< 4.10 The Vortex Panel Method >

* Thin airfoil theory

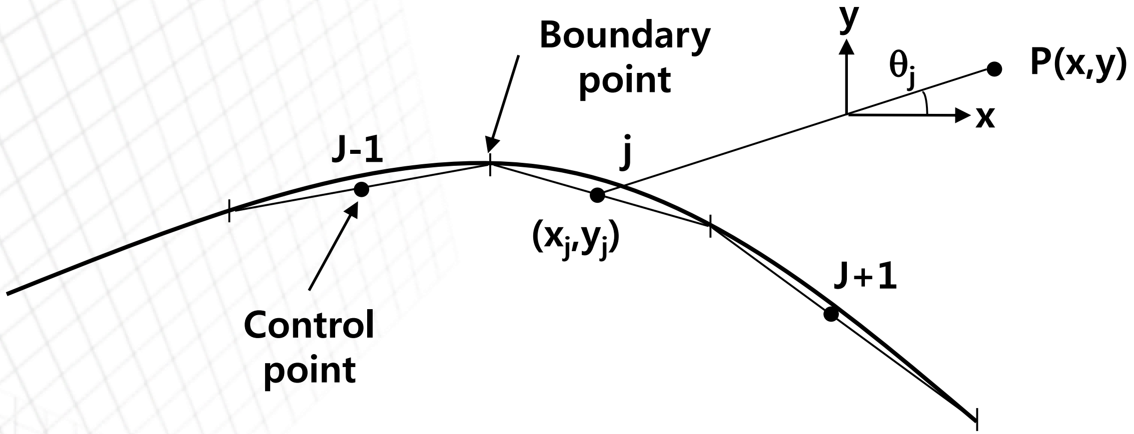
- Closed form
- Limited to thin airfoil, $\frac{t_{\max}}{c} \leq 12\%$

* Panel method _2017

- Vortex panel
- Source panel → non-lifting cases

* Exactly same idea of thin airfoil theory, but no closed form $\gamma(s)$
→ solve numerically

< 4.10 The Vortex Panel Method >



* The velocity potential at P due to j-th panel

$$\Delta\phi_i = -\frac{1}{2\pi}\int_j \theta_{pj}\gamma_j ds_j$$

$$\theta_{pj} = \tan^{-1}\frac{y-y_j}{x-x_j}$$

$$\Rightarrow \quad \phi(p) = \sum_{j=1}^{n}\phi_j = -\sum_{j=1}^{n}\frac{\gamma_j}{2\pi}\int_j \theta_{pj} ds_j$$

* Let's put point P at the control point of i-th panel

$$\phi(x_i,y_i) = -\sum_{j=1}^{n}\frac{\gamma_j}{2\pi}\int_j \theta_{ij} ds_j \quad where\ \theta_{ij} = \tan^{-1}\frac{y_i-y_j}{x_i-x_j}$$

< 4.10 The Vortex Panel Method >

* At the control points, the normal component of velocity is zero.

- The component of V_∞ normal to i-th panel

$$V_{\infty,n} = V_\infty \cos\beta_i$$

- The normal component of induced velocity at (x_i, y_i)

$$V_n = \frac{\partial}{\partial n_i}[\phi(x_i, y_i)]$$

$$= -\sum_{j=1}^{n} \frac{\gamma_j}{2\pi} \int_j \frac{\partial \theta_{ij}}{\partial n_i} ds_j$$

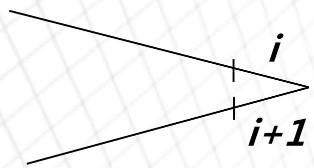
➔ $V_{\infty,n} + V_n = 0$ ➔ $V_\infty \cos\beta_i - \sum_{n=1}^{\infty} \frac{\gamma_j}{2\pi} \underbrace{\int_j \frac{\partial \theta_{ij}}{dn} ds_j}$ $= 0$

= J_{ij} : *f* (panel geometry)

< 4.10 The Vortex Panel Method >

* Boundary condition : $V_\infty \cos\beta_i - \sum_{n=1}^{\infty} \frac{\gamma_j}{2\pi} \int_j \frac{\partial \theta_{ij}}{dn} ds_j = 0$

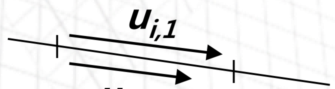
 + Kutta condition :



$$\gamma_i + \gamma_{i+1} = 0$$

* Now, we have (n+1) eq. with n unknowns ➔ ignore one of control points

* The flow velocity tangent to the surface = γ



$u_{i,2} = 0$ ← Inside the solid surface

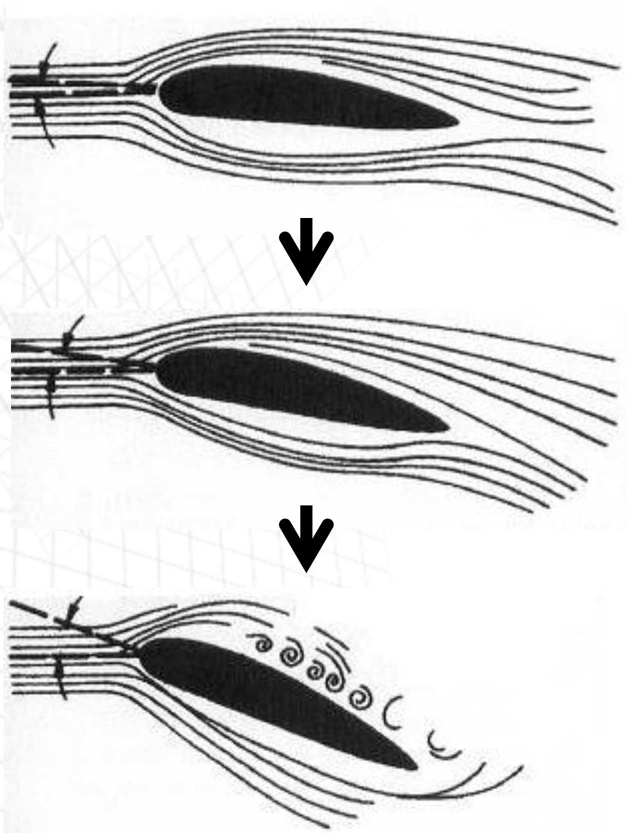
➔ $\gamma_i = u_{i1} - u_{i2} = u_{i1}$

* Total circulation : $\Gamma = \sum_{j=1}^{n} \gamma_j s_j$

* Lift : $L = \rho_\infty V_\infty \sum_{j=1}^{n} \gamma_j s_j$

< 4.12 The Flow over an Airfoil – the Real Case >

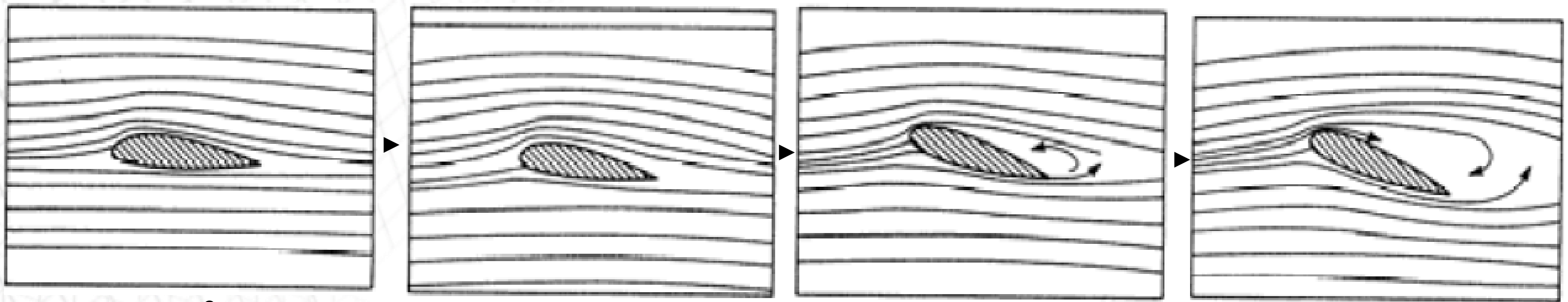
- Stall
  - Leading-edge stall



⇨ Flow separation takes place over the entire top surface of the airfoil after occurring at the leading edge

< 4.12 The Flow over an Airfoil – the Real Case >

- Stall
  - Trailing-edge stall

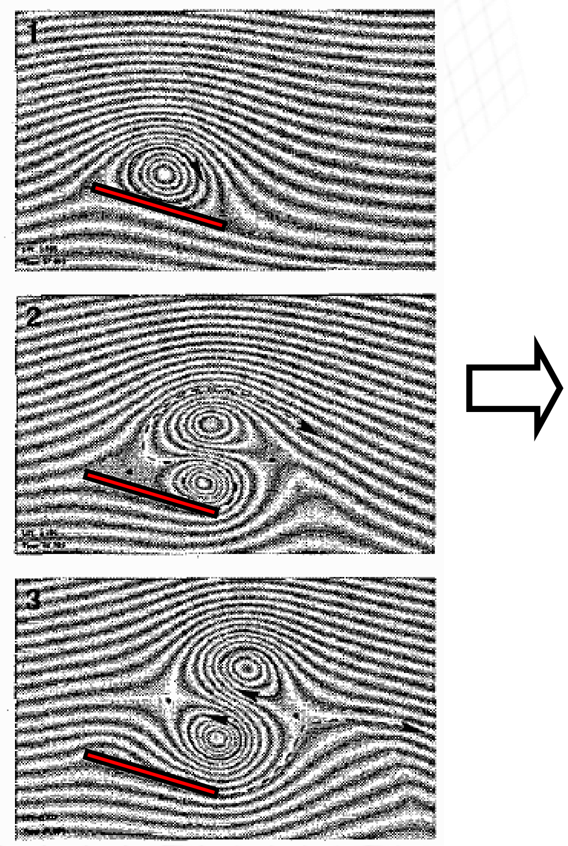


α = 5°   α = 10°   α = 15°   α =22.5°

⇨ Flow separation takes place from the trailing edge at thicker airfoils than leading-edge stall

< 4.12 The Flow over an Airfoil – the Real Case >

- Stall
  - Thin airfoil stall



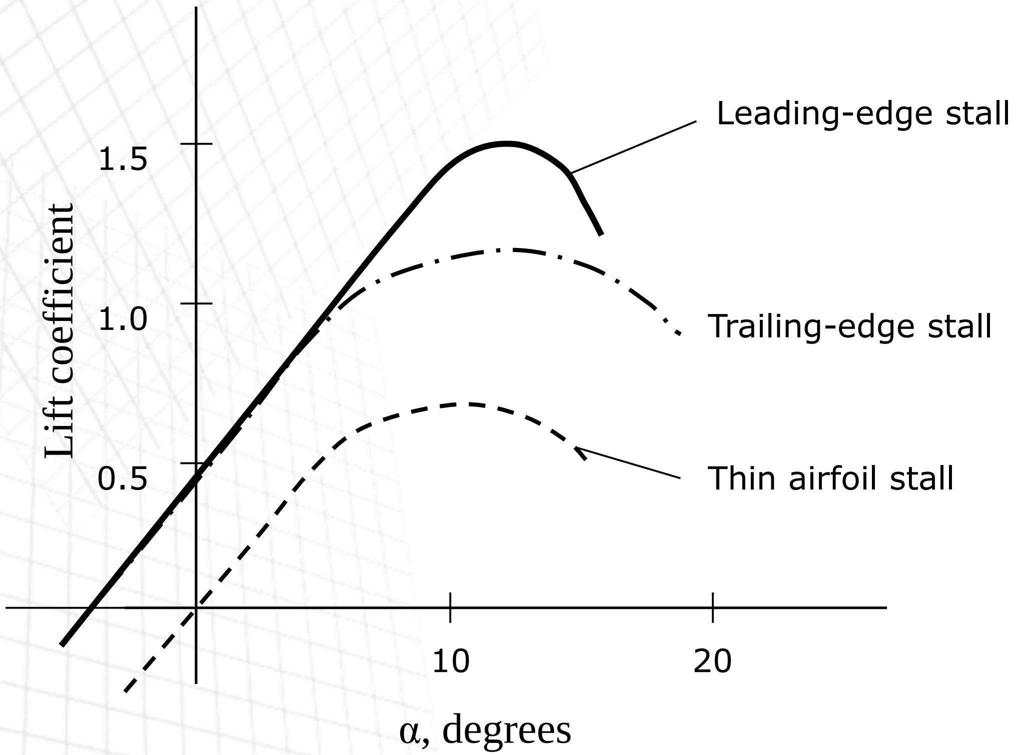
Leading-edge stall

Flow separation takes place over the entire surface of the airfoil after occurring at the leading edge

< 4.12 The Flow over an Airfoil – the Real Case >

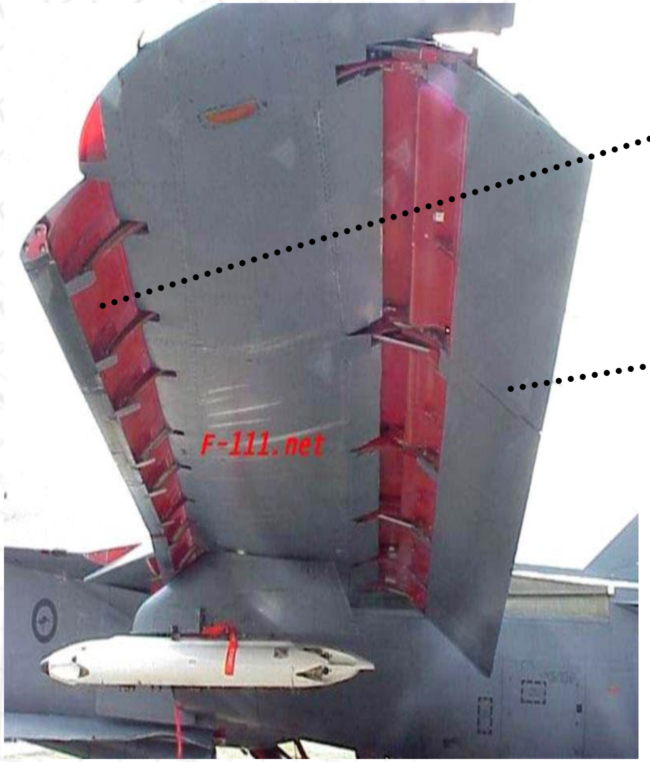
❖ Stall

- Lift-coefficient curves



< 4.12 The Flow over an Airfoil – the Real Case >

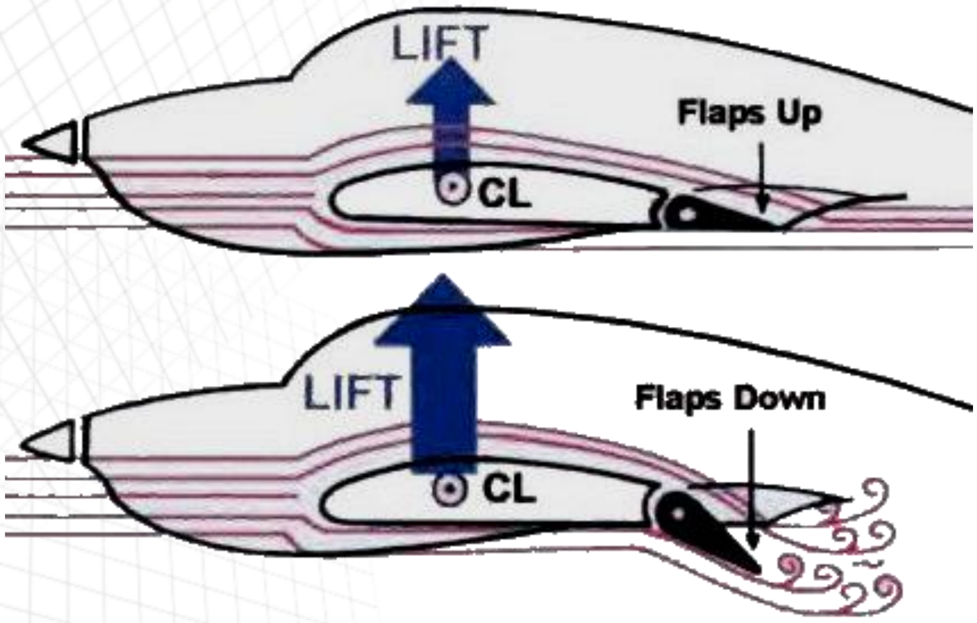
❖ High-lift devices



< 4.12 The Flow over an Airfoil – the Real Case >

❖ High-lift devices

- Trailing-edge flap (plain type)



More camber → Higher lift

< 4.12 The Flow over an Airfoil – the Real Case >

❖ High-lift devices

- Effect of slats and flaps