\beta^{fin} > 0$$

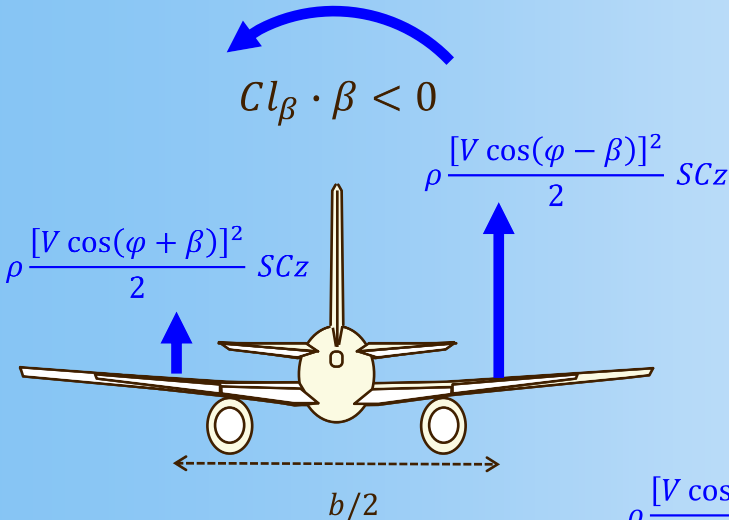
Wing sweep angle φ effect : Cl_β / Cn_β

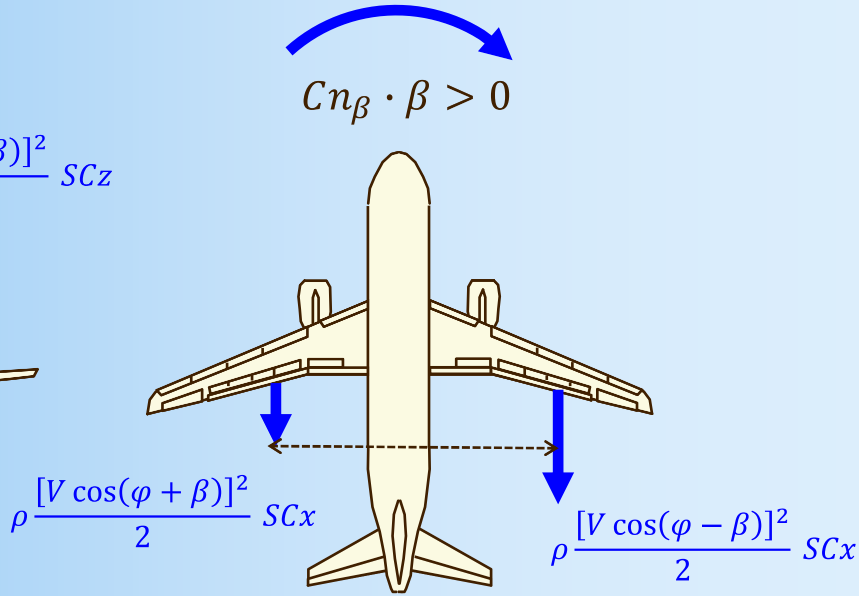


The right / left wing sees a modified normal velocity by : $V.\cos(\varphi - \beta)$ / $V.\cos(\varphi + \beta)$

➢ There is an unbalance in lift / drag forces

Wing sweep angle φ effect : Cl_β / Cn_β





$$SL \cdot Cl_\beta \cdot \beta = \frac{S}{2} \cdot Cz \cdot \left(\cos^2(\varphi+\beta) - \cos^2(\varphi-\beta)\right) \cdot \frac{b}{4}$$

$$SL \cdot Cl_\beta \cdot \beta = -\frac{S}{2} \cdot Cz \cdot 4\sin\varphi\cos\varphi \cdot \beta \cdot \frac{b}{4}$$

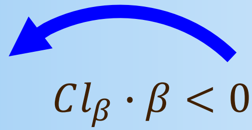
$$\Rightarrow \quad Cl_\beta = -\frac{1}{4} Cz \cdot \sin 2\varphi \cdot \frac{b}{L} < 0$$

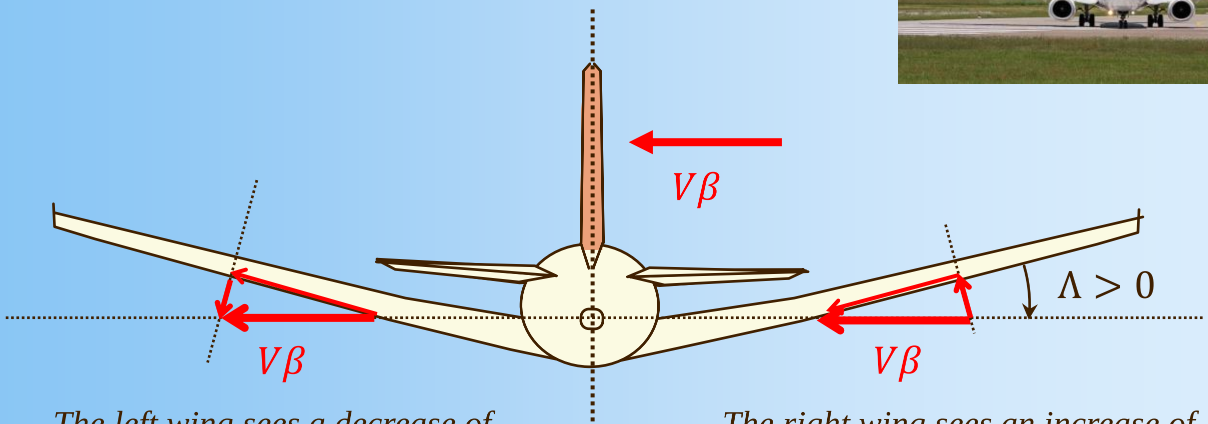
$$SL \cdot Cn_\beta \cdot \beta = \frac{S}{2} \cdot Cx \cdot \left(\cos^2(\varphi-\beta) - \cos^2(\varphi+\beta)\right) \cdot \frac{b}{4}$$

$$SL \cdot Cn_\beta \cdot \beta = \frac{S}{2} \cdot Cx \cdot 4\sin\varphi\cos\varphi \cdot \beta \cdot \frac{b}{4}$$

$$\Rightarrow \quad Cn_\beta = \frac{1}{4} Cx \cdot \sin 2\varphi \cdot \frac{b}{L} > 0$$

Wing dihedral angle Λ effect : Cl_{β}





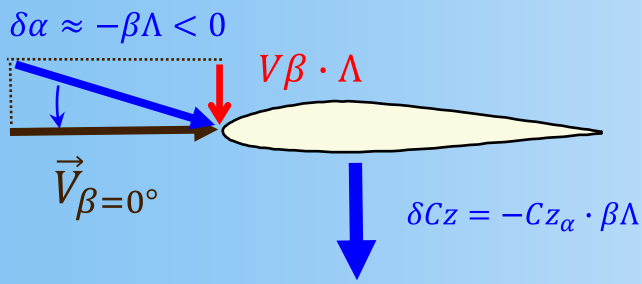
The left wing sees a decrease of angle of attack α

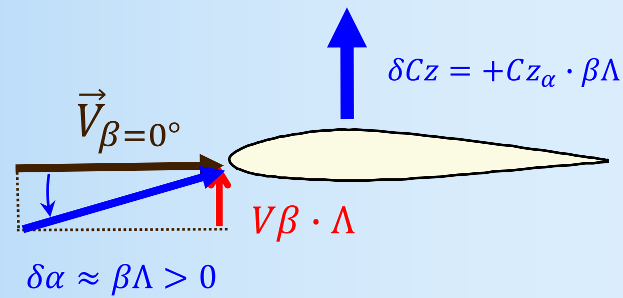
The right wing sees an increase of angle of attack α

The right / left wing sees a modified angle of attack

➢ There is an unbalance in lift forces

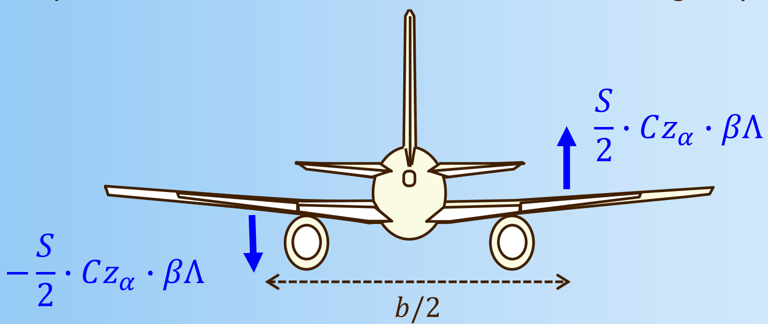
Wing dihedral angle Λ effect : Cl_β





The left wing sees a decrease of angle of attack α

The right wing sees an increase of angle of attack α



$$SL \cdot Cl_\beta \cdot \beta = -\frac{S}{2} \cdot Cz_\alpha \cdot \beta\Lambda \cdot \frac{b}{2} \quad \Rightarrow \quad Cl_\beta = -\frac{1}{4} \cdot Cz_\alpha \cdot \Lambda \cdot \frac{b}{L} < 0$$

Wing dihedral angle Λ effect : Cl_β

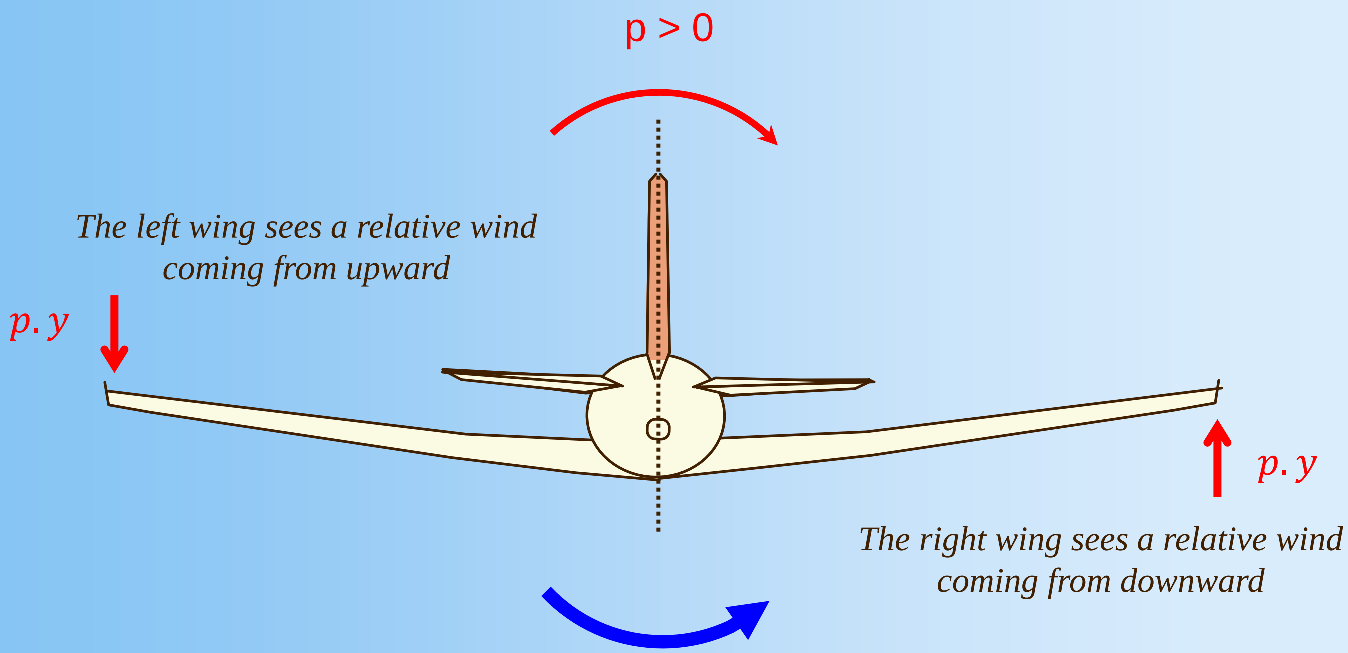
High wing produces too much negative Cl_β !

The wing dihedral shall be negative in order limit this effect

Lateral aero coefficients : β effect

Fuselage + Fin	$Cy_\beta = Cy_\beta^{fus} + Cy_\beta^{fin} < 0$
Fuselage + Fin Wing Sweep φ	$Cn_\beta = Cn_\beta^{fus} + Cn_\beta^{fin} = Cy_\beta^{fus} \cdot \frac{\Delta L_{fus}}{L} - Cy_\beta^{fin} \cdot \frac{\Delta L_{fin}}{L} > 0$ $Cn_\beta = \frac{1}{4} Cx \cdot \sin 2\varphi \cdot \frac{b}{L} > 0$
Wing Sweep φ Wing Dihedral Λ	$Cl_\beta = -\frac{1}{4} Cz \cdot \sin 2\varphi \cdot \frac{b}{L} < 0$ $Cl_\beta = -\frac{1}{4} \cdot Cz_\alpha \cdot \Lambda \cdot \frac{b}{L} < 0$

Roll rate effect on wing : Cl_p



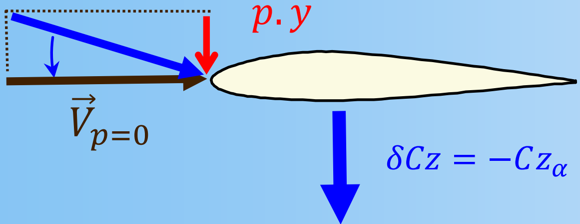
$$Cl_p \cdot \frac{pL}{V} < 0$$

The right / left wing sees a modified angle of attack

- There is an unbalance in lift forces

Roll rate effect on wing : Cl_p

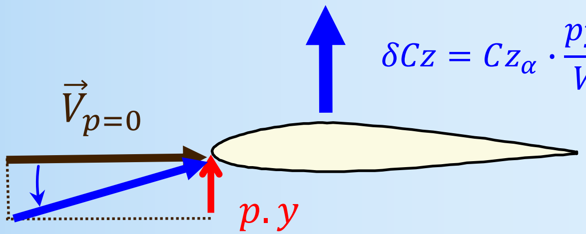
$$\delta\alpha \approx -\frac{py}{V} < 0$$



$$\delta Cz = -Cz_\alpha \cdot \frac{py}{V}$$

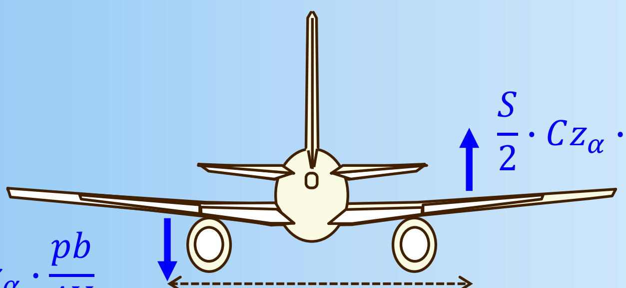
The left wing sees a relative wind coming from upward

$$\delta Cz = Cz_\alpha \cdot \frac{py}{V}$$



$$\delta\alpha \approx \frac{py}{V} > 0$$

The right wing sees a relative wind coming from downward



$$\frac{S}{2} \cdot Cz_\alpha \cdot \frac{pb}{4V}$$

$$-\frac{S}{2} \cdot Cz_\alpha \cdot \frac{pb}{4V}$$

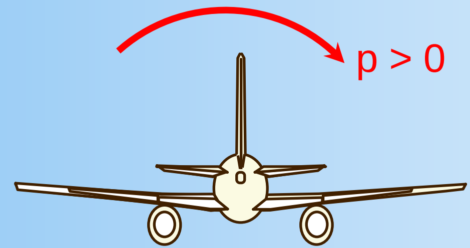
$b/2$

with the assumption that half of the wing see a mean variation

$$\delta\alpha \approx \frac{pb}{4V}$$

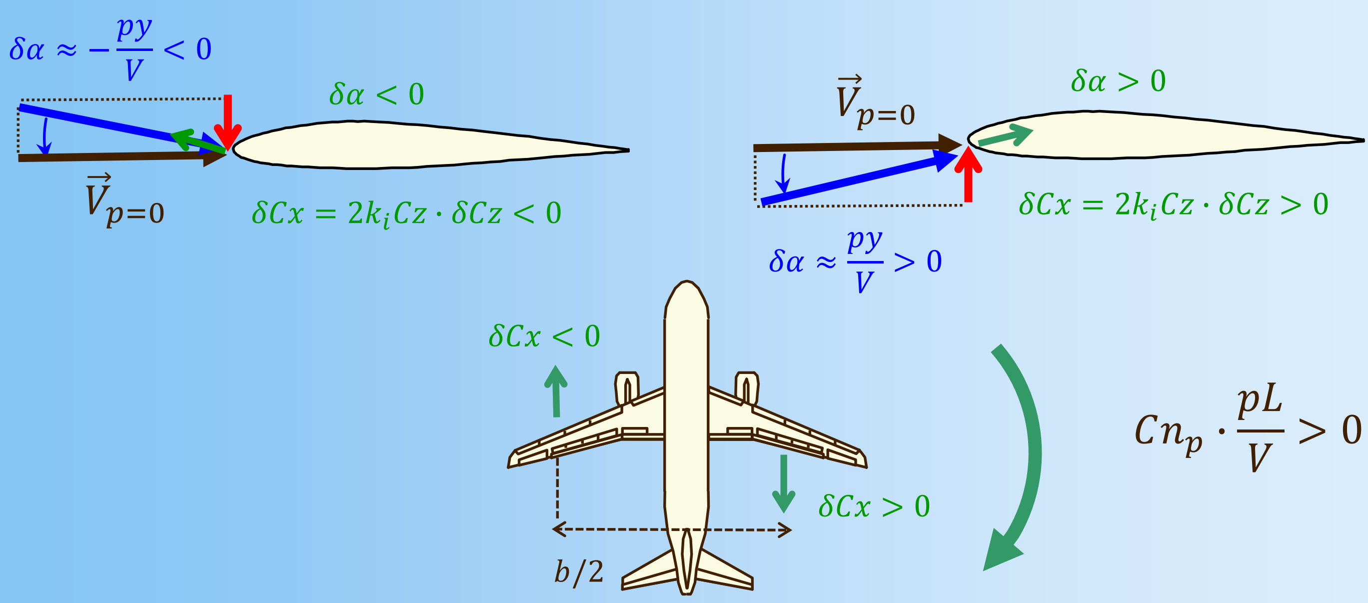
$$SL \cdot Cl_p \cdot \frac{pL}{V} = -\frac{S}{2} \cdot Cz_\alpha \cdot \frac{pb}{4V} \cdot \frac{b}{2} \Rightarrow Cl_p = -\frac{1}{16} \cdot Cz_\alpha \cdot \left(\frac{b}{L}\right)^2 < 0$$

Roll rate effect on wing : Cn_p / Drag effect



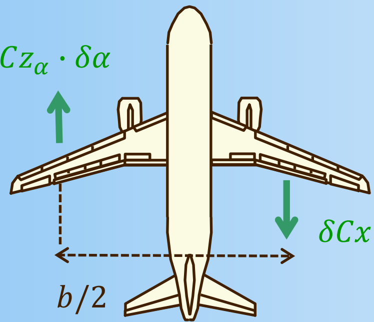
The right wing sees an increase of angle of attack α ; the left one a decrease

➢ *There is an unbalance in drag forces*



Roll rate effect on wing : Cn_p / Drag effect

$\delta Cx = 2k_i Cz \cdot Cz_\alpha \cdot \delta\alpha$



$\delta Cx = 2k_i Cz \cdot Cz_\alpha \cdot \delta\alpha$

$$Cn_p \cdot \frac{pL}{V} > 0$$

with the assumption that half of the wing see a mean variation : $\delta\alpha \approx \dfrac{pb}{4V}$

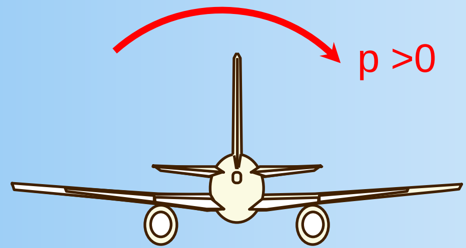
$$SL \cdot Cn_p \cdot \frac{pL}{V} = \frac{S}{2} \cdot 2k_i Cz \cdot Cz_\alpha \cdot \frac{pb}{4V} \cdot \frac{b}{2}$$

$$\Downarrow$$

$$Cn_p = \frac{1}{8} \cdot k_i Cz \cdot Cz_\alpha \cdot \left(\frac{b}{L}\right)^2 = \frac{1}{8} \cdot \frac{Cz \cdot Cz_\alpha}{\pi\lambda \cdot e} \cdot \left(\frac{b}{L}\right)^2$$

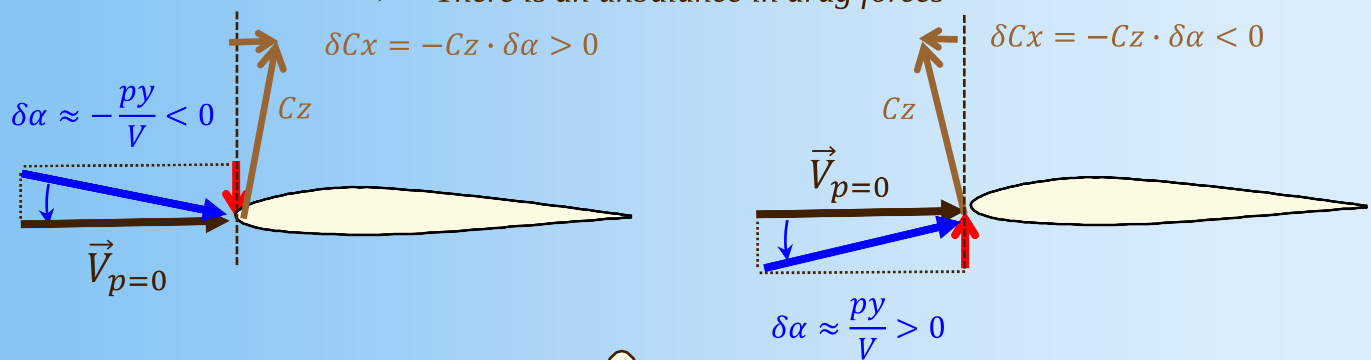
$k_i = 1/\pi\lambda e$ *with* e = *Oswald coefficient* =0,95

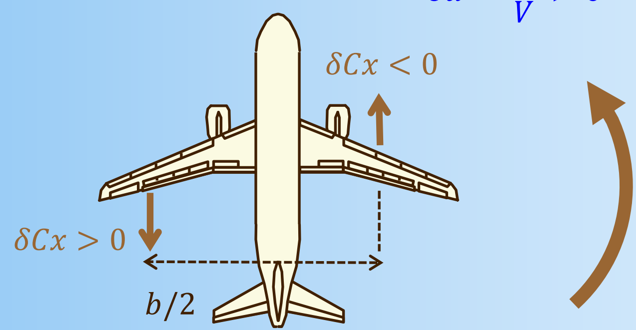
Roll rate effect on wing : Cn_p / Lift effect



The right wing sees a forward inclined lift ; the left one a backward

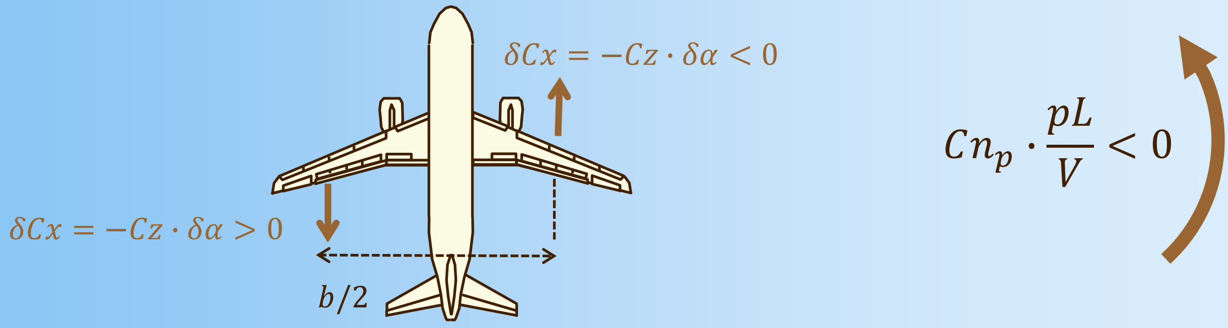
➢ *There is an unbalance in drag forces*





$$Cn_p \cdot \frac{pL}{V} < 0$$

Roll rate effect on wing : Cn_p / Lift effect



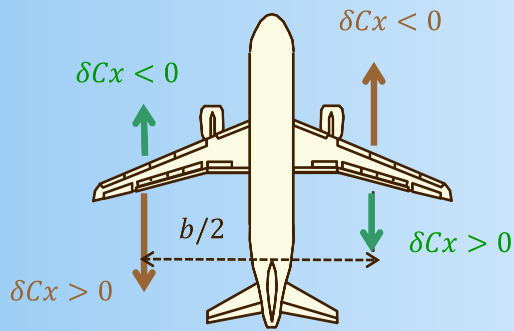
with the assumption that half of the wing see a mean variation : $\delta\alpha \approx \frac{pb}{4V}$

$$SL \cdot Cn_p \cdot \frac{pL}{V} = -\frac{S}{2} \cdot Cz \cdot \frac{pb}{4V} \cdot \frac{b}{2}$$

$$\Downarrow$$

$$Cn_p = -\frac{1}{16} \cdot Cz \cdot \left(\frac{b}{L}\right)^2$$

Roll rate effect on wing : Cn_p



$$Cn_p \cdot \frac{pL}{V} < 0$$

$$Cn_p = -\frac{1}{16} \cdot Cz \cdot \left(\frac{b}{L}\right)^2 \cdot \left(1 - \frac{2 \cdot Cz_\alpha}{\pi\lambda \cdot e}\right) < 0$$

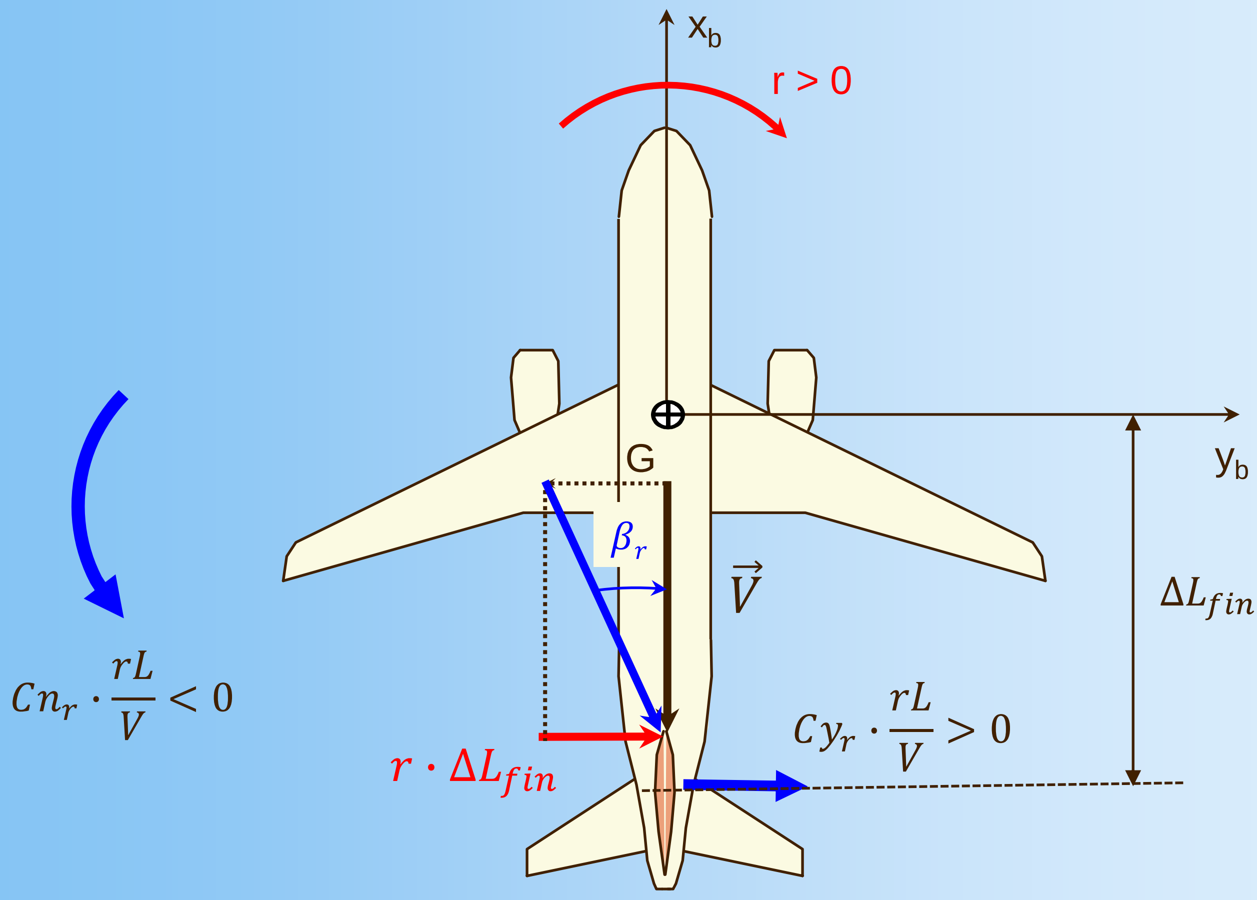
$$Cz_\alpha \approx 2\pi \rightarrow Cn_p \approx -\frac{1}{16} \cdot Cz \cdot \left(\frac{b}{L}\right)^2 \cdot \left(1 - \frac{4{,}2}{\lambda}\right) < 0$$

valid for high aspect ratio

Lateral aero coefficients : p effect

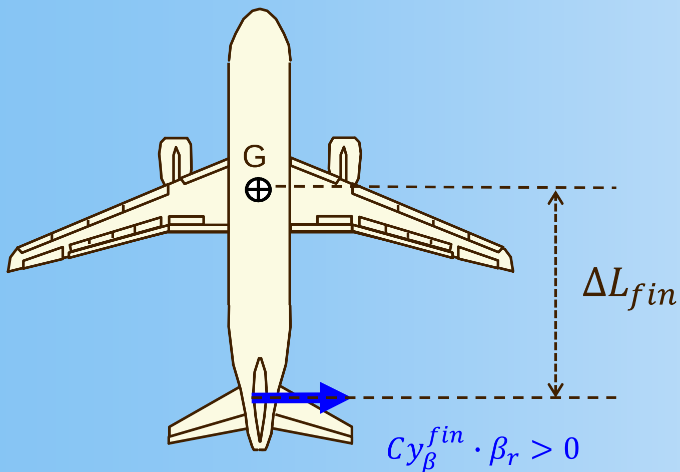
	$Cy_p \approx 0$
Wing Span b/L	$Cn_p \approx -\frac{1}{16} \cdot Cz \cdot \left(\frac{b}{L}\right)^2 \cdot \left(1 - \frac{4{,}2}{\lambda}\right) < 0$
Wing Span b/L	$Cl_p = -\frac{1}{16} \cdot Cz_\alpha \cdot \left(\frac{b}{L}\right)^2 < 0$

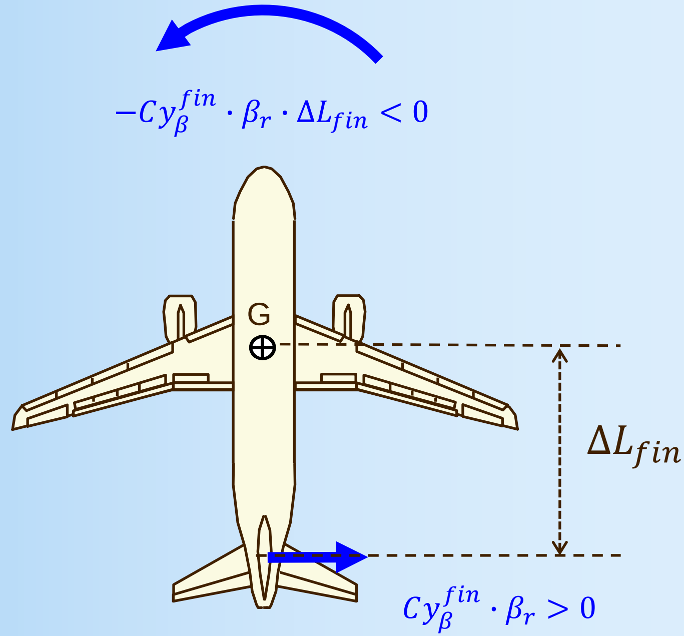
Yaw rate effect on fin : Cy_r / Cn_r



The VTP sees a relative wind coming from the left $\beta_r = -\frac{r.\Delta L_{fin}}{V} < 0$

Yaw rate effect on fin : Cy_r / Cn_r





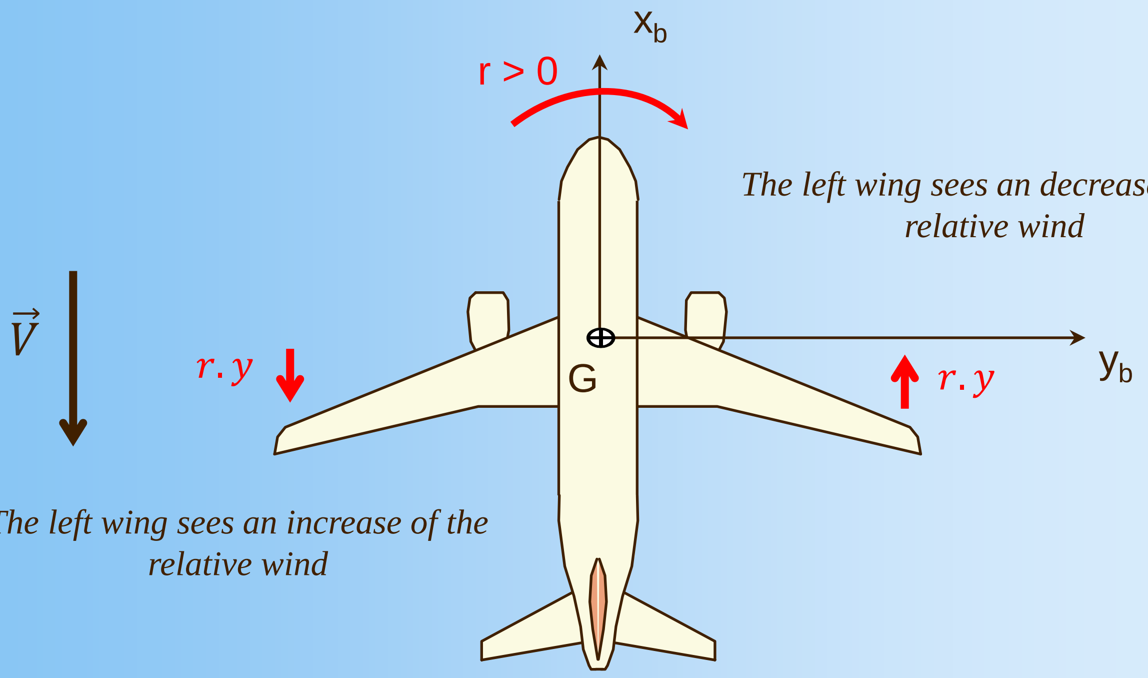
$$S \cdot Cy_r \cdot \frac{rL}{V} = S \cdot Cy_\beta^{fin} \cdot -\frac{r.\Delta L_{fin}}{V}$$

$$\Rightarrow \quad Cy_r = -Cy_\beta^{fin} \cdot \frac{\Delta L_{fin}}{L} > 0$$

$$SL \cdot Cn_r \cdot \frac{rL}{V} = -S \cdot Cy_\beta^{fin} \cdot -\frac{r.\Delta L_{fin}}{V} \cdot \Delta L_{fin}$$

$$\Rightarrow \quad Cn_r = Cy_\beta^{fin} \cdot \left(\frac{\Delta L_{fin}}{L}\right)^2 < 0$$

Yaw rate effect on wing : Cl_r



The left wing sees an decrease of the relative wind

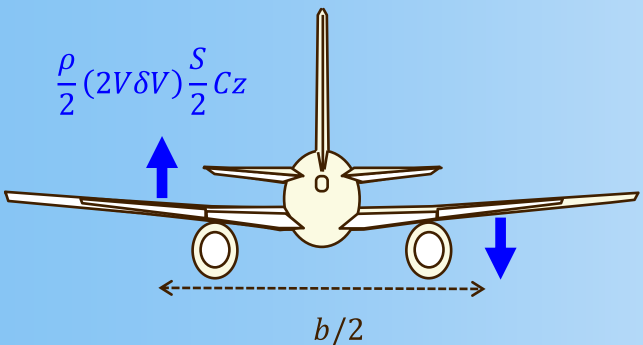
The left wing sees an increase of the relative wind

The right / left wing sees different relative velocities

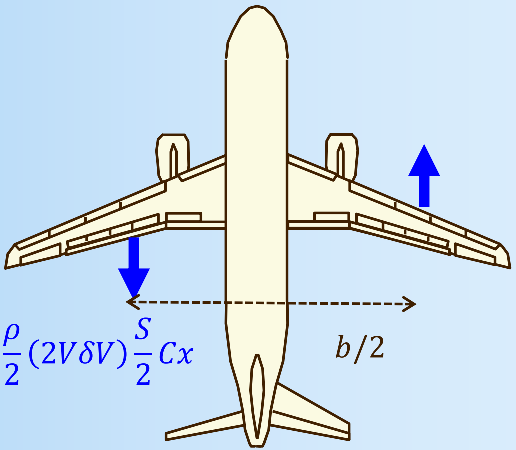
➢ There is an unbalance in lift / drag forces

Yaw rate effect on wing : Cl_r / Cn_r

$$Cl_r \cdot \frac{rL}{V} > 0$$



$$Cn_r \cdot \frac{rL}{V} < 0$$



with the assumption that half of the wing see a mean variation : $\delta V \approx \frac{rb}{4}$

$$\frac{\rho V^2}{2} \cdot SL \cdot Cl_r \cdot \frac{rL}{V} = \frac{\rho}{2}\left(2V\frac{rb}{4}\right)\frac{S}{2}Cz \cdot \frac{b}{2}$$

$$\frac{\rho V^2}{2} \cdot SL \cdot Cn_r \cdot \frac{rL}{V} = -\frac{\rho}{2}\left(2V\frac{rb}{4}\right)\frac{S}{2}Cx \cdot \frac{b}{2}$$

$$Cl_r = \frac{1}{8} \cdot Cz \cdot \left(\frac{b}{L}\right)^2 > 0$$

$$Cn_r = -\frac{1}{8} \cdot Cx \cdot \left(\frac{b}{L}\right)^2 < 0$$

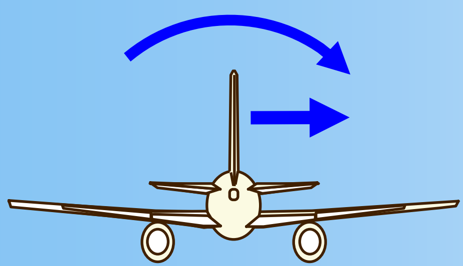
Lateral aero coefficients : r effect

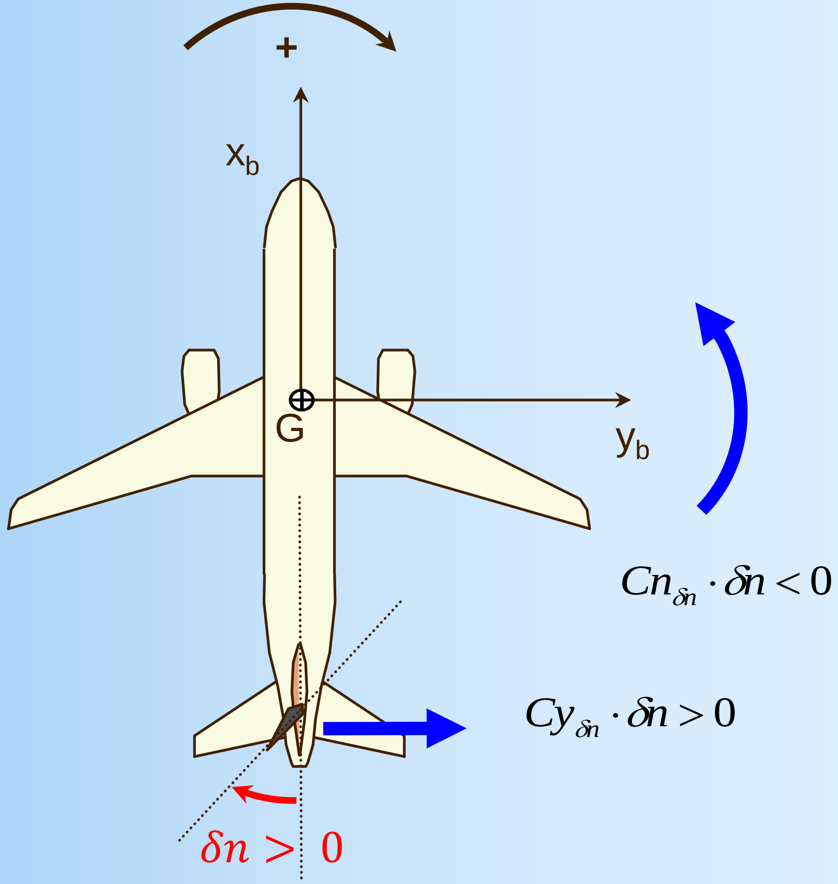
Fin	$Cy_r = -Cy_\beta^{fin} \cdot \frac{\Delta L_{fin}}{L} > 0$
Fin Wing Span b/L	$Cn_r = Cy_\beta^{fin} \cdot \left(\frac{\Delta L_{fin}}{L}\right)^2 < 0$ $Cn_r = -\frac{1}{8} \cdot Cx \cdot \left(\frac{b}{L}\right)^2 < 0$
Wing Span b/L	$Cl_r = \frac{1}{8} \cdot Cz \cdot \left(\frac{b}{L}\right)^2 > 0$

UPPER FIN / INVERSE ROLL

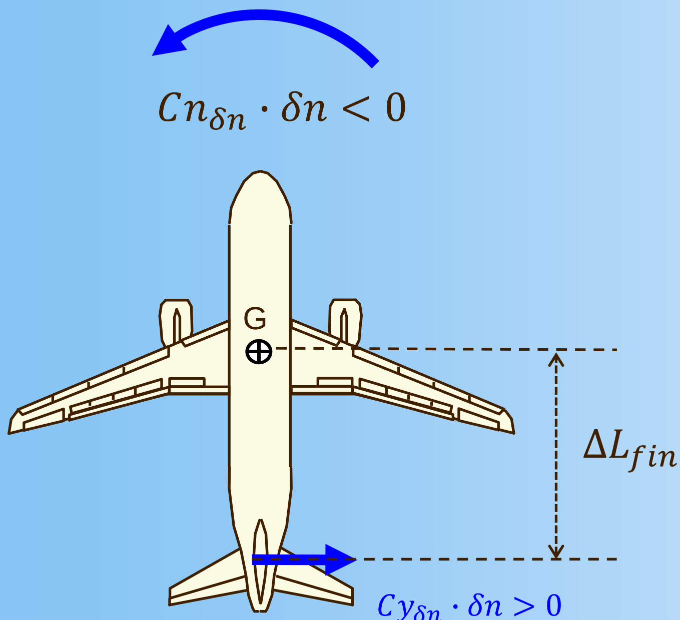
You apply a Yaw command for turning left, due to the relative Center of Gravity position, naturally, the aircraft banks on the right!

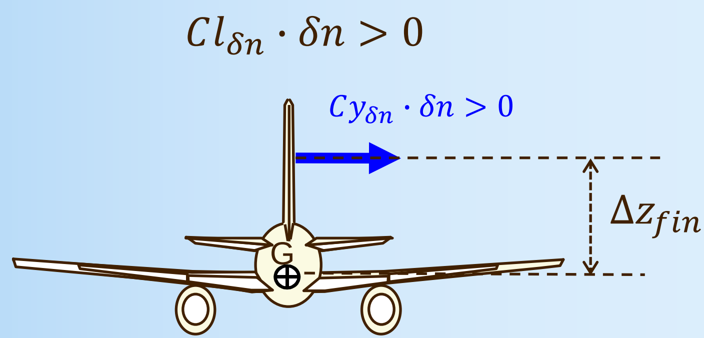
$$Cl_{\delta n} \cdot \delta n > 0$$





Rudder effect : $Cy_{\delta n}$ / $Cn_{\delta n}$





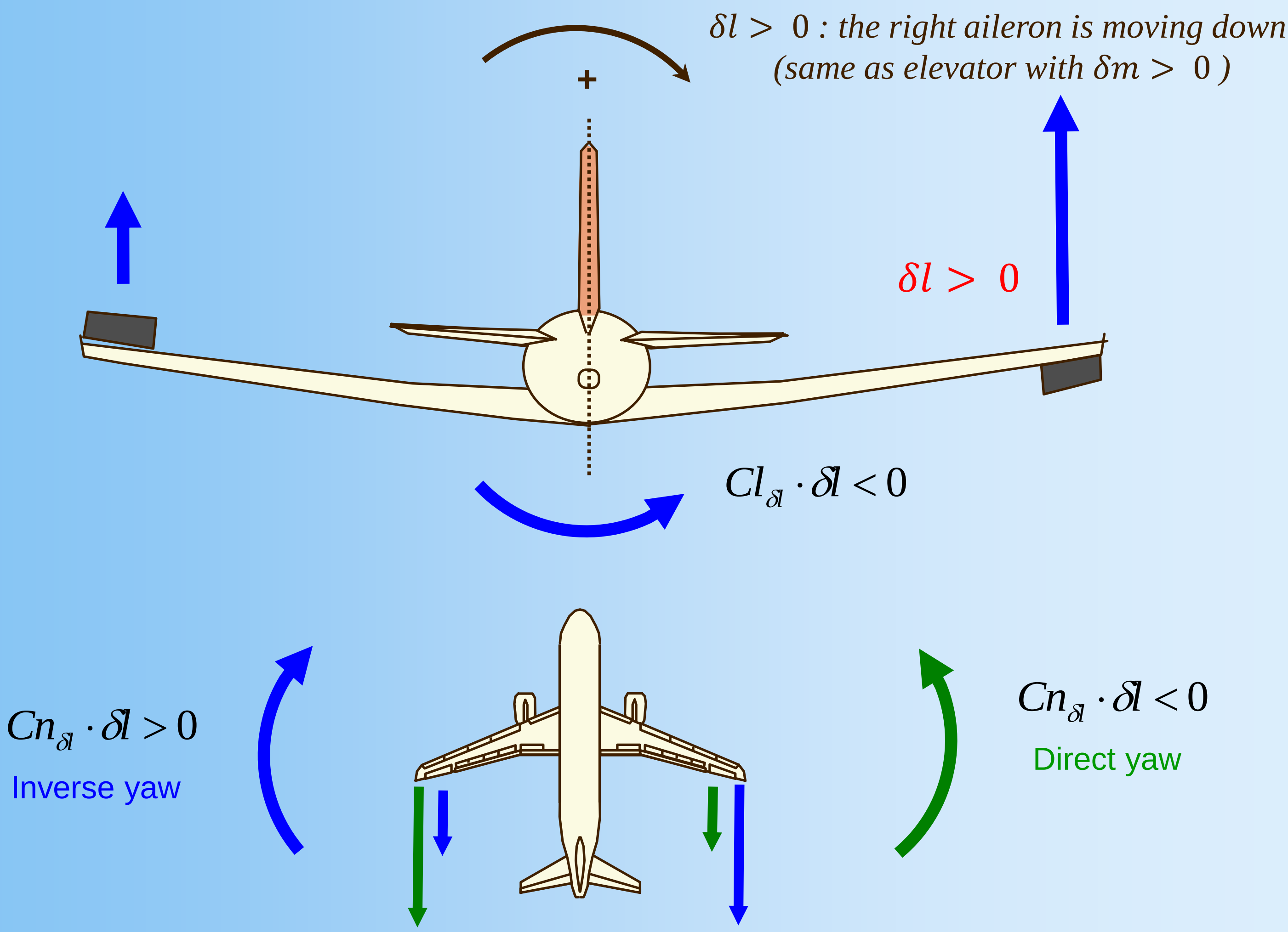
$$SL \cdot Cn_{\delta n} \cdot \delta n = -S \cdot Cy_{\delta n} \cdot \delta n \cdot \Delta L_{fin}$$

$$\Rightarrow \quad Cn_{\delta n} = -\overset{+}{Cy_{\delta n}} \cdot \frac{\Delta L_{fin}}{L} < 0$$

$$SL \cdot Cl_{\delta n} \cdot \delta n = S \cdot Cy_{\delta n} \cdot \delta n \cdot \Delta z_{fin}$$

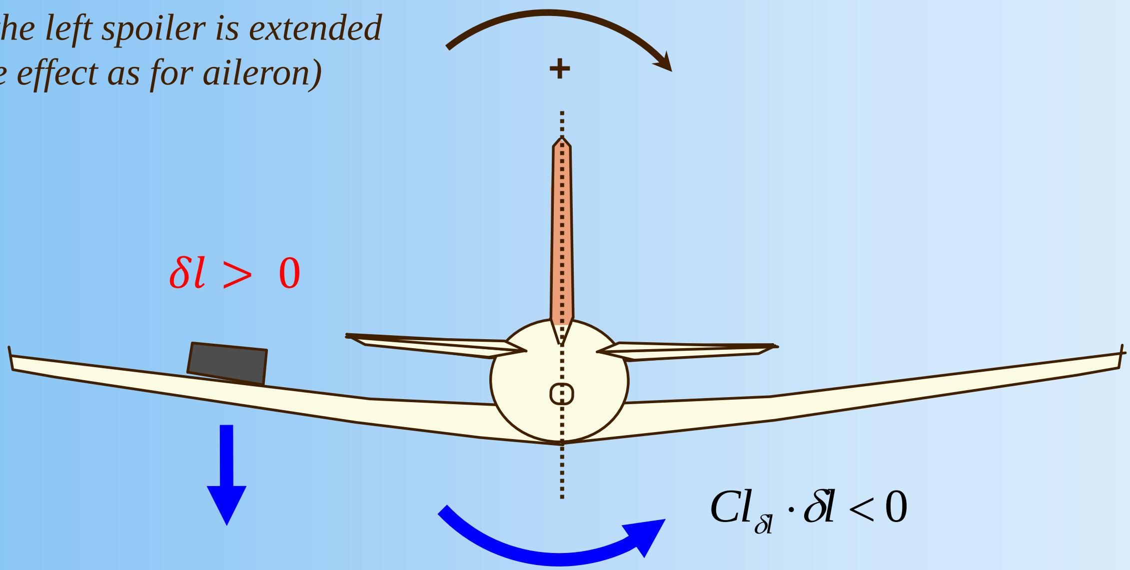
$$\Rightarrow \quad Cl_{\delta n} = \overset{+}{Cy_{\delta n}} \cdot \frac{\Delta z_{fin}}{L} > 0$$

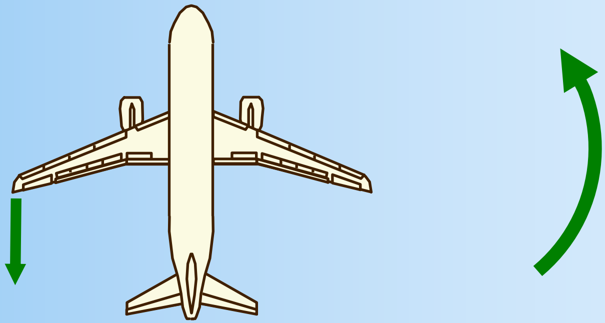
Ailerons effect : $Cl_{\delta l}$ / $Cn_{\delta l}$



Spoilers effect : $Cl_{\delta l}$ / $Cn_{\delta l}$

$\delta l > 0$: the left spoiler is extended
(same effect as for aileron)

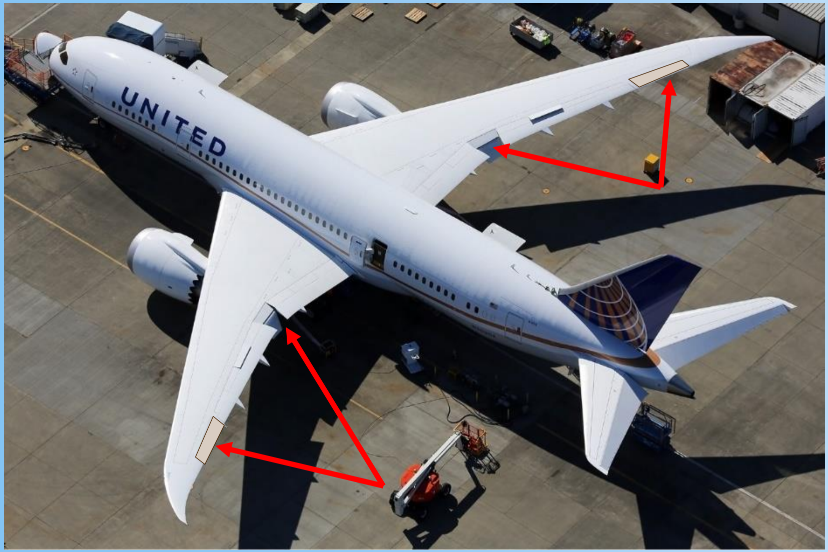


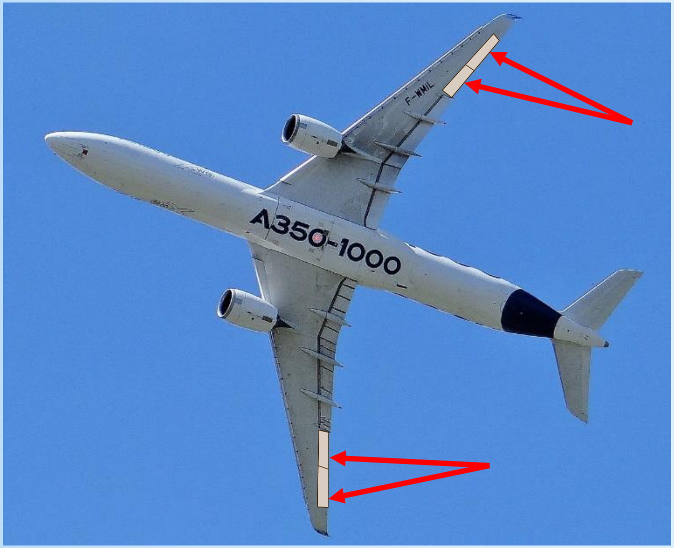


Lateral aero coefficients : δl / δn effect

Fin	$Cy_{\delta n} > 0$
Fin Aileron Spoiler	$Cn_{\delta n} = -Cy_{\delta n} \cdot \frac{\Delta L_{fin}}{L} < 0$ $Cn_{\delta l}$? $Cn_{\delta l} < 0$
Aileron Spoiler Fin	$Cl_{\delta l} < 0$ $Cl_{\delta l} < 0$ $Cl_{\delta n} = Cy_{\delta n} \cdot \frac{\Delta z_{fin}}{L} > 0$

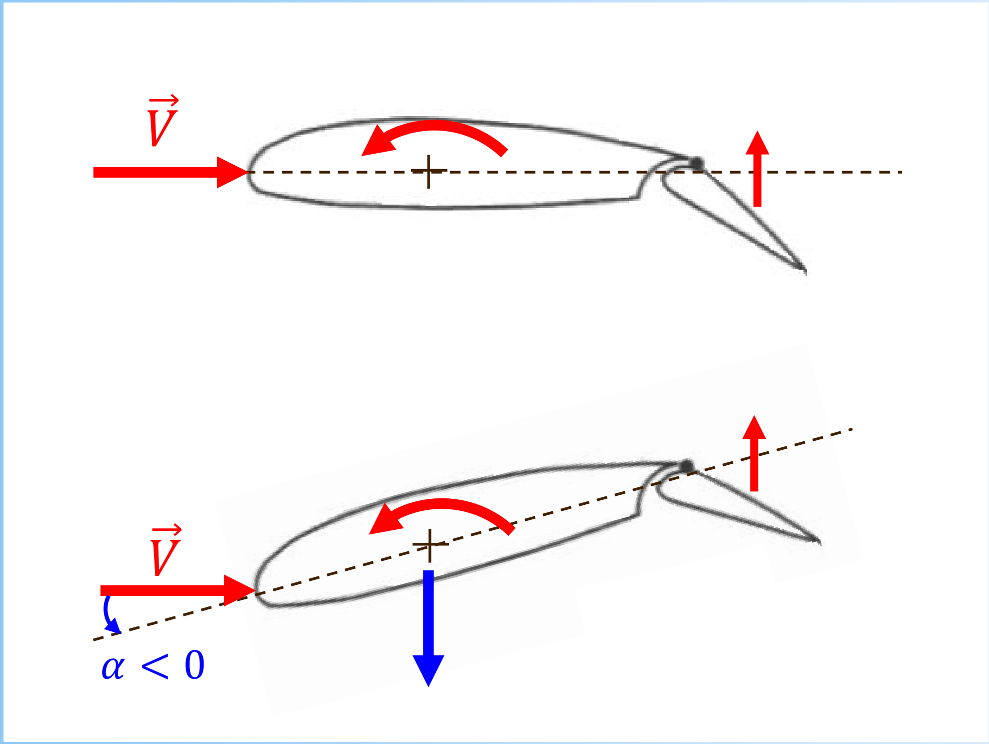
Ailerons architecture solutions





- With its flexible wing concept, Boeing has converged on a solution with one outer aileron + one inner aileron located between the inner/outer flap
- With a more conventional wing, Airbus has converged on the usual solution with one outer aileron + one inner aileron both located at the wing tip

Aeroelastic : ailerons reversal issue



Aileron reversal results when the aileron structure is insufficiently stiff in torsion
The aerodynamic twisting of the wing is caused by ailerons as speed is increased
This may reduce, neutralize, or reverse the direction of the lift
This effect may limit the operational flight domain

Spoiler : principle and function

Spoilers are hinged plates which are extended on the upward part of the wing (40° max deflection). The primary effect of the spoiler is to destroy the process of pressure differential between both side of the wing, and so to destroy the local lift. A secondary effect is to generate a big amount of drag (when totally extended)

Used asymmetrically, by producing a lift decrease, you can generate a moment for controlling the roll.

Used symmetrically,

- For descent emergency, by generating a big amount of drag
- At ground, after touch down at landing, they are used as lift dumper for improving the braking by putting more down forces on the wheels and producing a big amount of drag

Spoiler versus airbrakes

Air brakes differ from spoilers : air brakes are designed to increase drag while making little change to lift, whereas spoilers drastically reduce the lift generation
Airbrakes are used for descent emergency and at landing to slow the aircraft.


Air brakes on the rear fuselage
of a BAE 146

Split speed brakes inboard of
"tailerons“ / F-16 Falcon

Lateral aerodynamic coefficients

	Cy	Cl	Cn
$\beta\ (°)$ Side Slip	$Cy_\beta < 0$	$Cl_\beta < 0$ dihedral effect	$Cn_\beta > 0$ lateral stability
$p\ (°/s)$ Roll Rate	$Cy_p \approx 0$	$Cl_p < 0$ roll damping	$Cn_p < 0$ bent lift
$r\ (°/s)$ Yaw Rate	$Cy_r > 0$	$Cl_r > 0$	$Cn_r < 0$ yaw damping
$\delta l\ (°)$ Aileron	$Cy_{\delta l} \approx 0$	$Cl_{\delta l} < 0$	$Cn_{\delta l}$? direct / inverse
$\delta n (°)$ Rudder	$Cy_{\delta n} > 0$	$Cl_{\delta n} > 0$ inverse roll	$Cn_{\delta n} < 0$

$Cn_{\delta l} < 0$
direct for spoiler