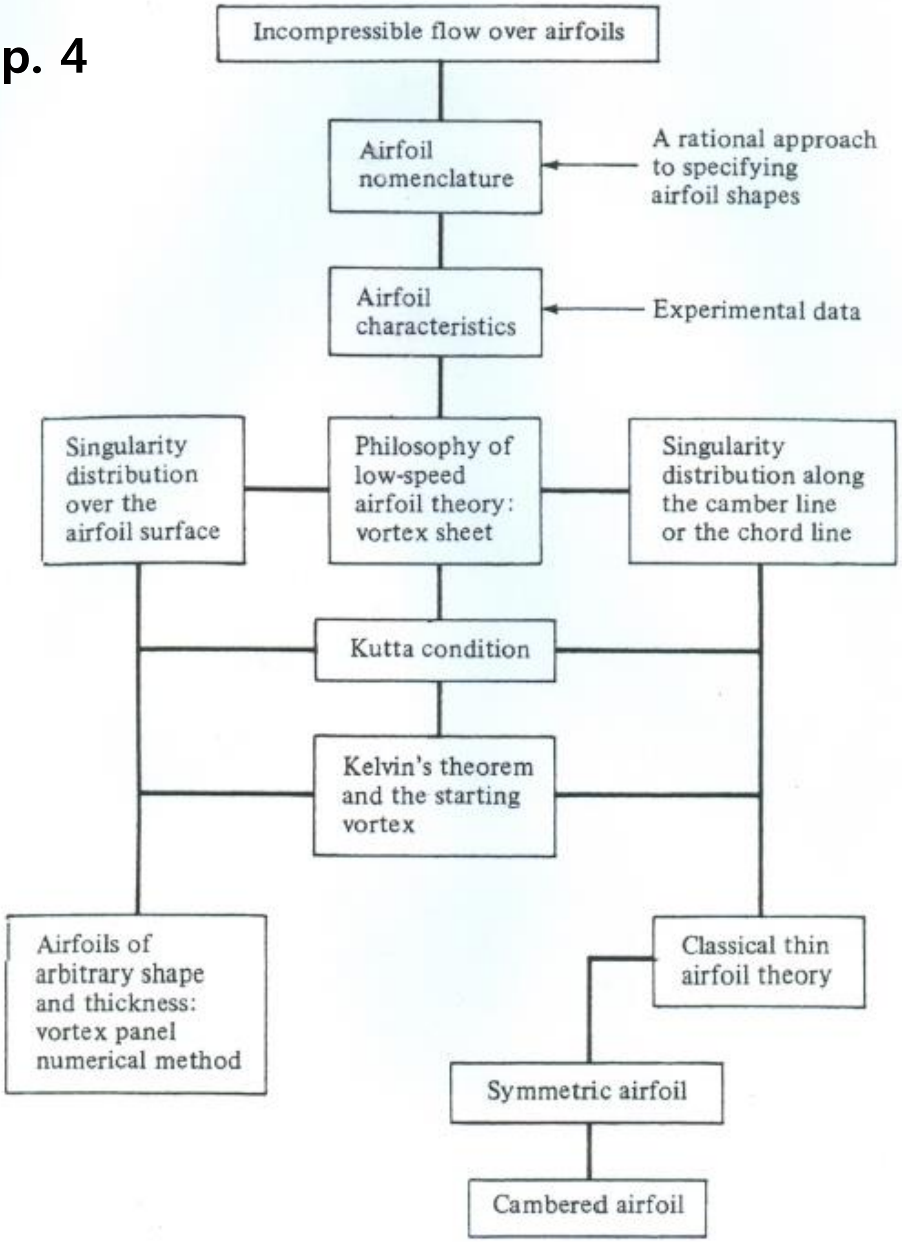
Road map for Chap. 4



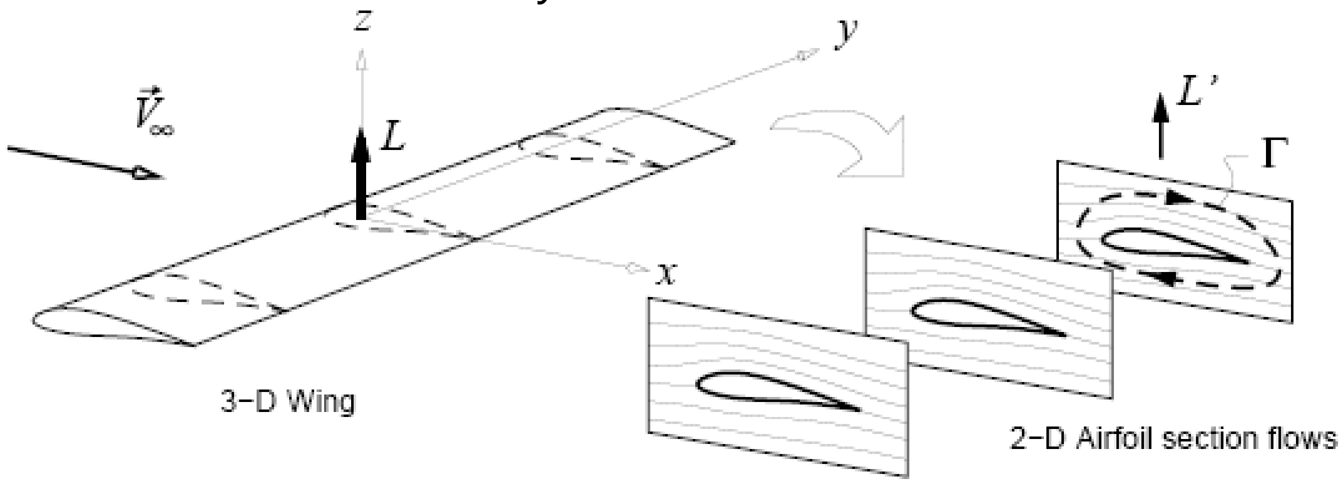
< 4.1 Introduction >

❖ Incompressible flow over airfoils

Prandtl (20C 초) ➔ Airfoil (2D)

➔ Wing (3D)

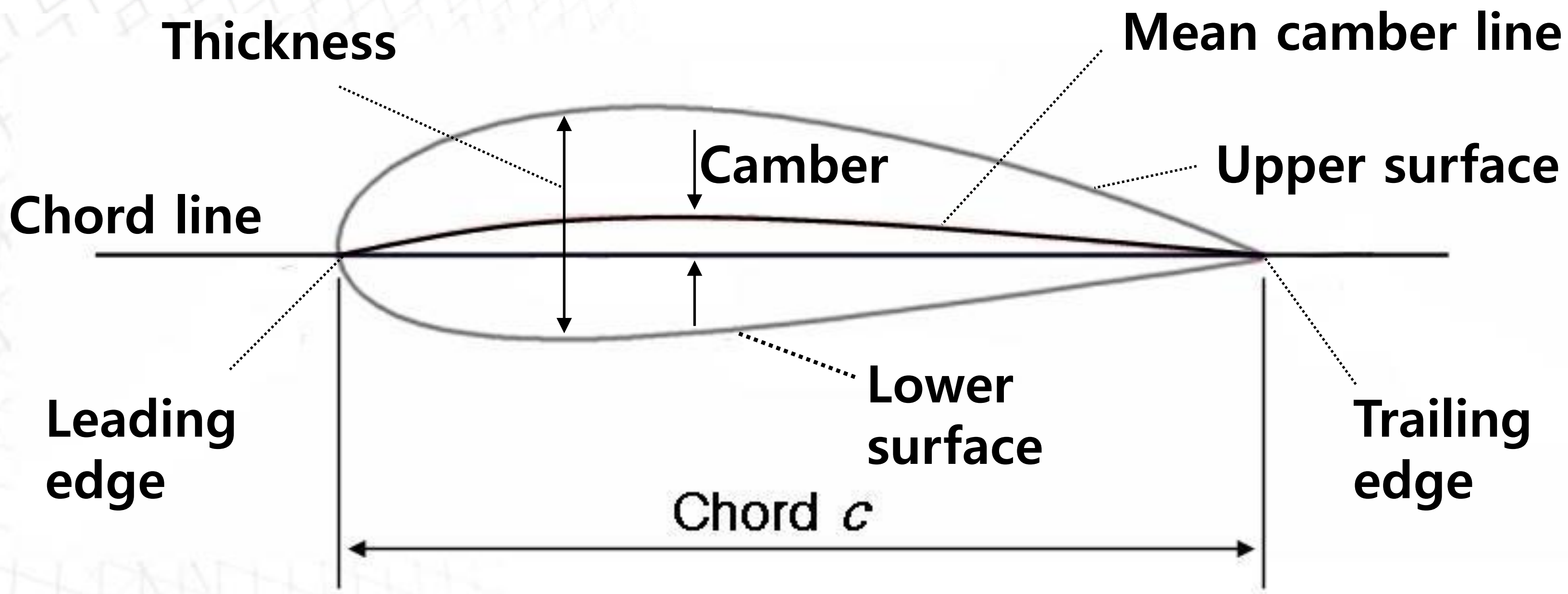
➔ Body



Airfoil : any section of the wing cut by a plane normal to y-axis

< 4.2 Airfoil Nomenclature >

❖ **NACA (National Advisory Committee for Aeronautics) series**



< 4.2 Airfoil Nomenclature >

❖ NACA (National Advisory Committee for Aeronautics) series

NACA 4-digit series

* NACA2412

2 : max. camber = 2% of the chord
4 : the location of max. camber = 40% of the chord
12 : max. thickness = 12% of the chord
If the airfoil is symmetric, it becomes NACA00XX

NACA 5-digit series

* NACA23012

2 : 2*0.3/2 = 0.3 design C_L
30 : 30/2 % = the location of max. camber
12 : max. thickness = 12% of the chord

< 4.2 Airfoil Nomenclature >

❖ NACA (National Advisory Committee for Aeronautics) series

6-digit series laminar flow airfoil

* NACA65-218
  6 : series designation
  5 : min. pressure location = 50% of the chord
  2 : design C_L= 0.2
  18 : max. thickness = 18% of the chord

Other notations

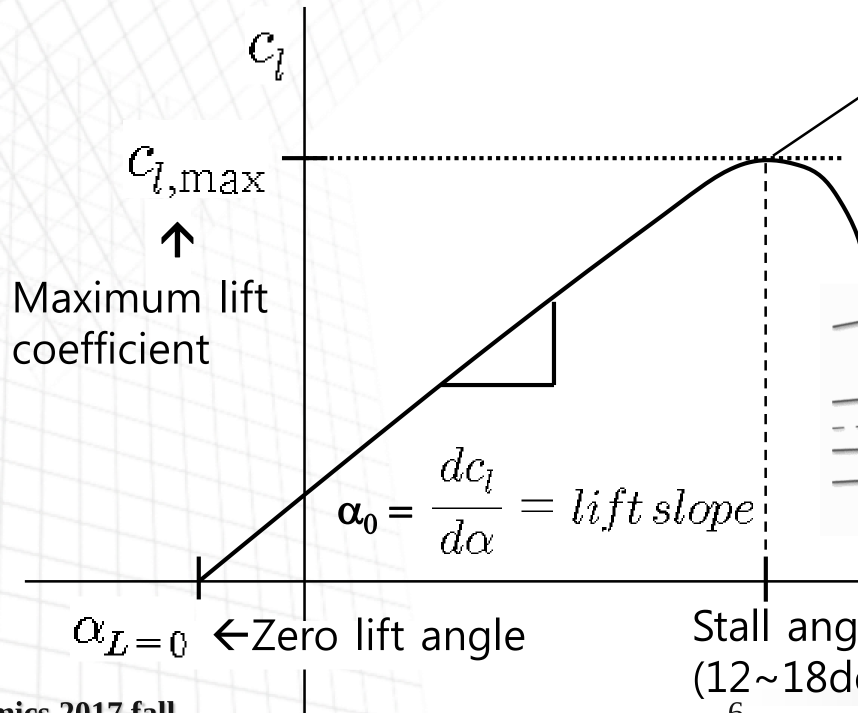
* SC1095
* VR12

< 4.3 Airfoil Characteristics >

*** 1930~40 NASA carried numerous experiments on NACA airfoil characteristics**
(Measured C_l, C_d, C_m → 2-D data)

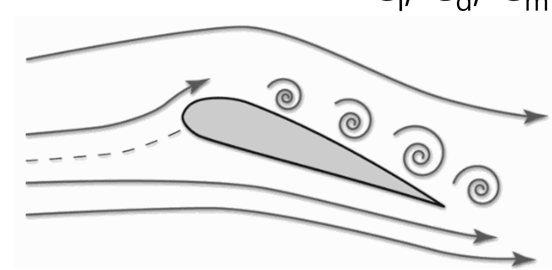
*** In the future, new airfoils should be designed and tested**
(consideration of aerodynamic, dynamic & acoustic limitation)

*** Typical lift characteristics of an airfoil**



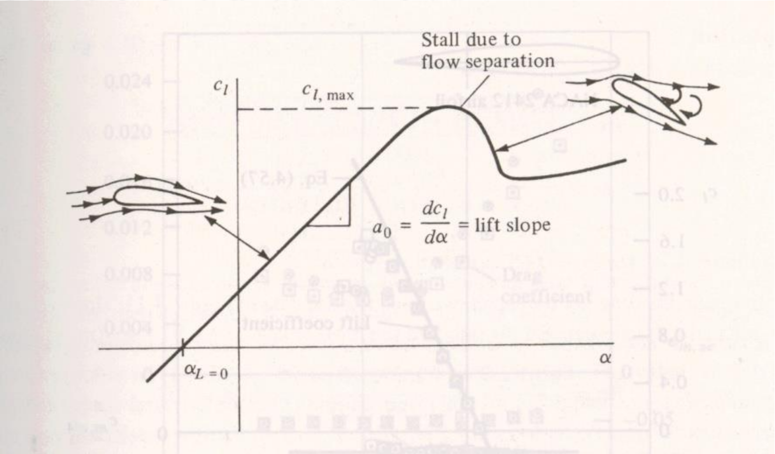
Stall → Sepatation
Dynamic stall

How to measure
C_l, C_d, C_m?



α : angle of attack

< 4.3 Airfoil Characteristics >



[Def.]

α, angle of attack : the angle between the freestream velocity and the chord

[Note]

1. α_0 is not usually a function of Re.

2. $C_{l,max}$ is dependent on Re.

< 4.3 Airfoil Characteristics >

❖ Typical drag & pitching moment characteristics

* Aerodynamic drag = Pressure drag (form drag) + Skin friction drag

↓

Sensitive to Re.

Pressure drag + Skin friction drag = Profile drag

* AC (Aerodynamic Center)

[Def.] The point about which the moment is independent of AOA

➔ Subsonic : AC=c/4
Supersonic : AC=c/2

< 4.4 Vortex Sheet >

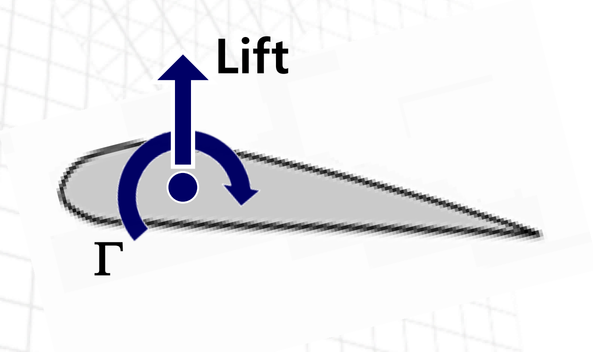
❖ Kutta-Joukowski Theorem

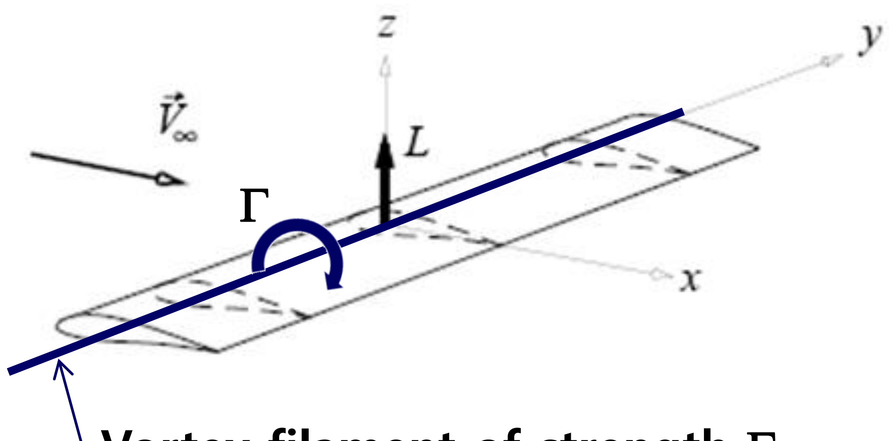
* Kutta (German), Joukowski(Russia)

* Incompressible, inviscid flow

$$\boxed{L = \rho_\infty v_\infty \Gamma}$$

* Γ : positive clockwise





Vortex filament of strength Γ

< 4.4 Vortex Sheet >

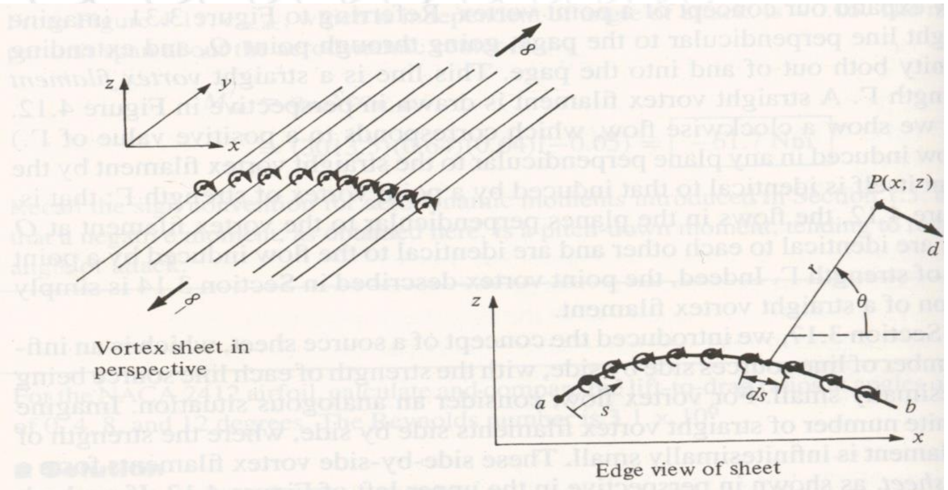


Figure 4.13 Vortex sheet.

* $\gamma(s)$ = the strength of vortex sheet per unit length along *s*

* From Biot-Savart Law

$$dV_\theta = -\frac{\gamma ds}{2\pi r}$$

* Velocity potential for vortex flow

$$V_\theta = \frac{1}{r}\frac{\partial \phi}{\partial \theta} \rightarrow d\phi = -\frac{\gamma ds}{2\pi}\theta$$

* Velocity potential at P

$$\rightarrow \boxed{\phi(x,z) = -\frac{1}{2\pi}\int_a^b \theta \gamma ds}$$

< 4.4 Vortex Sheet >

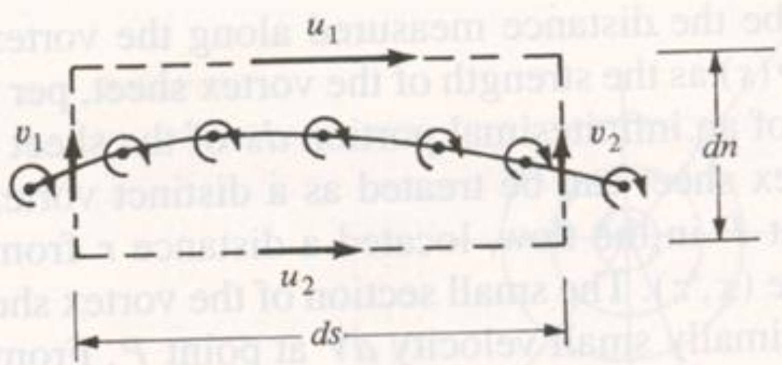


Figure 4.14 Tangential velocity jump across a vortex sheet.

* Circulation around the dashed path

$$\Gamma_1 = \gamma ds = \oint \vec{V} d\vec{l}$$

$$= -\{(v_2 - v_1)dn + (u_2 - u_1)ds\}$$

$$= (v_1 - v_2)dn + (u_1 - u_2)ds$$

* If $dn \to 0$ $\gamma ds = (u_1 - u_2)ds$ ➔ $\therefore \gamma = u_1 - u_2$

(Note)

The local strength of the vortex sheet is equal to <u>the difference (jump) in tangential velocity</u> across the vortex sheet

< 4.4 Vortex Sheet >

* "Vortex Sheet" - Application for inviscid, incompressible flow

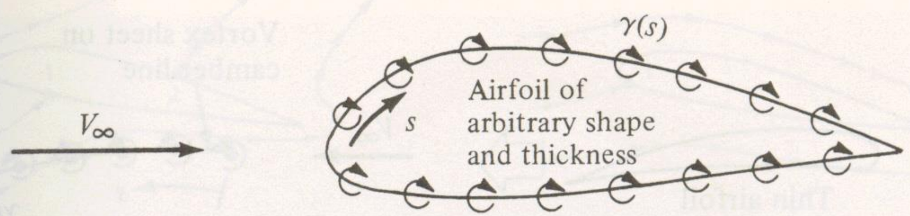


Figure 4.15 Simulation of an arbitrary airfoil by distributing a vortex sheet over the airfoil surface.

* Calculate g(s) to form the streamlines with a give airfoil shape

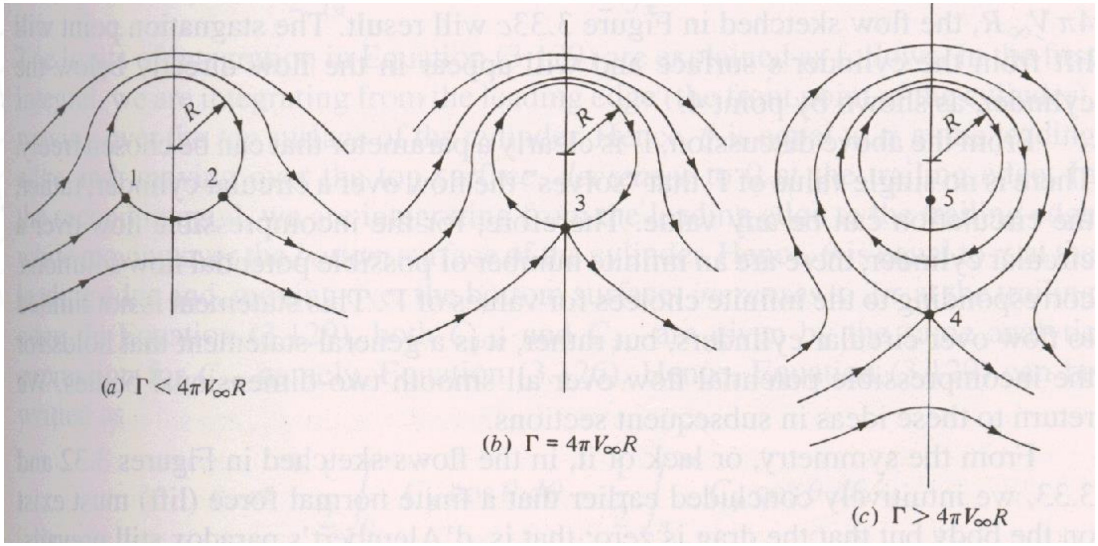
(Note)

"Vortex sheet method" is more than just a mathematical device; it also has a physical meaning

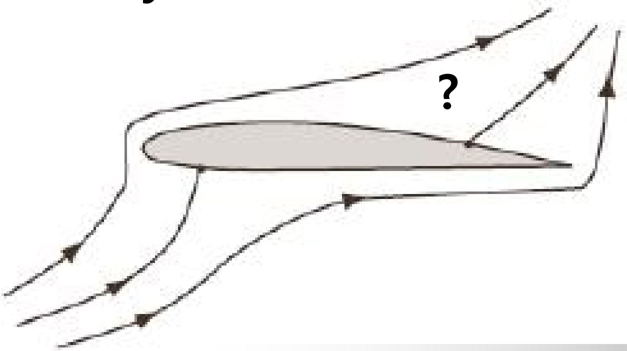
ex. : Replacing the boundary layer ($\nabla \times \vec{V} \neq 0$) with a vortex sheet

< 4.5 The Kutta Condition >

* For a circular cylinder,

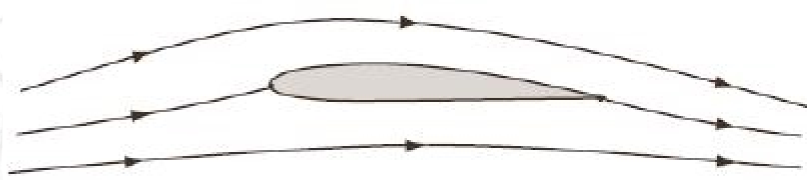


* For a given α, ➔ should have only one solution



< 4.5 The Kutta Condition >

* From the experiments, we know that the velocity at the trailing-edge in finite. ➔ Kutta Condition



$\gamma(TE)=V_1-V_2=0$

$V(TE)=$finite

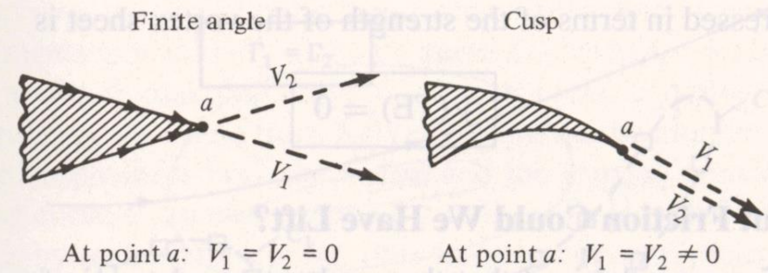


Figure 4.19 Different possible shapes of the trailing edge and their relation to the Kutta condition.

* The <u>circulation</u> around the airfoil is the value to ensure that the flow smoothly leaves the trailing edge.

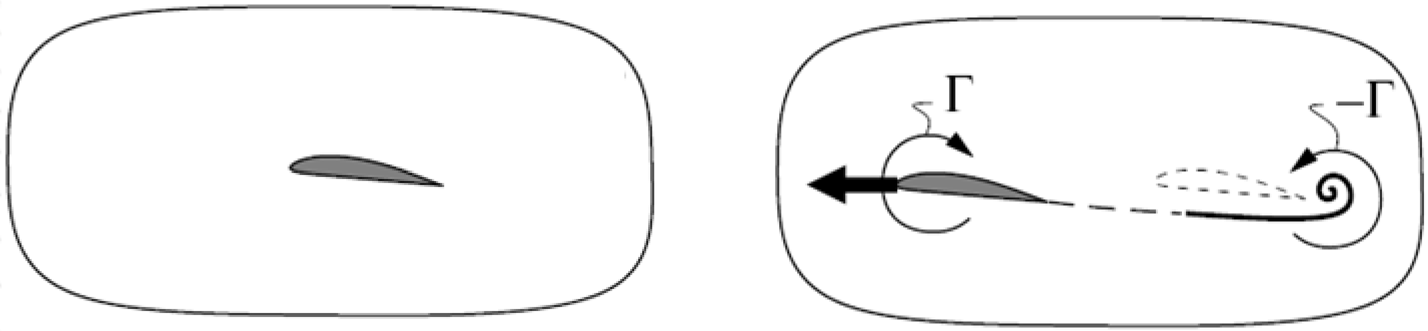
< 4.6 Kelvin's Circulation Theorem >

* Assume)
1. Inviscid
2. Incompressible
3. No body forces

$$\frac{D\Gamma}{Dt} = 0$$

➔ The time rate of change of circulation around a closed curve consisting of the same fluid elements is zero

Ex) Starting vortex



[at rest] ➔ [after the start]