## Viscous flow Types of drags:

Pressure drag: Mostly affect bluff bodies. It causes high pressure upstream and low pressure downstream, this is due to flow separation. Acts perpendicular to the surface

Flow separation: Where the streamlines can no longer follow the curvature of the of the body.

CHAPTER 4 Basic Aerodynamics


Frictionless flow: no drag
Figure 4.37 Comparison between ideal frictionless flow and real flow with the effects of friction.

Skin friction drag: Is produced by the friction of the air molecules with the surface which creates a shear stress at the surface. This acts in a direction tangential to it.


Boundary layer: Is the region of fluid that is affected by viscosity. At this region, the fluid velocity is retarded, and right at the surface, the flow velocity is zero. The B.L has an impact on viscous forces, as well as pressure forces. As you get farther away from the surface the velocity increases, until you reach he edge of the B.L where the velocity equals the local flow velocity. The BL thickness grows as the flow moves over the body.


[^0]

The shear stress at the wall is given by $\mathrm{T}=\mu(\mathrm{dV} / \mathrm{dy}) \mathrm{y}=0$ $\mu$ is the viscosity of the gas, which varies with $T$

$$
\tau_{\omega}=\mu\left(\frac{d V}{d y}\right)_{y=0}
$$

For air at standard sea-level temperature:

$$
\begin{gathered}
\text { SI: } \mu=1.7894^{\star} 10^{\wedge}-5 \mathrm{~kg} /(\mathrm{m})(\mathrm{s}) \\
\text { English: } \mu=3.7373 \times 10-7 \text { slug /(ft)(s) }
\end{gathered}
$$

Viscosity can be calculated:

$$
\begin{aligned}
& \mu=1.458\left(\frac{T^{3 / 2}}{T+11.04}\right) \times 10^{-6} \mathrm{~kg} / \mathrm{m} \cdot \mathrm{~s} \\
& \mu=2.27\left(\frac{T^{3 / 2}}{T+199}\right) \times 10^{-8} \quad \text { succors } / 1 \mathrm{~b} \cdot \mathrm{~s}
\end{aligned}
$$

Reynolds number: Non dimensional parameter. Describes the behavior of viscosity. High Reynolds number indicates low viscosity, and low Reynolds number indicates high viscosity.

* measured from the leadiarg edge
* Dimensionless varies witor $x \rightarrow$ Givinct cocos REYNODS NuMBER

$$
R_{e_{x}}=\frac{\rho_{\infty} V_{\infty} x}{\mu_{\infty}}
$$

AIRFOIL:


PLATE:


CUCLINDER:


Types of Flows:

Laminar flow: Streamlines are smooth and regular, and a fluid element moves smoothly along a streamline.

Boundary layer thickness


$$
\delta=\frac{5.2 x}{\sqrt{R_{x}}}
$$

Total Skin friction coefficient

$$
C_{x}=\frac{0.664}{\sqrt{R_{e x}}} \text { LocAL } \quad C_{f}=\frac{1.328}{\sqrt{R_{e_{L}}}} \text { torn }
$$

Turbulent flow: The streamlines break up and a fluid element moves in a random, irregular fashion.
Boundary layer thickness.

$$
f=\frac{0.37 x}{\operatorname{Re}_{x}{ }^{0.2}}
$$

Total Skin friction coefficient

$$
C_{f x}=\frac{0.0592}{\left(R_{e x}\right)^{0.2}} \text { cocos } \quad C_{f}=\frac{0.074}{R_{e_{i}}^{0.2}} \text { Torn }
$$

Laminar shear stress is less than the turbulent shear stress. Therefore, the skin friction is higher for turbulent flow. Turbulent boundary layer is thicker and grows faster. $\tau_{\omega}$ Laminar $<\tau_{\omega \text { turement }}$

The flow always starts out from the leading edge as laminar, and then at the transition point the boundary layer becomes completely turbulent where the boundary layer grows at a faster rate.

Critical Reynolds Number: This point where transition occurs is called the critical point, which corresponds to a critical Reynolds number.

$$
\operatorname{Recr}_{\text {cr }}=\frac{\rho v x_{\text {can }}}{\mu}
$$



Airfoil nomenclature: NACA Airfoils
4-digit series: NACA $\times \times \times$
1 digit: max camber
2nd :location of max camber
3rd and 4th thickness of airfoil

5-digit series: NACA $\times \times \times \times \times$
1st digit: design lift coefficient, multiplied by 3/20
2nd digit: max camber, divided by 20
Ord digit: 0 refers to normal camber, 1 refers to reflex camber
4th and 5th: thickness

There is also a 6th digit series

Example: Given the NACA 4510, what is the max camber and where does it occur?


Max Chamber:

$$
4 \%
$$

Max Chamber Location:

$$
50 c / 0
$$

NACA charts:

-Angle of attack
-Cl (does not depend on Re unless we want to know Cl max)
-Cl with flaps
-aC: lift curve slope for Cl vs alpha. It's value is $\frac{2 \mathrm{pin}}{\mathrm{rad}} 0.11 / \mathrm{deg}$
$-A l p h a L=0$. This is the angle where the lift equals zero. It equals zero degrees for a symmetric airfoil.
-Stall: corresponds to the point where we have a max Cl and you get a dramatic loss of lift.
-Pitching moment coefficient about the quarter cord Cmc/4-Drag polar (Cl vs Cd)
-Cd
-Pitching moment coefficient about the aerodynamic center Cmac
-Cdmin


```
* GIVEN FOR
        2D INFINITE
        UING.
* hale to conlert
USINEN EQURTIONS
        FOR REAL
3D wings.
```


(7)

For a 2D infinite airfoil ( $\mathrm{Cl}, \mathrm{Cd}, \mathrm{Cm}$ ) and a 3D finite wing (CL,CD,CM), the lift and drag coefficient are different.

This is because for an airfoil section, the end effects are removed when testing in a wing tunnel.
For a 3D wings these end effects produce a downward component called downwash. This causes an induced drag, which increases the total drag and reduces the lift.

Downwash causes the relative wind in the proximity of the airfoil section to be inclined slightly downward through a small angle called the induced angle of attack. This in turn reduces the angle of attack felt by the local airfoil section to a value smaller than the geometric angle of attack. This smaller angle of attack is called the effective angle of attack. The effective angle of attack for a 3D wing is equivalent to the geometric angle of attack for a 2D airfoil.

2 Main things going from 2D to 3D

1. Induced drag must be added to the finite wings:

2. The slope of the life curve for a finite wing is less that that for an infinite wing: $a<a 0$

INDUCED DRAM COEFFICIENT: $C_{D i}=\frac{C^{2}}{\text { TARE }}, A R=\frac{b^{2}}{s}$

INDUCED DRAM: $D_{i}=q_{00} \frac{C^{2}}{\text { TARE }}, \quad q_{00}=\frac{1}{2} \rho_{\infty} V_{\infty}^{2}$
TOTAL DRAM COEFICLENT: $C_{D}=C_{d}+C_{D i}$


CHANGE IN LIFT SLOPE

General span efficiency factor: e

$$
\begin{aligned}
& \alpha_{i}=\frac{G}{\pi e_{1} A R} \text { (RADIANS) } \\
& \alpha_{i}=\frac{57.3 C}{\pi C_{1} A R} \text { (DEGREES) }
\end{aligned}
$$

* FLOL over a finite cong at an anole of attack ' $\alpha$ ' IS THE SAME AS FLOL OUER A INFINITE wING At an anile of ATtACK ' $\alpha_{\text {eff' }}$.

wHERE $a_{0}=\frac{d C_{L}}{d\left(\alpha-\alpha_{i}\right)}$ (ID)

$$
\begin{gathered}
a=\frac{a_{0}}{1+57.3 a_{0} /(\pi e, A R)} \quad(3 D) \\
C_{L}=a_{2}=a\left(\alpha-\alpha_{L=0}\right) \\
(9)
\end{gathered}
$$

PRESSURE COEFFICIENT:

$$
\begin{aligned}
& C_{p}=\frac{P-P_{\infty}}{q_{\infty}}=\frac{P-P_{\infty}}{\frac{1}{2} P_{0} V_{\infty}^{2}} \\
& C_{P}=1-\left(\frac{V}{V_{\infty}}\right)^{2} \text { incompeessibivith }
\end{aligned}
$$



PRANDTL - GILALRT COMPREESSIBIUTY:

* accurate for $0.3<m_{\infty}<0.7$

$$
C_{p}=\frac{C_{p, 0}}{\sqrt{1-m_{\infty}^{2}}}
$$

- compressibility correction dove to HELP DREDIET THE EFFECT OF Mos on $C$.

COMPRESSIBILITY COERRECTION FOR coefficient of lift:

$$
C_{l}=\frac{C_{l m=0}}{\sqrt{1-m_{\infty}^{2}}}
$$

$$
(10)
$$

Example 1:
A model wing of a constant chord length is placed in a low speed subsonic wind tennis, spanning the test section. The wing has an NACA 2412 airfoil and chord length of 1.3 m . The flow is in the test section at a velocity of $50 \mathrm{~m} / \mathrm{s}$ at a standard sea level conditions. If the wing is at an angle of attack of $4^{\circ}$, calculate $\mathrm{cl}, \mathrm{cd}$, and $\mathrm{cm}, \mathrm{c} / 4$ and the lift, drag and moments about the quarter chord per unit span.

$$
\text { given: Naker } 2412
$$

$$
c=1.3 \mathrm{~m}
$$

$$
v_{\infty}=50 \mathrm{~m} / \mathrm{s}
$$

$$
\alpha=4^{\circ}
$$

Sear Icuc\ Conditions

$$
\begin{aligned}
& \rho_{\infty}=1.225 \quad 16 \mathrm{~g} / \mathrm{m}^{3} \\
& \mu=1.784 \times 10^{-5} \mathrm{~kg} / \mathrm{m} \cdot \mathrm{~s}
\end{aligned}
$$

Find $C_{1}, C_{m}, c / 4$

$$
\begin{aligned}
& \text { Apecudix D. NACA 2412, } \alpha=4^{\circ} \\
& C l=0.63 \\
& \mathrm{~cm}, \mathrm{c} / 4=-0.035
\end{aligned}
$$

$$
C d=0.007
$$

$$
R e=p_{0} V_{\infty} e^{-e q \cdot 4.9}
$$

$$
R_{c}=\frac{\left(1.225 \mathrm{~kg} / \mathrm{m}^{3}\right)(50 \mathrm{~m} / \mathrm{s})(1.3 \mathrm{~m})}{\left(1.289 \times 10^{-5} \mathrm{ky} / \mathrm{m} . \mathrm{s}\right)}
$$

$$
R_{e}=4.45 \times 10^{6}
$$

Dray, Lift, moment

$$
S=C(1)=1.3(1)=1.3 \mathrm{~m}^{2}
$$

Dynamic Pressure

$$
q_{\infty}=\frac{1}{2} y_{\infty} v_{\infty}^{2}=\frac{1}{2}\left(1.225 \mathrm{ky} / \mathrm{m}^{3}\right)(50 \mathrm{mcs})^{2}
$$

$$
\begin{aligned}
& q_{\infty}=1531 \mathrm{~N} / \mathrm{m}^{2} \\
& L=q_{0} S C_{V}=(1531)(1.3)(0.63)=1254 \mathrm{~N} \\
& D=q_{0 s c d}=(1531)(1.3)(0.007)=D=13.9 \mathrm{~N} \\
& m_{c / 4}=q_{\infty} S c_{m, c / 4} C= \\
& =1531(1.3)(-0.035)(1.3)=-90.6 \mathrm{~N} \cdot \mathrm{~m}
\end{aligned}
$$

Example 2:
The wingspan of the Wright Flyer I biplane is 40 ft and 4 inches, and the plan dorm area of each wing is $255 \mathrm{ft} \wedge 2$. Assume that the wing is rectangular. If the aircraft is flying with a velocity of $30 \mathrm{mi} /$ hr at standard sea level conditions. Calculate the skin friction drag and the wings. Assume the transition Reynolds number is $6.5 \times 10^{\wedge} 5$. The areas of laminar and turbulent flow are illustrated by areas $A$ and $B$ below.


$$
\begin{aligned}
& b=40 \mathrm{ft} 4 \mathrm{in} \rightarrow 40.33 \mathrm{ft} \\
& S=A+B=255 \mathrm{ft}^{2} \\
& c=s / b \Rightarrow 255 / 40.33=6.32 \mathrm{ft}
\end{aligned}
$$

$$
\begin{aligned}
& v_{\infty}=30 \mathrm{mi} / \mathrm{hr} \rightarrow 30(88 / 60)=44 \mathrm{ft} / \mathrm{s} \\
& R_{c_{c}}=\frac{\rho_{\Delta} v_{\infty} c}{\mu_{\infty}}=\frac{0.002377(44)(6.32)}{3.7373 \times 10^{-7}}=1.769 \times 10^{6} \\
& R_{c o r}=\frac{\rho_{\infty} v_{\infty} X_{c r}}{\mu_{\rho}} \\
& x_{c r}=\frac{\operatorname{Recr}_{c r} \mu_{\infty}}{\rho_{\infty} w_{\infty}}=\frac{6.5 \times 10^{5}\left(3.2373 \times 10^{-7}\right)}{(0.002377)(44)}=2.32 \mathrm{ft}
\end{aligned}
$$

(1) calculate turblent flow over entire wing, $S$

$$
\begin{aligned}
& C_{f}=\frac{0.079}{\operatorname{Re}_{c}^{0.2}}=\frac{0.074}{\left(1.769 \times 10^{6}\right)^{0.2}}=0.00417 \\
& q_{\infty}=\frac{1}{2} \rho_{\Delta} v_{\infty}^{2}=\frac{1}{2}(0.002377)(44)^{2}=2.3016 / \mathrm{ft}^{2} \\
& \left(D_{f}\right)_{s}=q_{\infty} S C_{f}=2.30(255)(0.00417)=2.44616
\end{aligned}
$$

(2) Calculate $\left(D_{f}\right)_{A}$ assuming turbulent flow

$$
\begin{aligned}
& C_{f}=\frac{0.074}{R e_{c r}^{0.2}}=\frac{0.074}{(6.5 \times 105)^{0.2}}=0.00509 \\
& \left(D_{f}\right)_{A}=q_{\infty} A C_{f}=2.30\left(\frac{(2.32 \times 40.33)}{\uparrow}(0.00509)=1.09516\right.
\end{aligned}
$$

Dray on Part B: (turbulent)
$\left(D_{f}\right)_{B}=\left(D_{f}\right)_{S}-\left(D_{f}\right)_{A}=2.44616-1.09516=1.35116$
(3) Calculate dry on part $A$, laminar

$$
\begin{aligned}
& C_{f}=\frac{1.328}{\operatorname{Re}_{c r}{ }^{0.5}}=\frac{1.328}{\left(6.5 \times 10^{5}\right)^{0.5}}=0.00165 \\
& \left(D_{f}\right)_{A}=q_{\infty} A C_{f}=2.30(2.32 \times 40.33)(0.00165)=0.35416
\end{aligned}
$$

Total dray acting on one surface of one wing:

$$
\begin{aligned}
D_{f}= & \left(D_{f}\right)_{A}+\left(D_{f}\right)_{B} \\
& \uparrow \quad \text { laminar turbulent } \\
& \quad 0.35418+1.35116=1.70516
\end{aligned}
$$

Total dry acting on wings (on both top \& bottom surfaces):

$$
D_{\text {fatal }}=(1.70516)(4)=6.82016
$$

Example 3：
Consider an infinite wing with a NACA 1412 airfoil section and a chord length of 3 ft ．The wing is at an angle of attack of $5^{\circ}$ in an airflow velocity of $100 \mathrm{ft} / \mathrm{s}$ at standard sea level．Calculate the lift drag and moment about the quarter chord per unit span．

NACA 1412
－cころトか
－$\alpha=s^{\circ}$
oincinite winy

$$
\begin{aligned}
& S L-c o n d i t i o n s \\
& 9 \rho=2.3769 .20^{-3} \frac{s(u g}{t+3} \\
& \cdot \mu_{\infty}=3.7373 .10^{-7} \frac{\mathrm{Ls} \cdot \mathrm{~s}}{\mathrm{ft}^{2}}
\end{aligned}
$$

Find：
－Ll Per unit span
$-D^{\prime}$
－Mo．256

$$
R_{e}=\frac{\rho_{\infty} V_{\infty} c}{\mu_{\infty}}=1.9 .10^{6}
$$

From data：

$$
\begin{aligned}
& C l l_{l}(\alpha=5)=0.67 \quad c_{d}\left(C_{Q}=0.67\right)=0.007 \\
& \epsilon_{M_{0.25 C}}\left(\alpha-5^{0}\right)=0.025 \\
& h^{\prime}=q C C_{\theta} \quad q_{\infty}=\frac{1}{2} \rho_{\infty} V_{\infty}^{2}=11.8845 \frac{\mathrm{lb}}{\mathrm{ff}} \\
& D^{\prime}=q<C_{d} \\
& M_{0.2 s c}^{\prime}=q c^{2} c_{M 0.2 s}
\end{aligned}
$$

$$
\begin{aligned}
& L^{\prime}=23.89 \frac{15}{f t} \quad M_{0.25 c^{\prime}}^{\prime}=-2.67 \frac{15 \mathrm{ft}}{\mathrm{ft}} \\
& D^{\prime}=0.25 \frac{16}{\mathrm{ff}}
\end{aligned}
$$

Example 4:
Given a NACA 4415 in a high speed, subsonic wind tunnel with a $C_{\ell}$ measured as 0.85 at a Mach number of 0.7 . Find alpha.

$$
\text { Find } \begin{aligned}
& c_{10} \\
& c_{l}=\frac{C_{l 0}}{\sqrt{1-M_{\infty}^{2}}} \\
& c_{l_{0}}=C_{l} \sqrt{1-M_{\infty}^{2}} \\
& c_{l_{0}}=.85 \sqrt{1-.7^{2}} \\
& c_{l 0}=.607 \\
& \alpha=2^{\circ}
\end{aligned}
$$

Example 5:
Consider a Cessna Cardinal, with wing area of $16.2 \mathrm{~m}^{\wedge} 2$, an aspect ratio of 7.31 , a span efficiency factor of 0.62 and a weight of 9800 N . Solve for the aircrafts induced drag

$$
\begin{aligned}
& V_{\infty}=251 \mathrm{~km} / \mathrm{hr} \\
& D_{i}=\frac{1}{2} \rho_{\infty} V_{\infty}^{2} S C_{D i} \\
& \qquad C_{D i}=\frac{C_{L}^{2}}{\pi e A R}
\end{aligned}
$$

$$
\begin{aligned}
& L=\frac{1}{2} \rho_{-} V_{\infty}^{2} S C_{L}=W \\
& C_{L}=\frac{W}{1 / 2 \rho_{\infty} V_{-}^{2} S} \\
& C_{L}=\frac{9800}{1 / 2 \cdot 1.225 \cdot\left(251 \cdot \frac{1000}{3620}\right)^{2} \cdot 16.2} \\
& C_{L}=.2032
\end{aligned}
$$

$$
\begin{aligned}
& D_{i}=\frac{1}{2 \rho_{a}} v_{2}^{2} s \cdot \frac{c_{i}}{\pi_{e} A R}=\frac{1}{2}(1.225)\left(251 \cdot \frac{1000}{3600}\right)^{2}(16.2)\left(\frac{.2032^{2}}{\pi(.62)(7.31)}\right) \\
& D_{i}=139.8 \mathrm{~N}
\end{aligned}
$$


[^0]:    Figure 4.39 Flow in real life, with friction. The thickness of the boundary layer is greatly overemphasized for clarity

