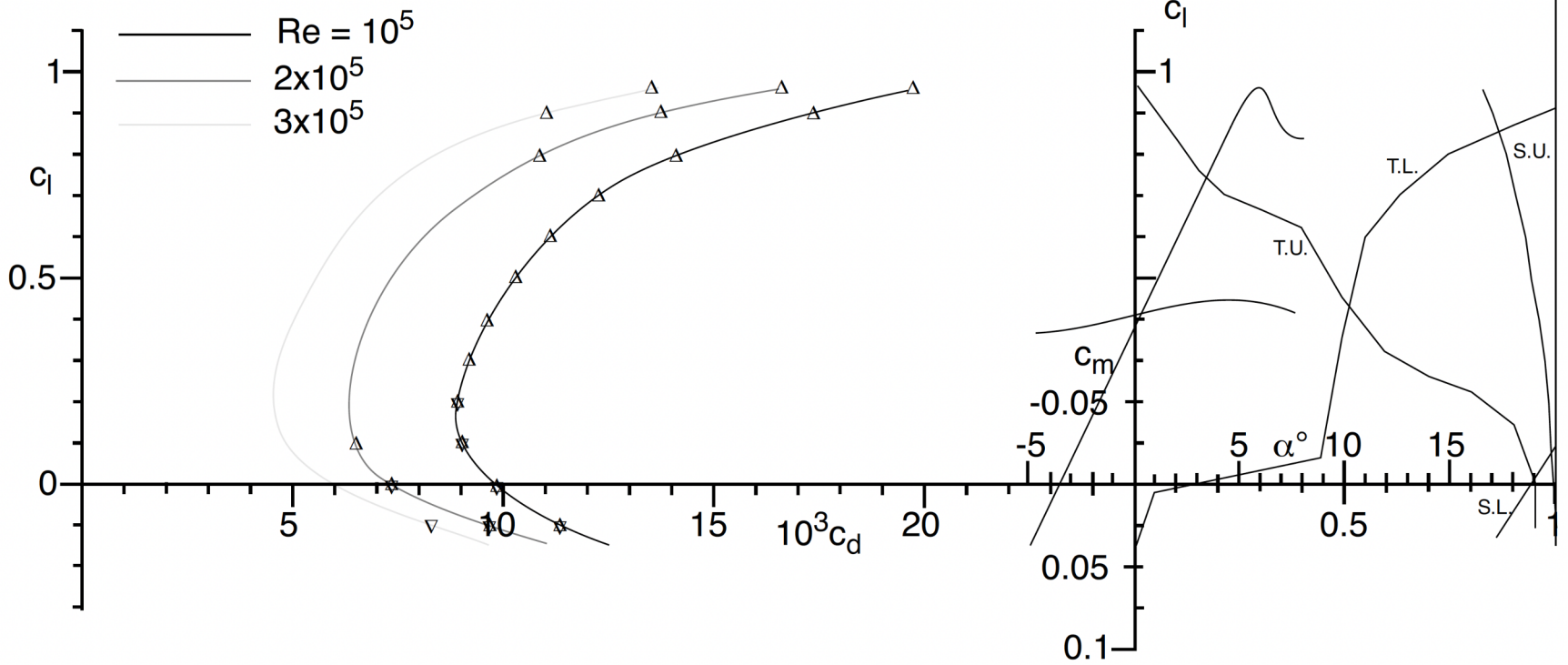
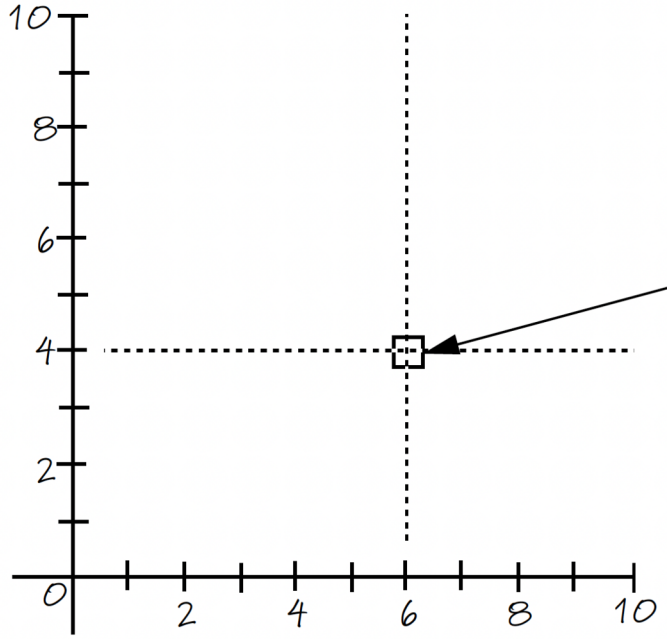


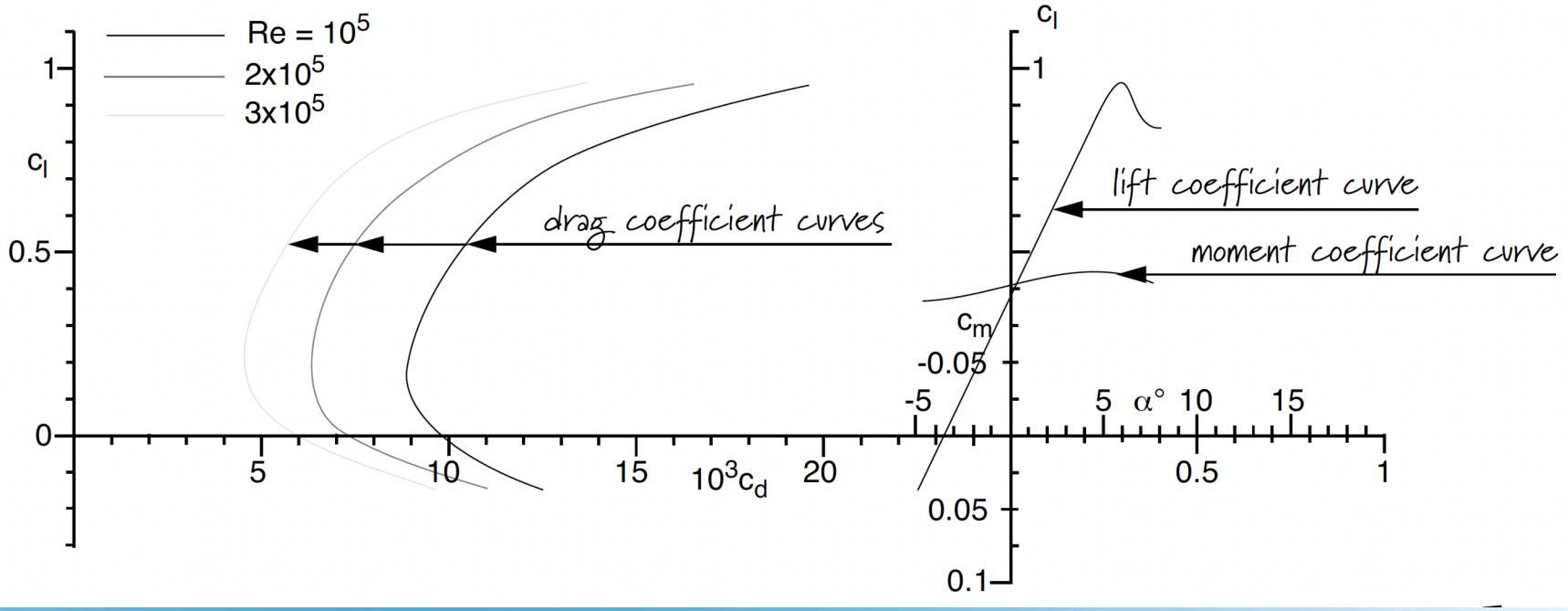
Understanding Polars Without Math
Part 1

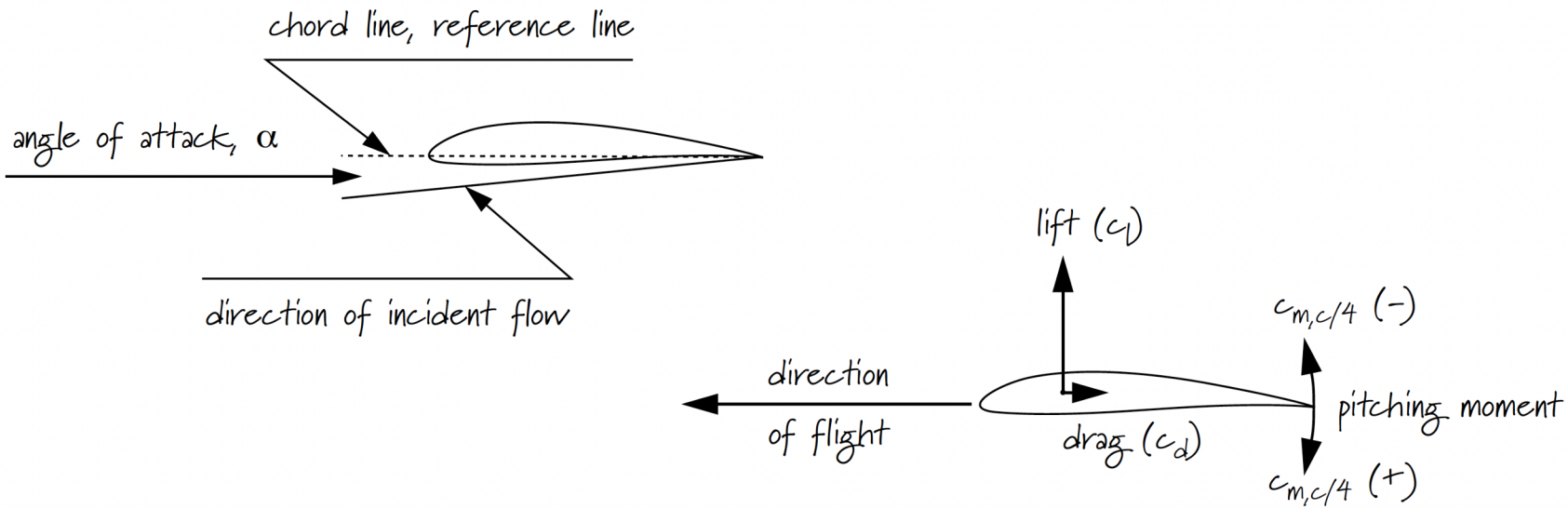
- Overview


Re = 10^5
2x10^5
3x10^5
c_l
1
0.5
0
5
10
15
10^3c_d
20
c_l
1
c_m
-0.05
0.05
0.1
-5
5
α°
10
15
0.5
1
T.L.
S.U.
T.U.
S.L.

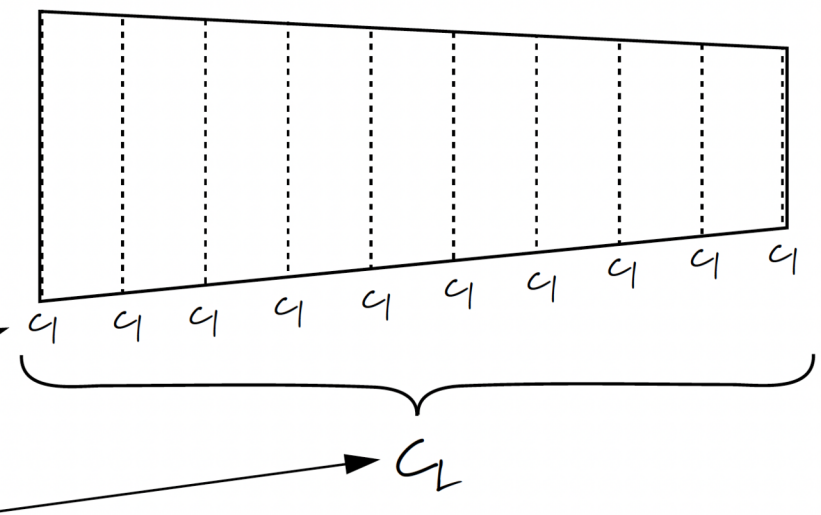


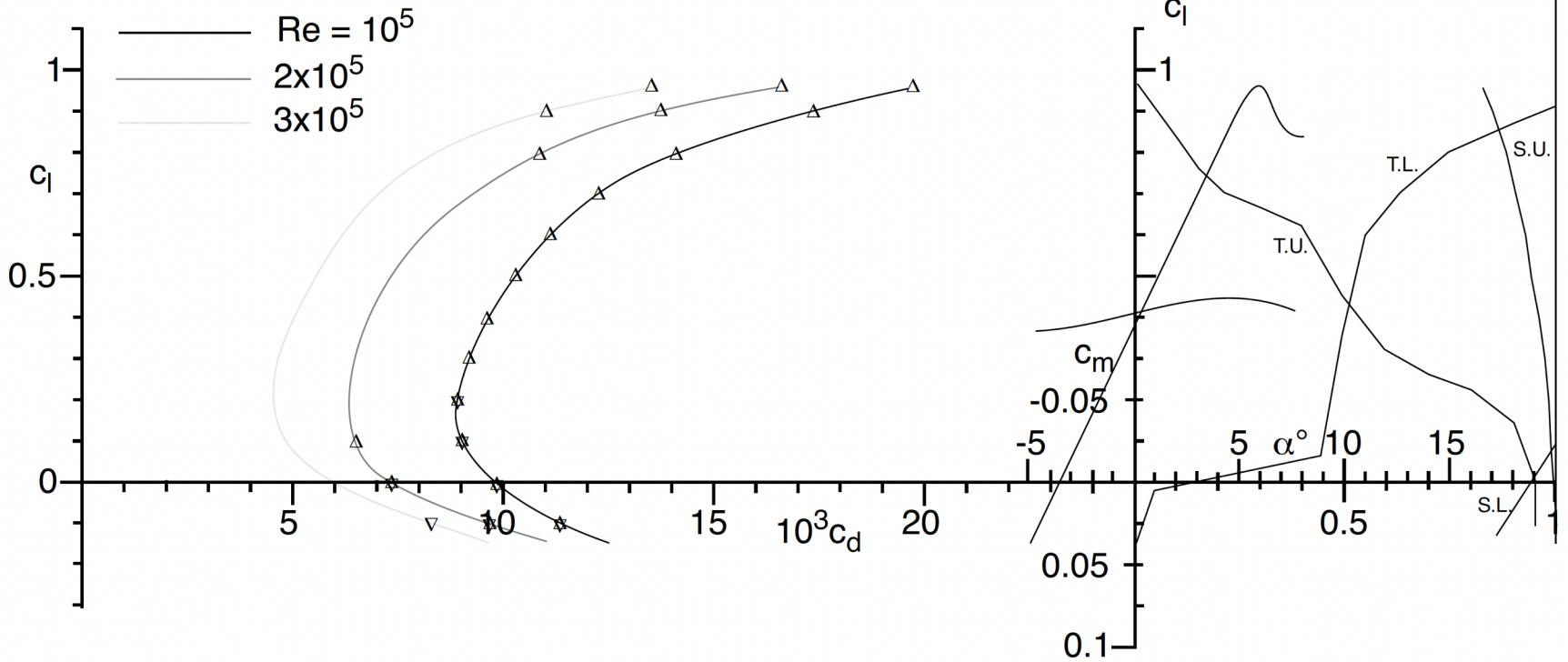
The relationship between the square's coordinates and its position is a reversible one. One can take the values 6 and 4 and locate the square, or can start with the square itself and find its coordinates, (6,4).


Re = 10^5
2x10^5
3x10^5
c_l
1
0.5
0
5
10
15
10^3c_d
20
drag coefficient curves
c_l
1
lift coefficient curve
moment coefficient curve
c_m
-0.05
-5
5
α°
10
15
0.5
1
0.05
0.1

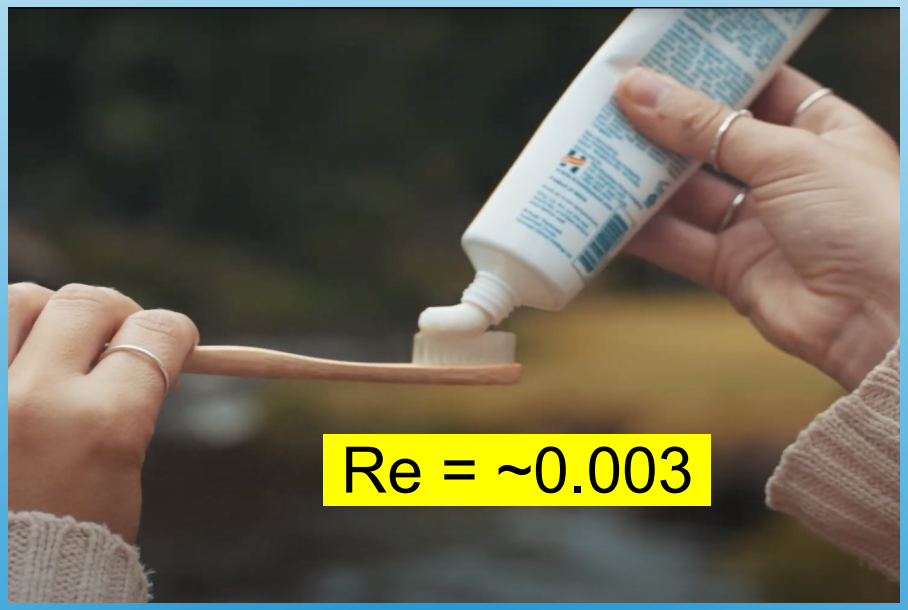

chord line, reference line
angle of attack, α
direction of incident flow
lift (c_l)
$c_{m,c/4}$ (−)
direction
of flight
drag (c_d)
pitching moment
$c_{m,c/4}$ (+)

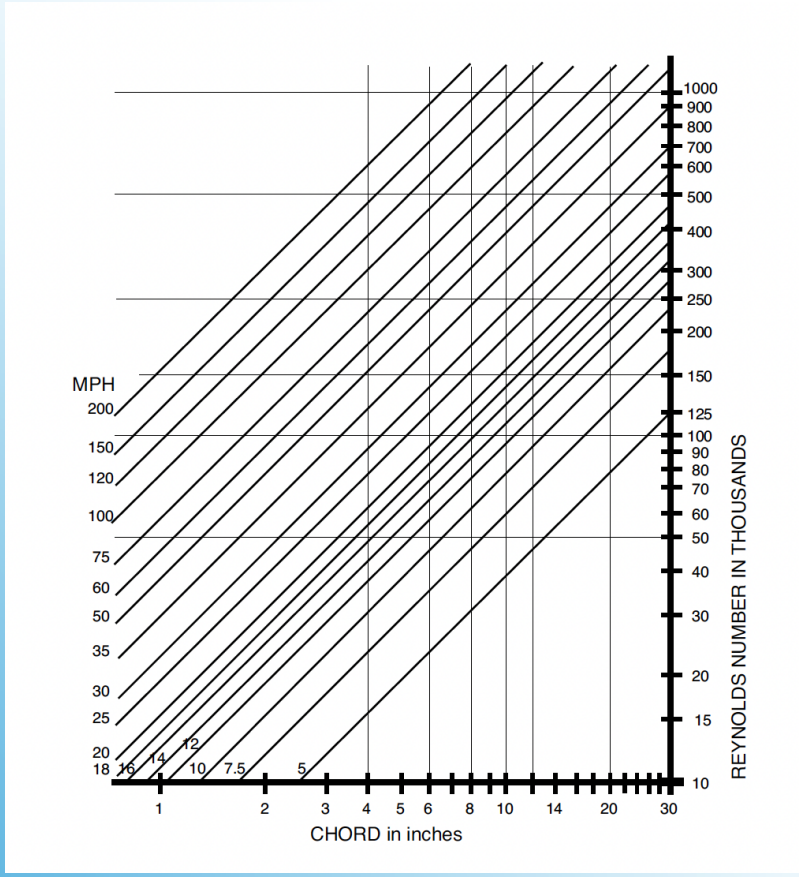
There is a difference between a section and an airfoil or flying surface (like a wing). A section consists of only two dimensions, height (thickness) and length (chord), while an airfoil or a flying surface is defined by three dimensions, height (thickness), length (chord) and width (span). Section coefficients are designated by the use of lower case letters (c_l), while coefficients for airfoils, wings, and entire aircraft are designated by the use of upper case letters (C_L).

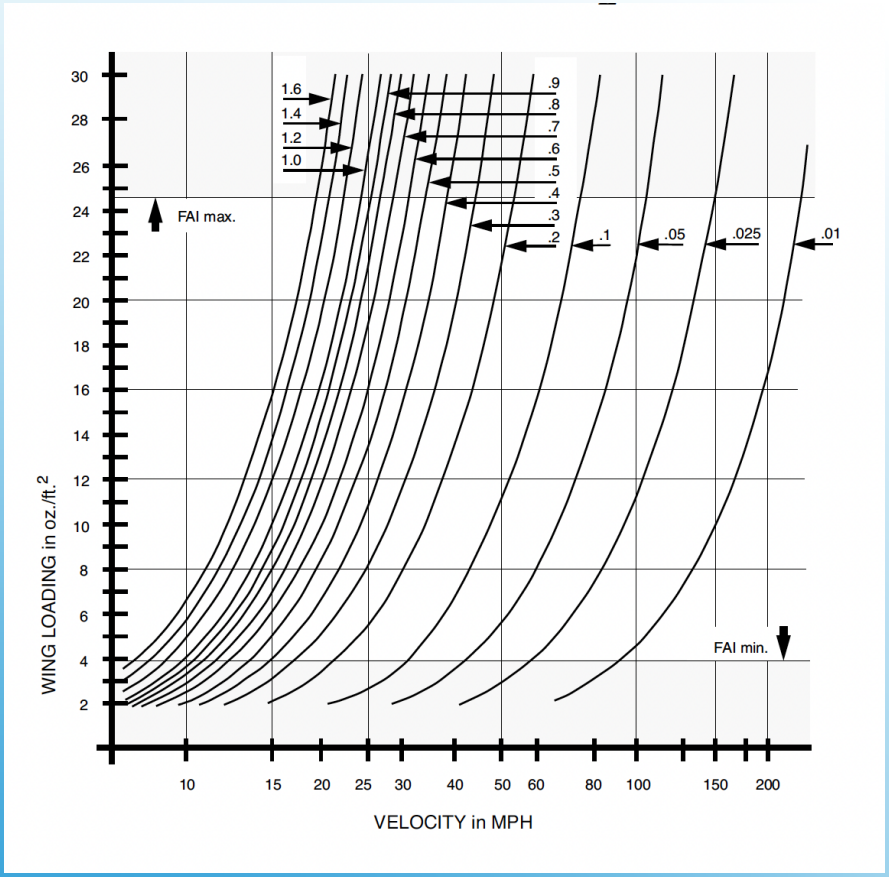



Re = 10^5
2x10^5
3x10^5
c_l
1
0.5
0
5
10
15
10^3c_d
20
c_l
1
T.L.
S.U.
T.U.
c_m
-0.05
-5
5
α°
10
15
0.5
S.L.
0.05
0.1

The wing root
of an airliner flying at 500 mph
at 30,000’
has a Reynolds number
of 12-18 million



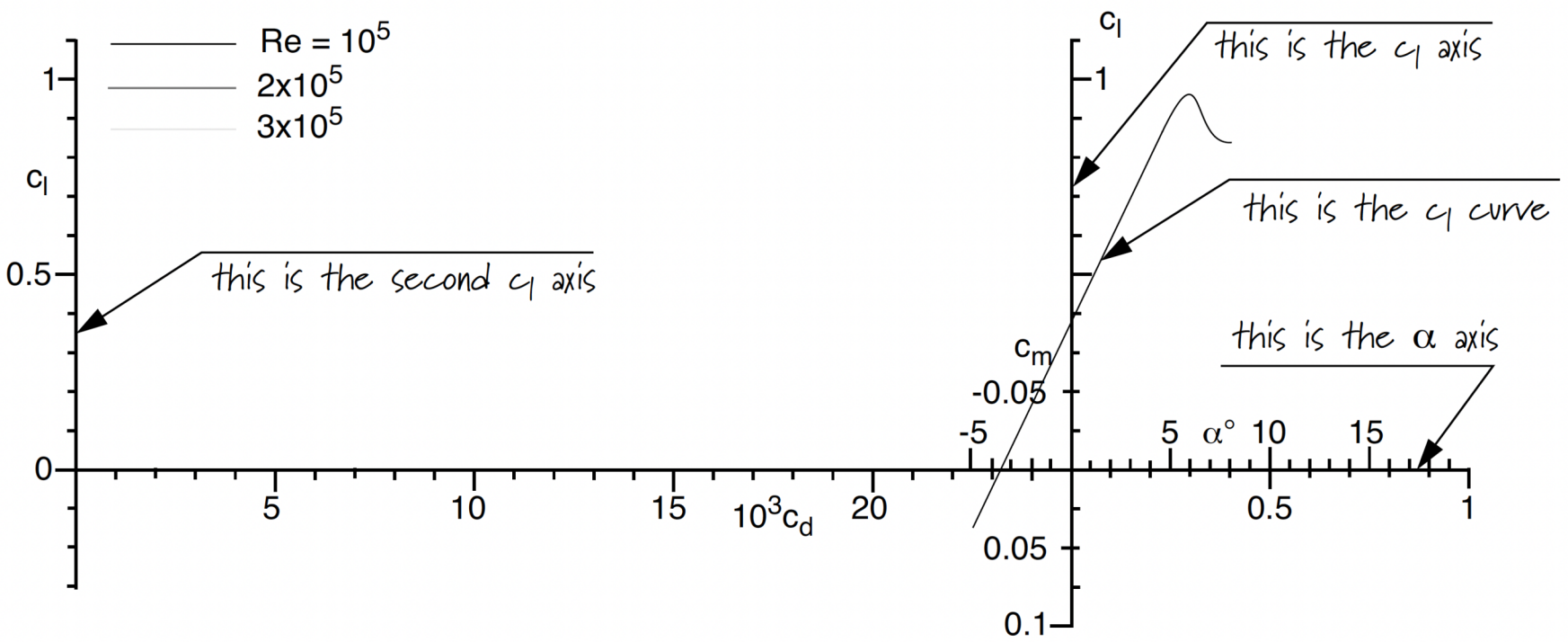


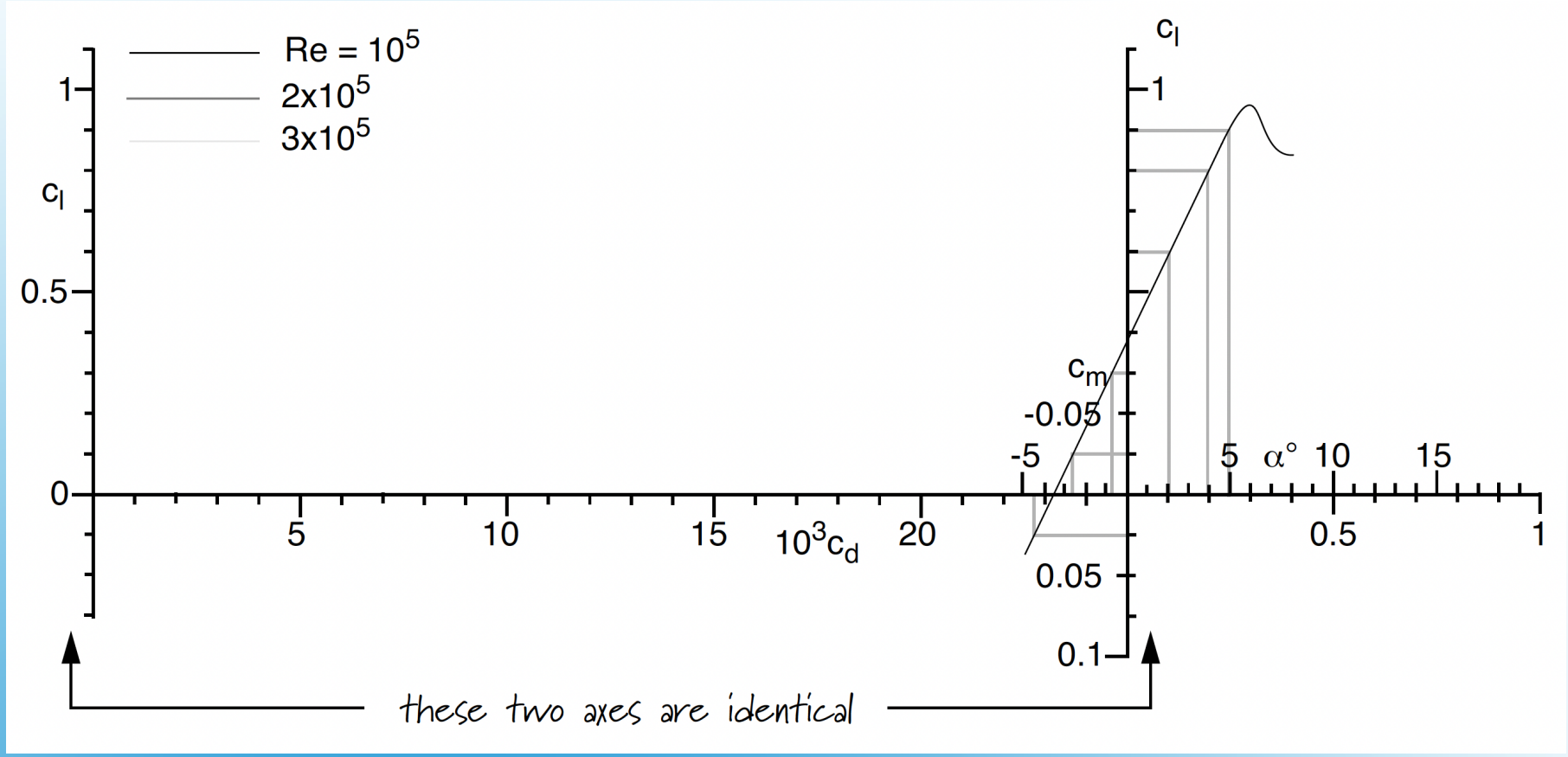

WING LOADING in oz./ft.2
30
28
26
24
22
20
18
16
14
12
10
8
6
4
2
1.6
1.4
1.2
1.0
.9
.8
.7
.6
.5
.4
.3
.2
.1
.05
.025
.01
FAI max.
FAI min.
10
15
20
25
30
40
50
60
80
100
150
200
VELOCITY in MPH

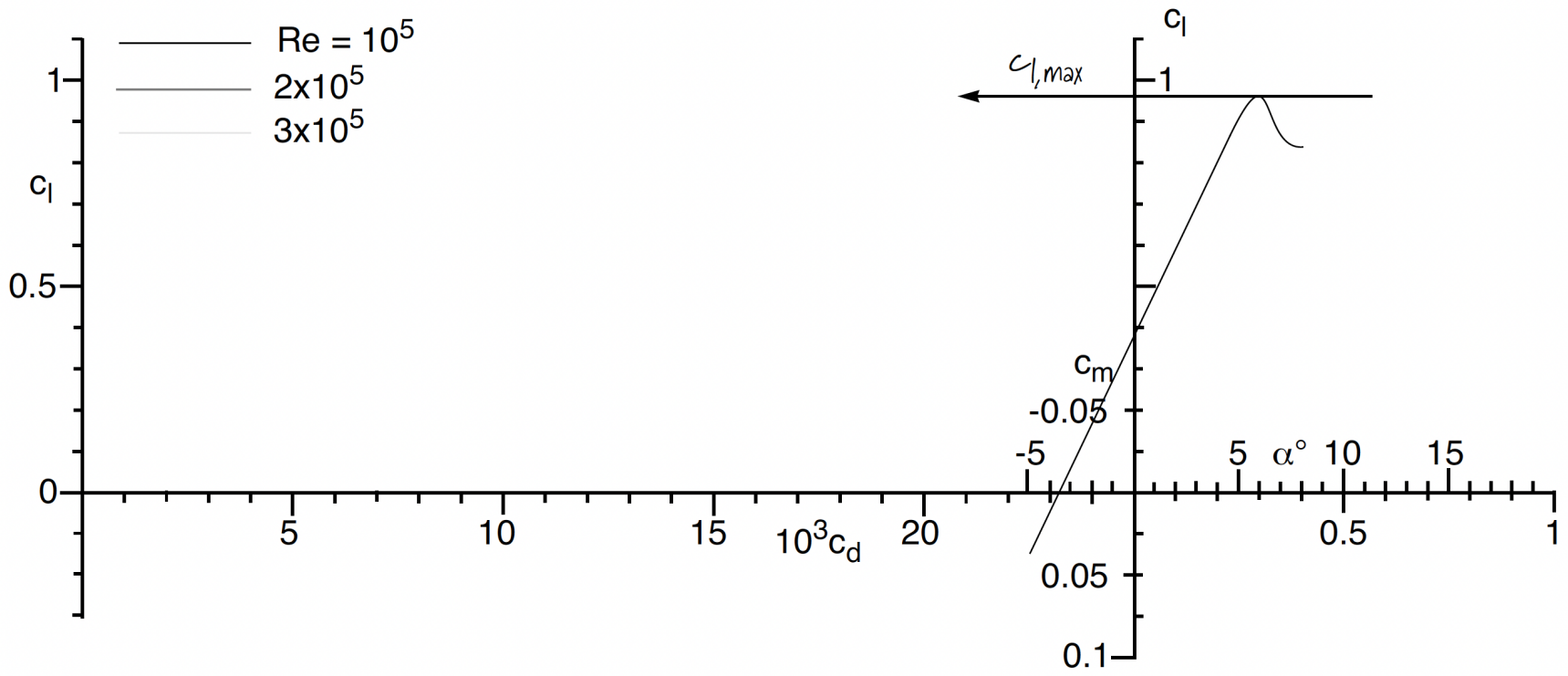
The Basic Curves

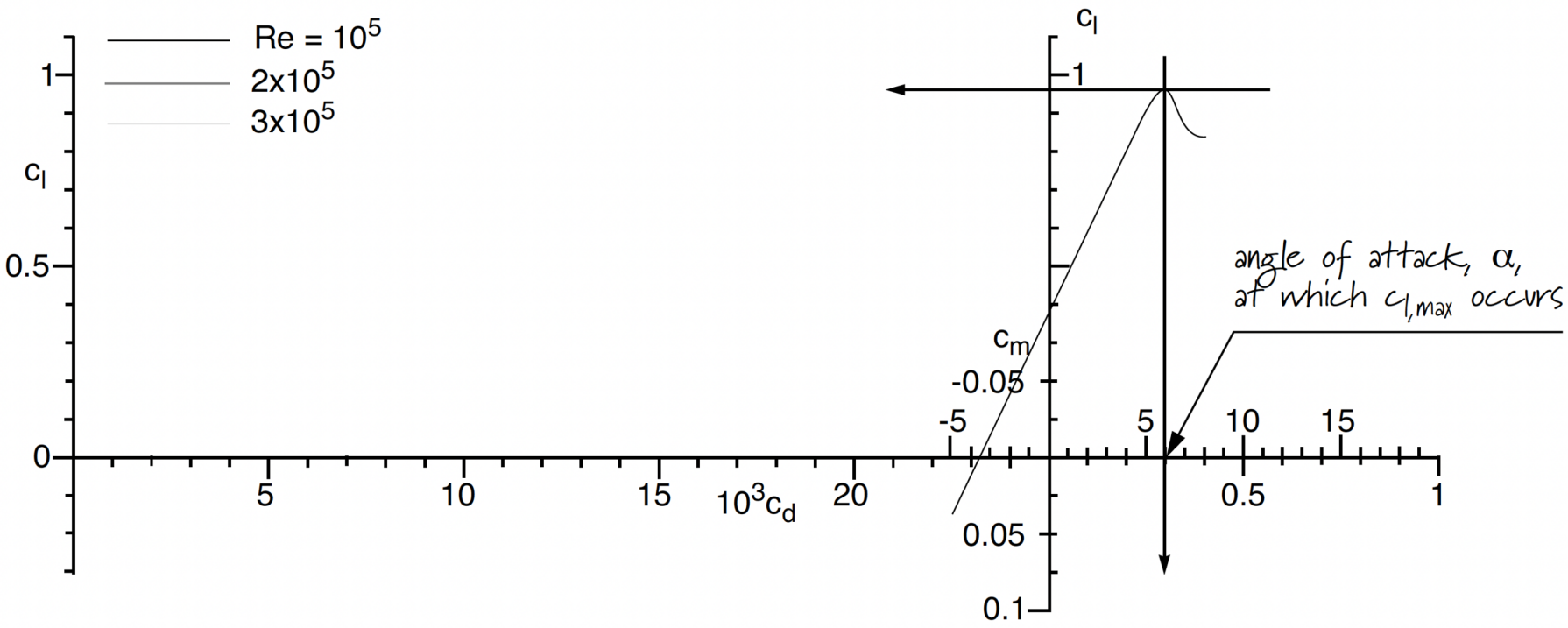
- Lift coefficient
- Pitching moment
- Drag coefficient

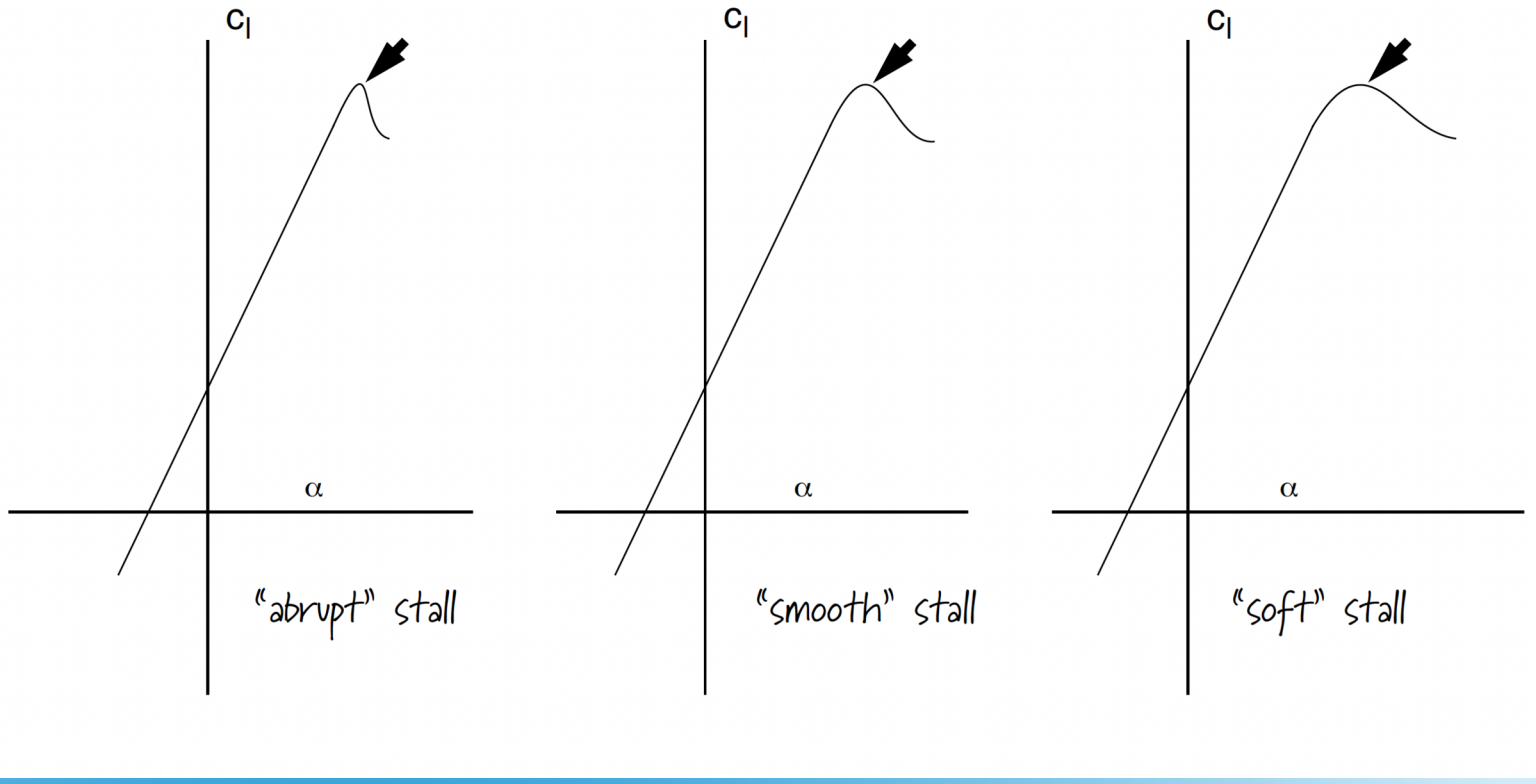
- The Lift Coefficient (c_l) Curve

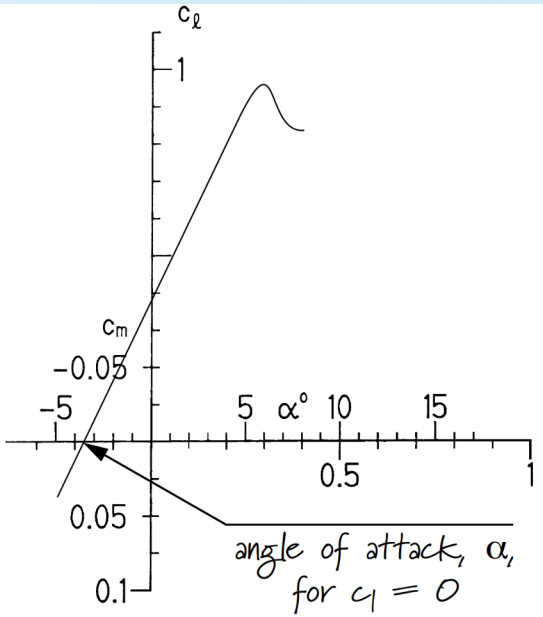

Re = 10^5
2x10^5
3x10^5
c_l
1
0.5
0
this is the second c_l axis
5
10
15
10^3c_d
20
c_l
1
this is the c_l axis
this is the c_l curve
this is the α axis
c_m
-0.05
-5
5
α°
10
15
0.5
1
0.05
0.1

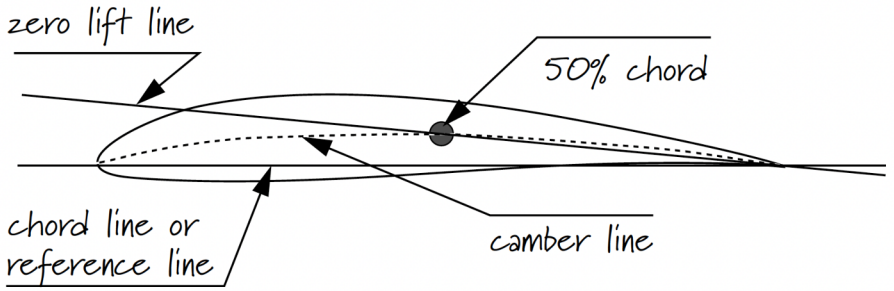

Re = 10^5
2x10^5
3x10^5
c_l
1
0.5
0
5
10
15
10^3c_d
20
c_l
1
c_m
-0.05
-5
5
α°
10
15
0.5
1
0.05
0.1
these two axes are identical


Re = 10^5
2x10^5
3x10^5
c_l
1
0.5
0
5
10
15
10^3c_d
20
c_l
c_l,max
1
c_m
-0.05
-5
5
α°
10
15
0.5
1
0.05
0.1

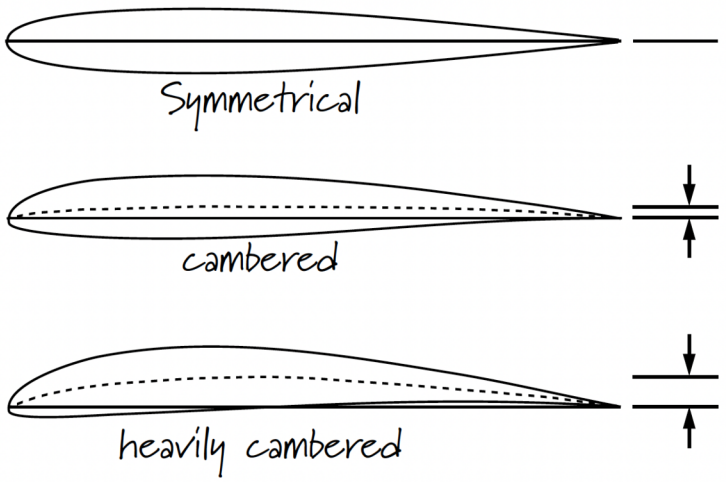

Re = 10^5
2x10^5
3x10^5
c_l
1
0.5
0
5
10
15
10^3c_d
20
c_l
1
c_m
-0.05
0.05
0.1
-5
5
10
15
0.5
1
angle of attack, α, at which c_l,max occurs

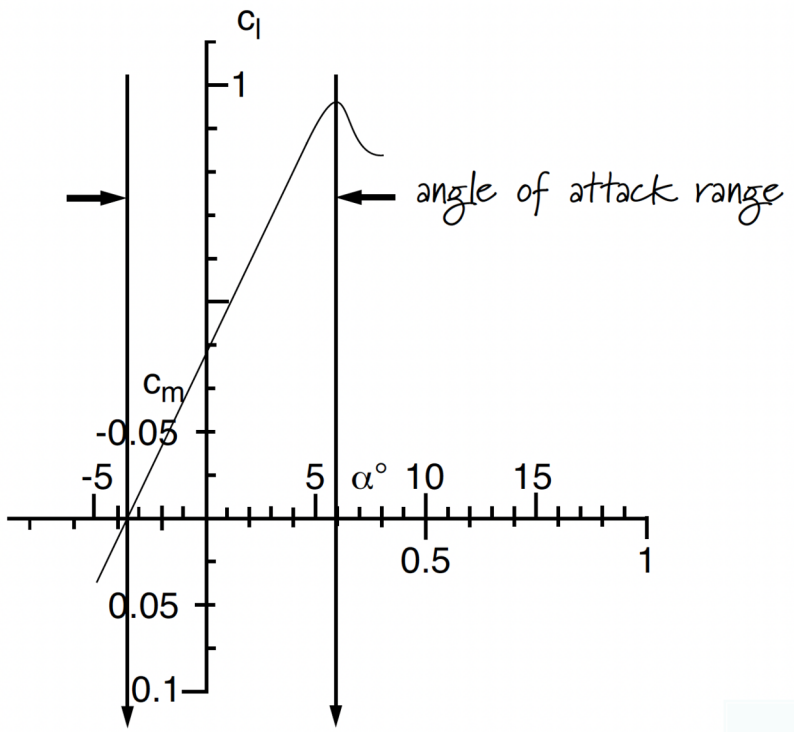

c_l
α
"abrupt" stall
c_l
α
"smooth" stall
c_l
α
"soft" stall

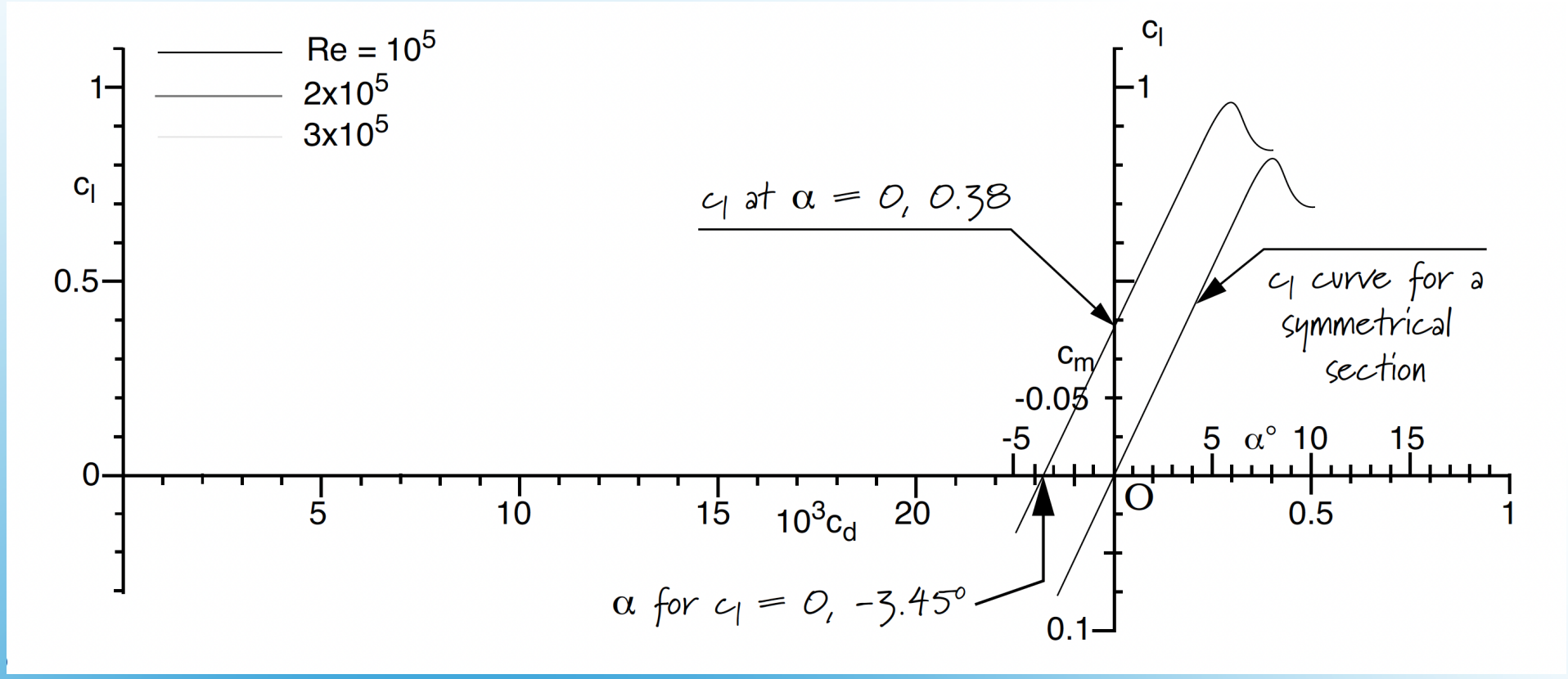




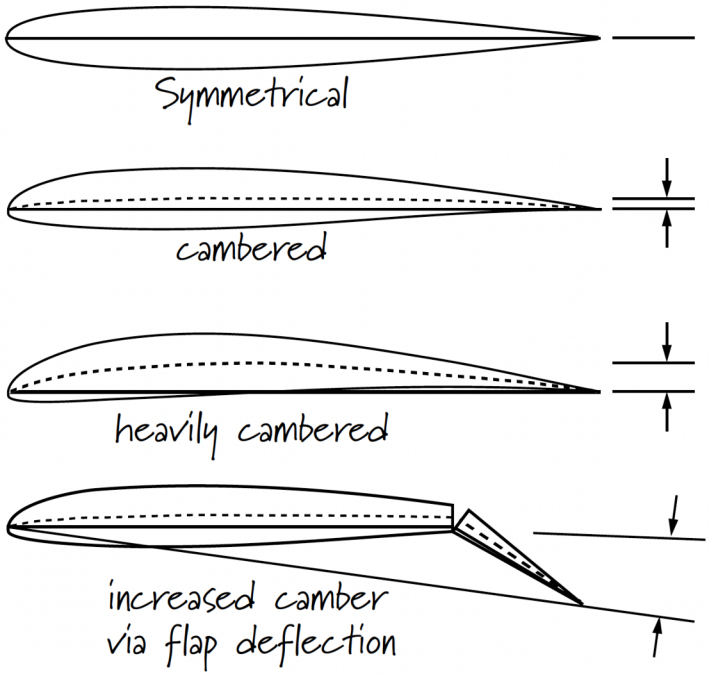
As a rough approximation, the zero lift line passes through the trailing edge and the mean camber line at 50% chord.

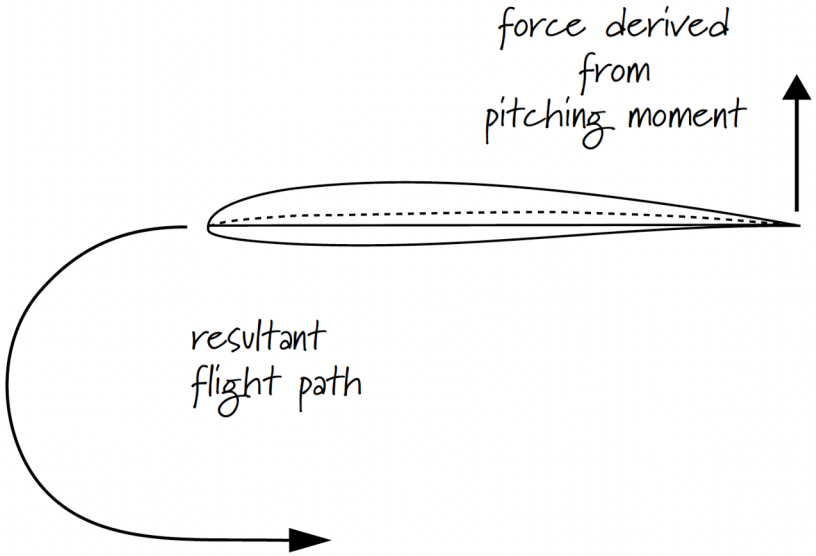

Symmetrical
cambered
heavily cambered

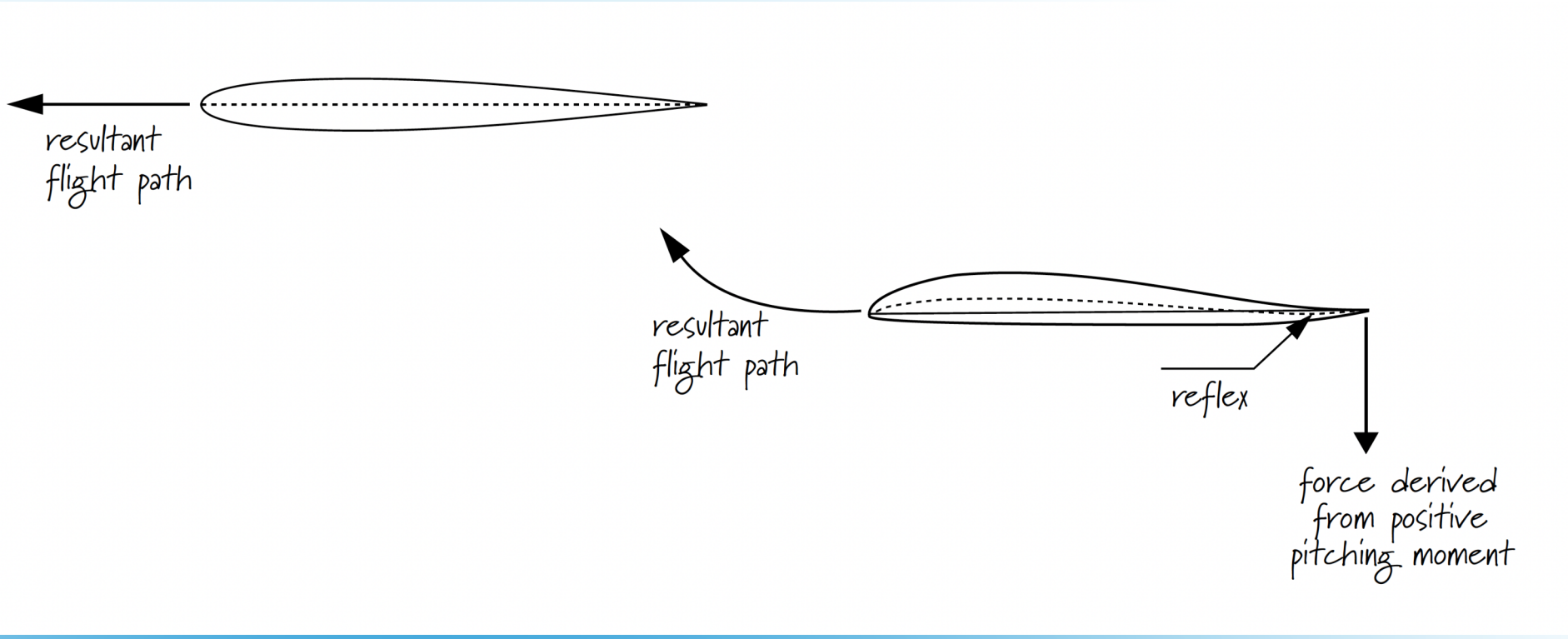

c_l
1
angle of attack range
c_m
-0.05
-5
5
α°
10
15
0.5
1
0.05
0.1

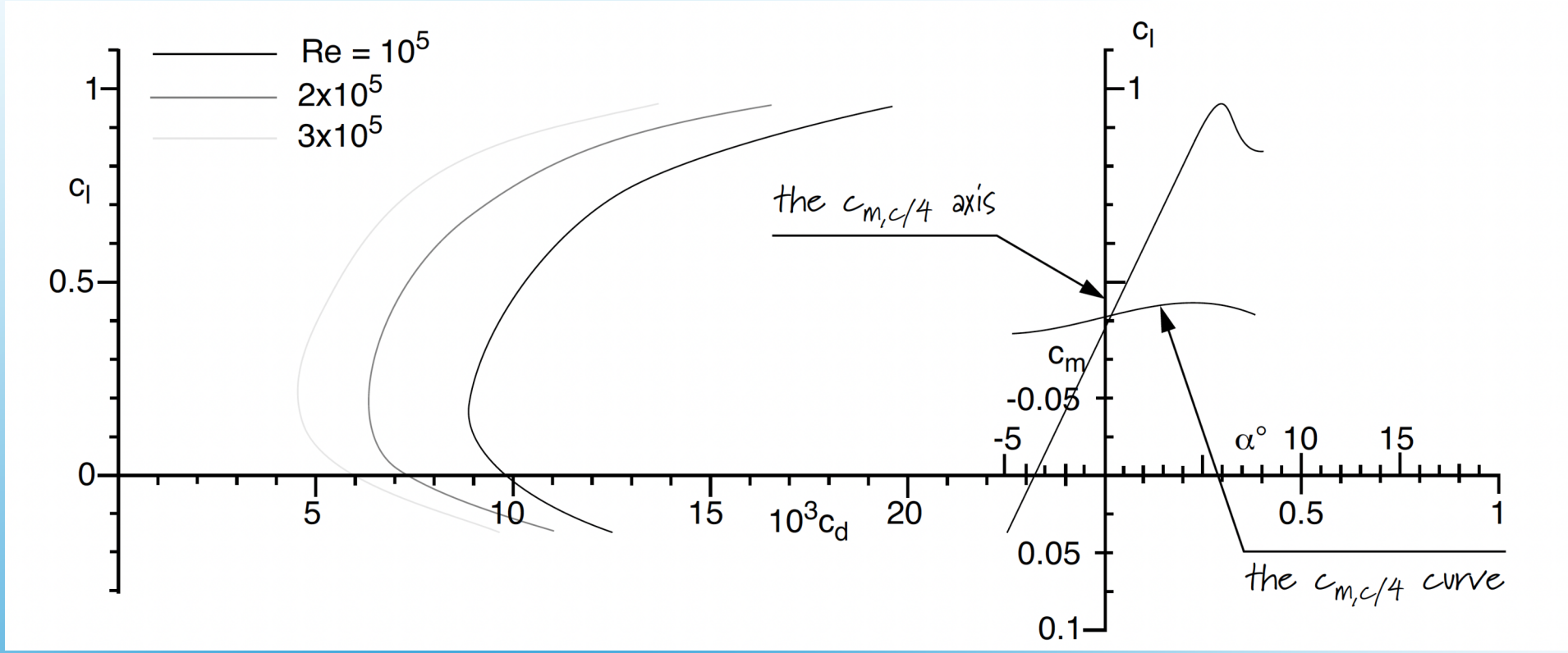

Re = 10^5
2x10^5
3x10^5
c_l
1
0.5
0
5
10
15
10^3c_d
20
c_l at α = 0, 0.38
α for c_l = 0, -3.45°
c_l curve for a symmetrical section
c_m
-0.05
0.1
-5
5
α°
10
15
O
0.5
1

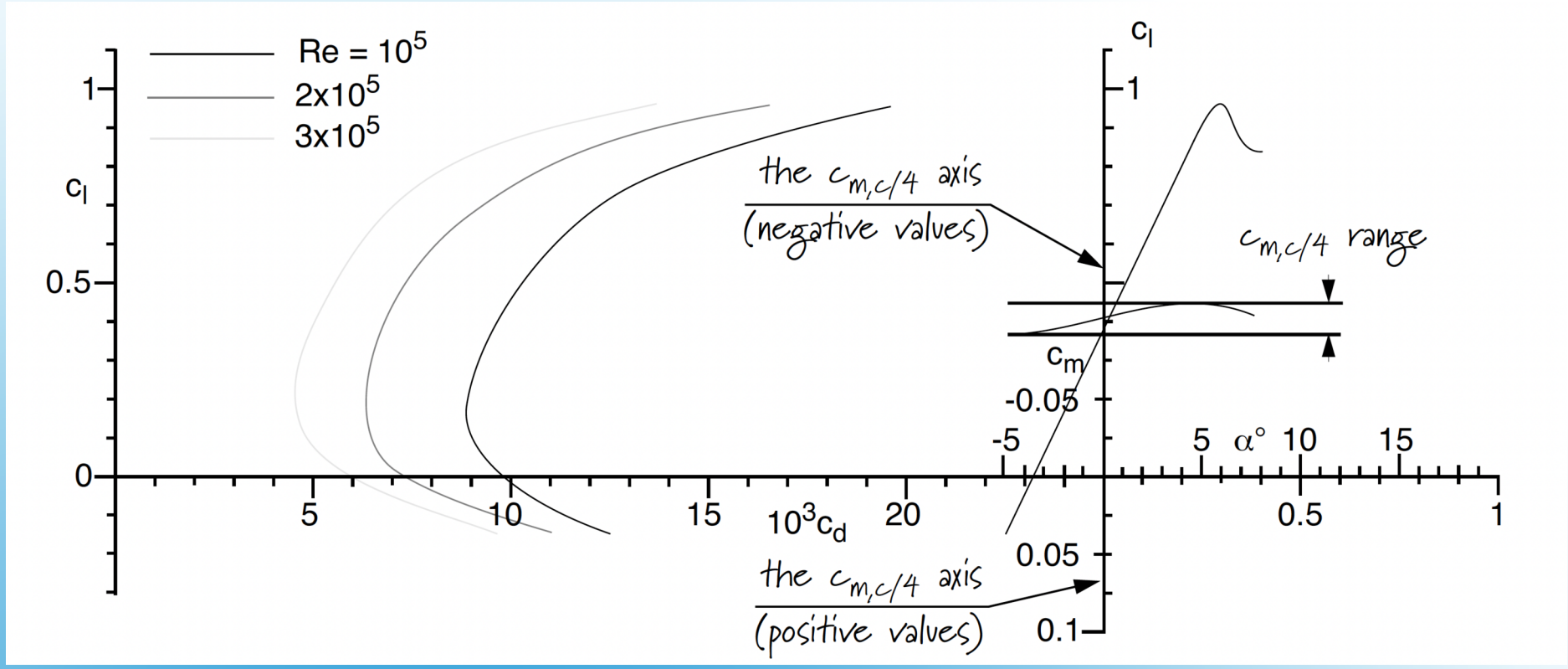
- The Pitching Moment (c_m) Curve

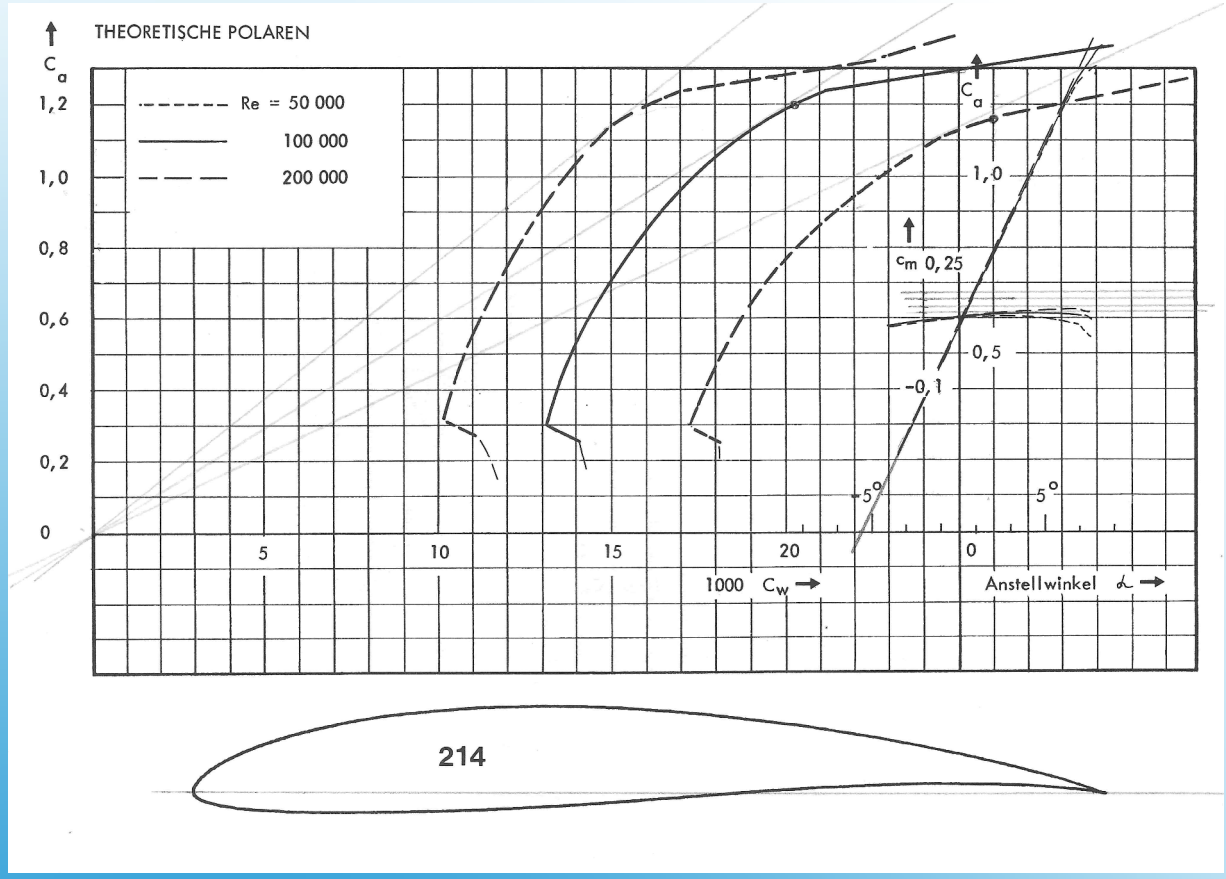

Symmetrical
cambered
heavily cambered
increased camber
via flap deflection

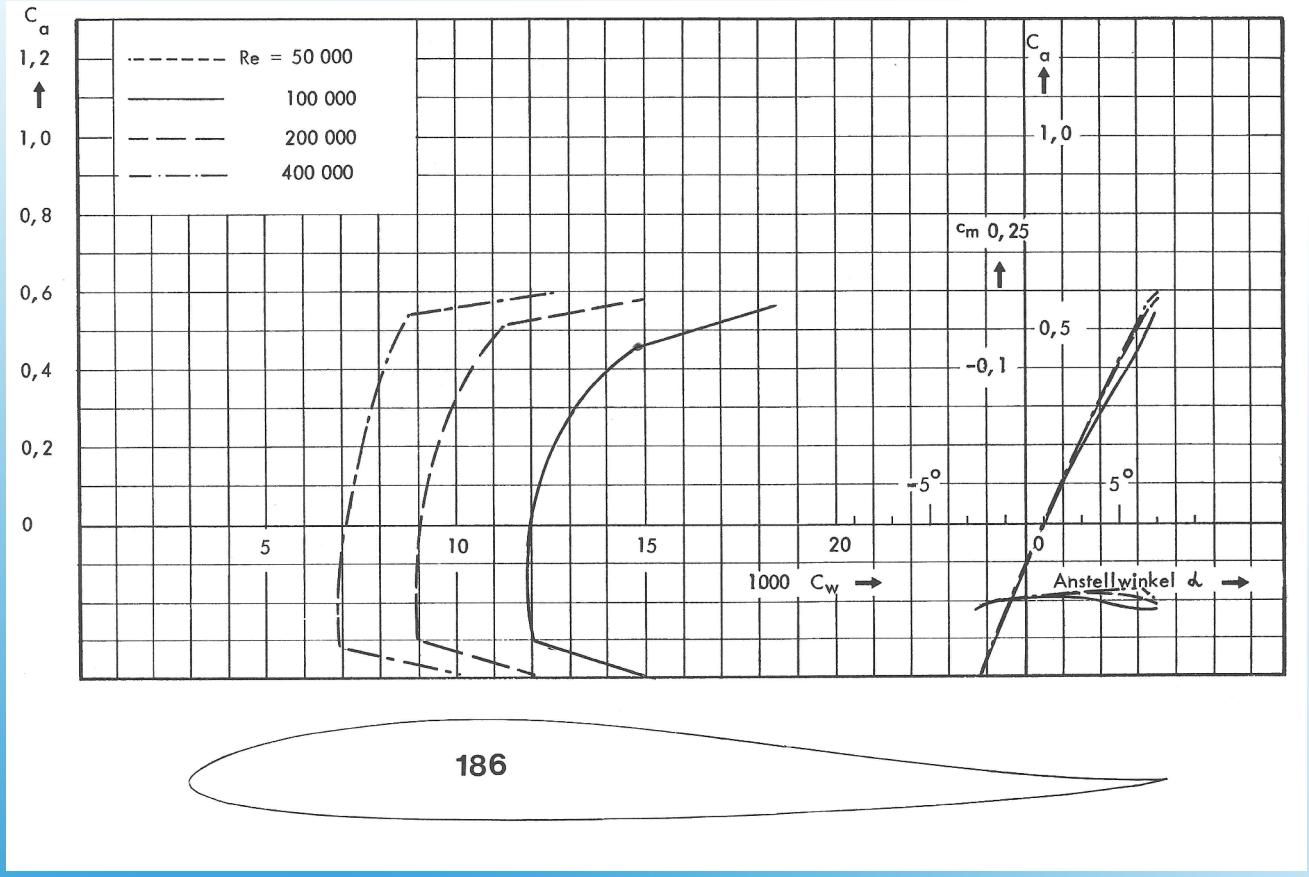

force derived
from
pitching moment
resultant
flight path


resultant
flight path
resultant
flight path
reflex
force derived
from positive
pitching moment

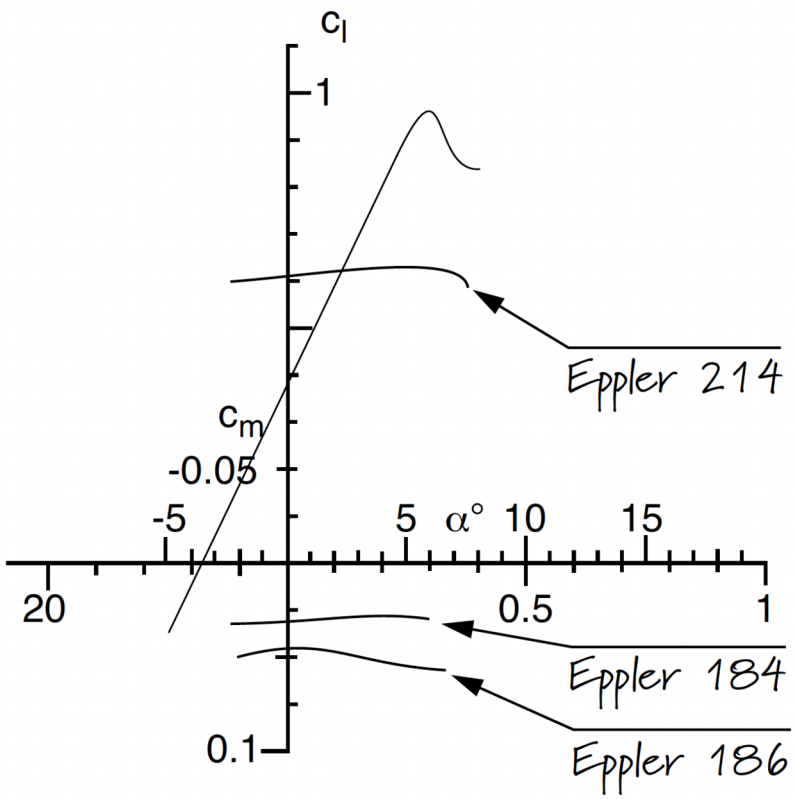

Re = 10^5
2x10^5
3x10^5
c_l
1
0.5
0
5
10
15
10^3c_d
20
c_l
1
the c_m,c/4 axis
c_m
-0.05
-5
α°
10
15
0.5
1
0.05
0.1
the c_m,c/4 curve

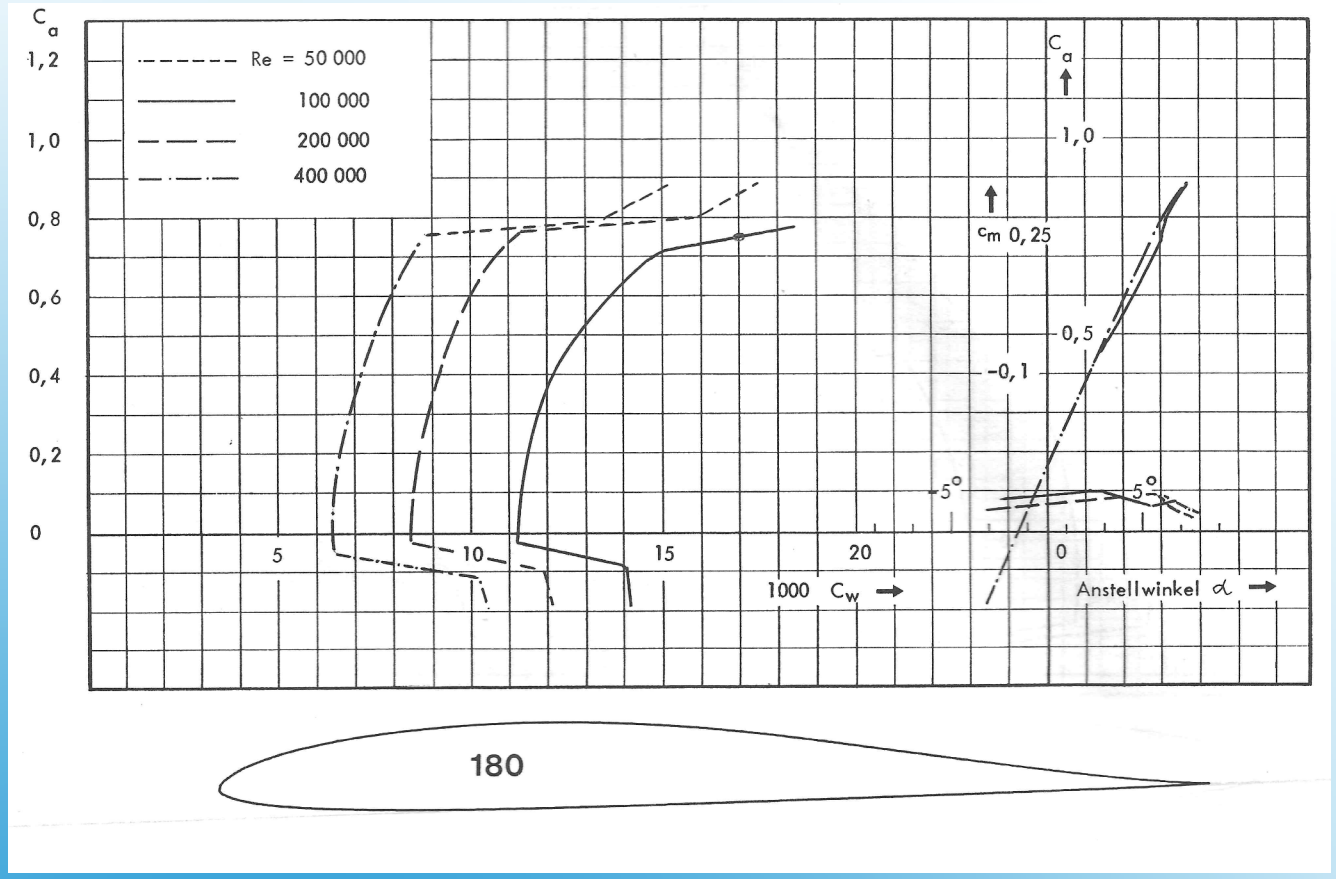

Re = 10^5
2x10^5
3x10^5
c_l
1
0.5
0
5
10
15
10^3c_d
20
the c_m,c/4 axis
(negative values)
the c_m,c/4 axis
(positive values)
c_l
1
c_m,c/4 range
c_m
-0.05
0.05
0.1
-5
5
α°
10
15
0.5
1

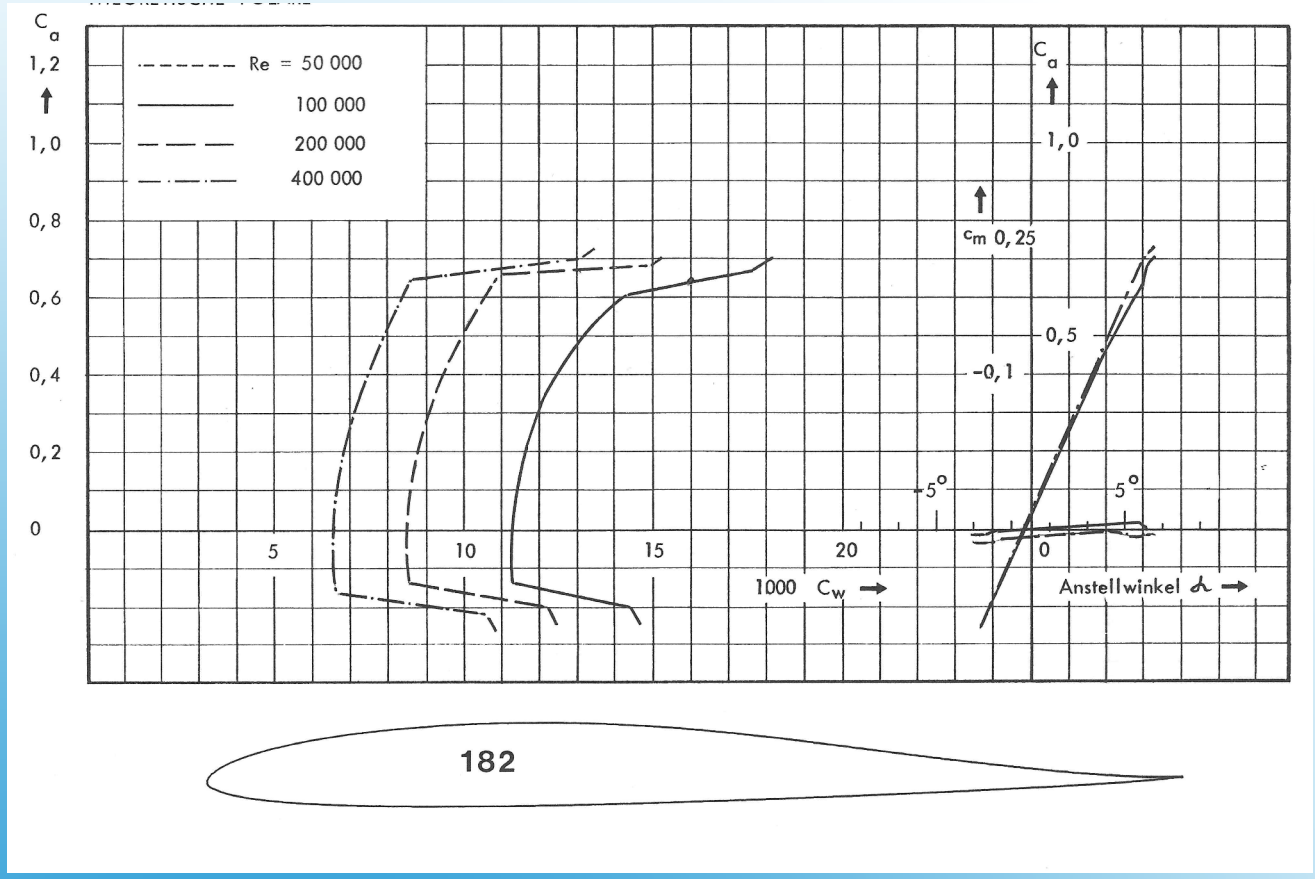

THEORETISCHE POLAREN
C_a
1,2
1,0
0,8
0,6
0,4
0,2
0
Re = 50 000
100 000
200 000
5
10
15
20
1000 C_w
C_a
1,0
c_m 0,25
0,5
-0,1
-5°
5°
0
Anstellwinkel α
214

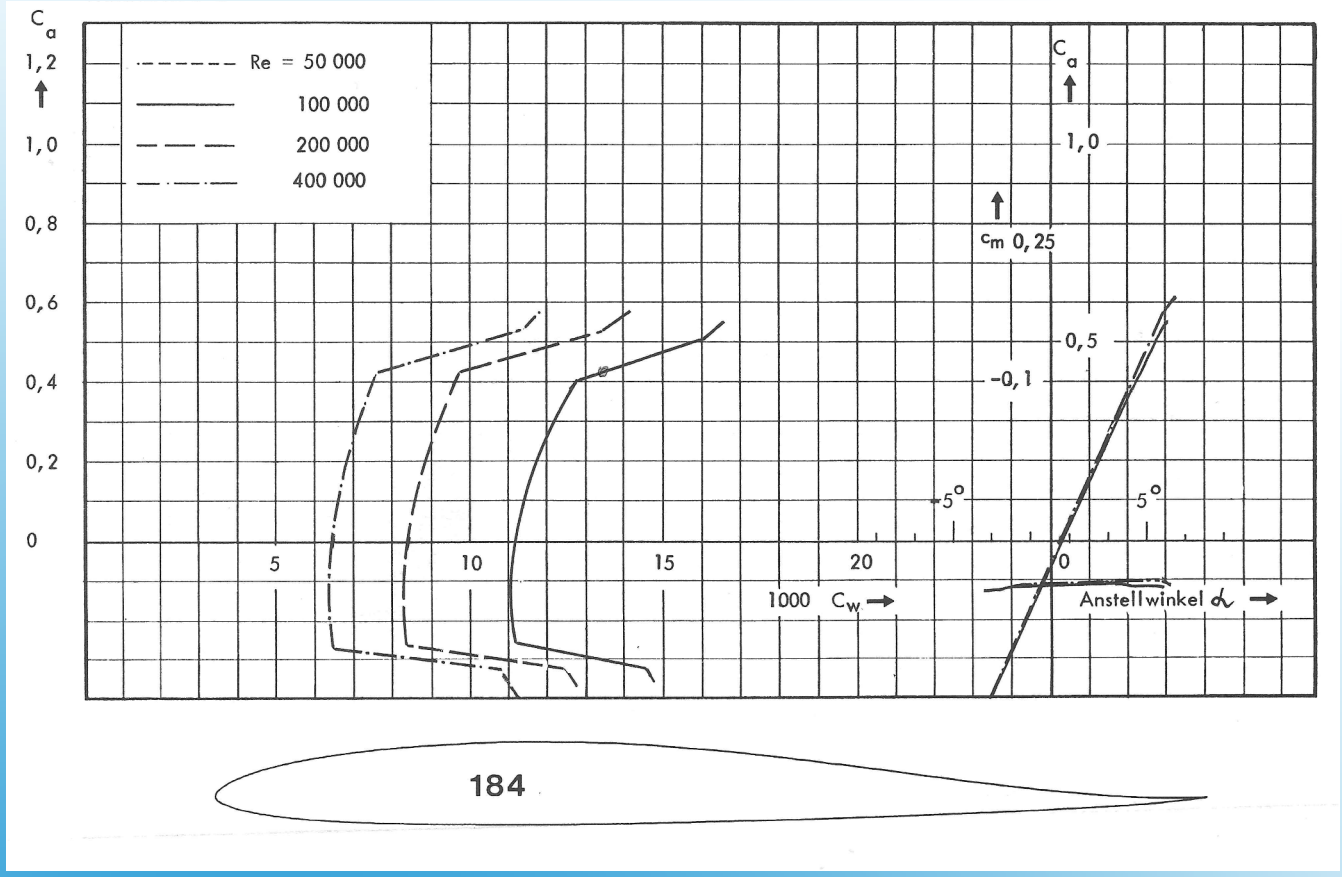

C_a
1,2
1,0
0,8
0,6
0,4
0,2
0
Re = 50 000
100 000
200 000
400 000
5
10
15
20
1000 C_w
C_a
1,0
c_m 0,25
0,5
-0,1
-5°
5°
0
Anstellwinkel α
186

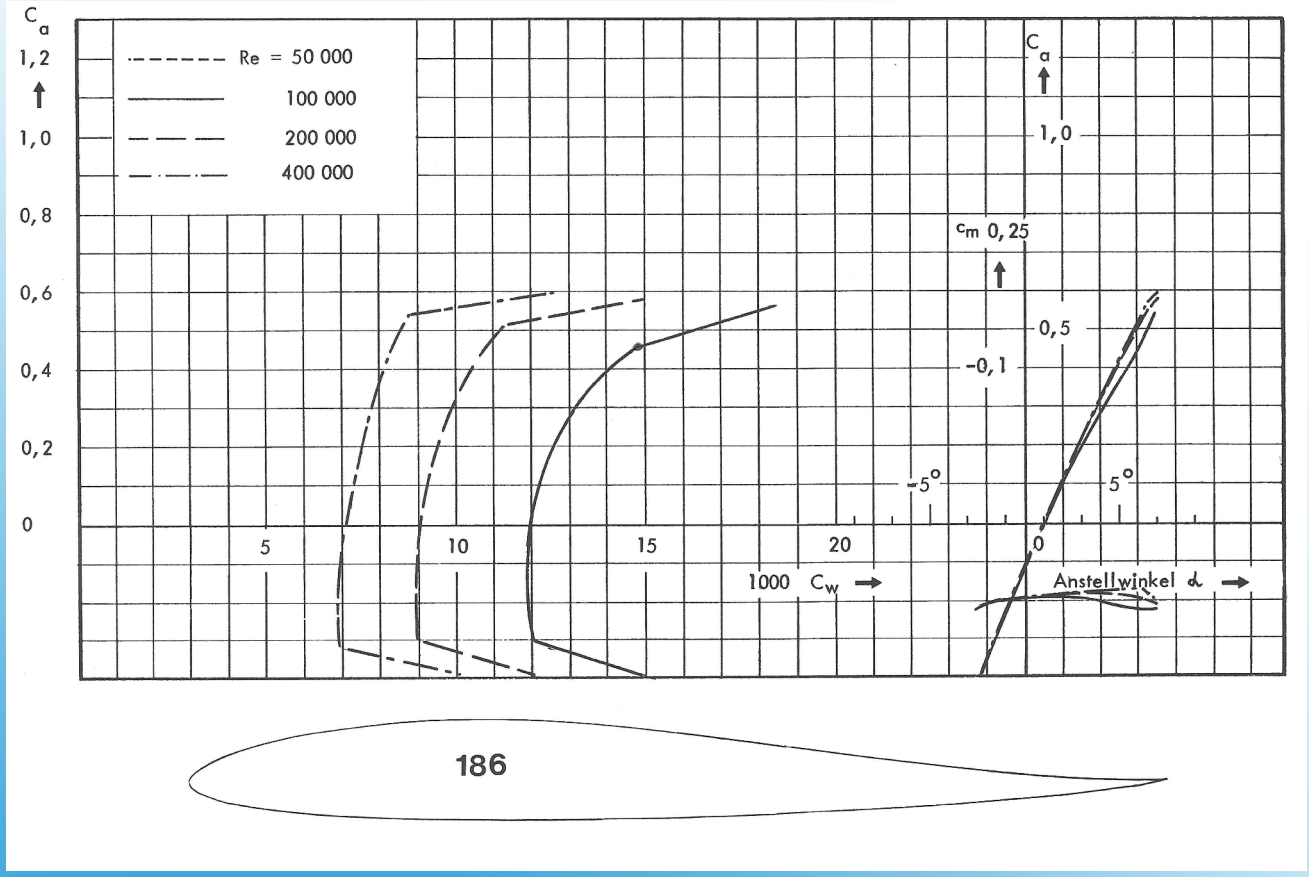
Remember, as the moment coefficient becomes more negative the $c_{m,c/4}$ curve will move higher on the polar.

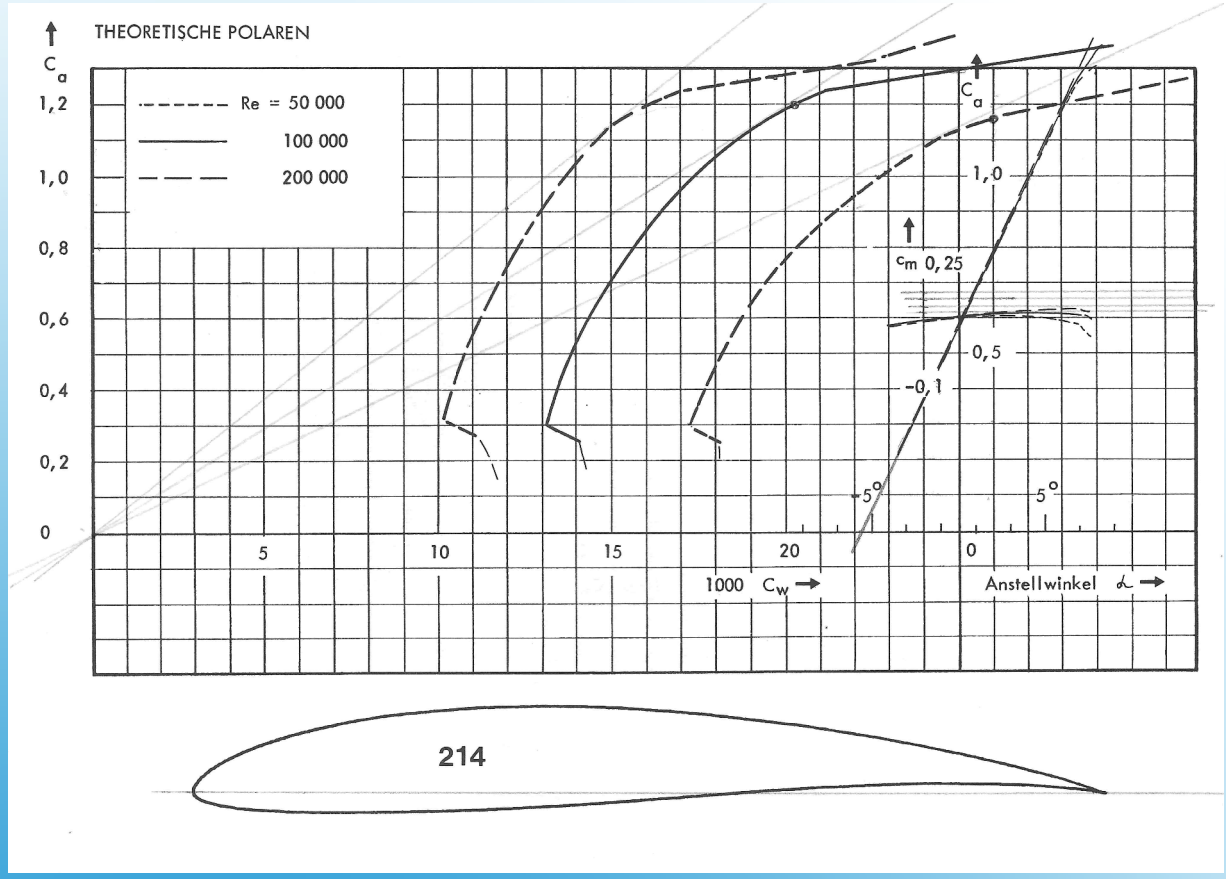


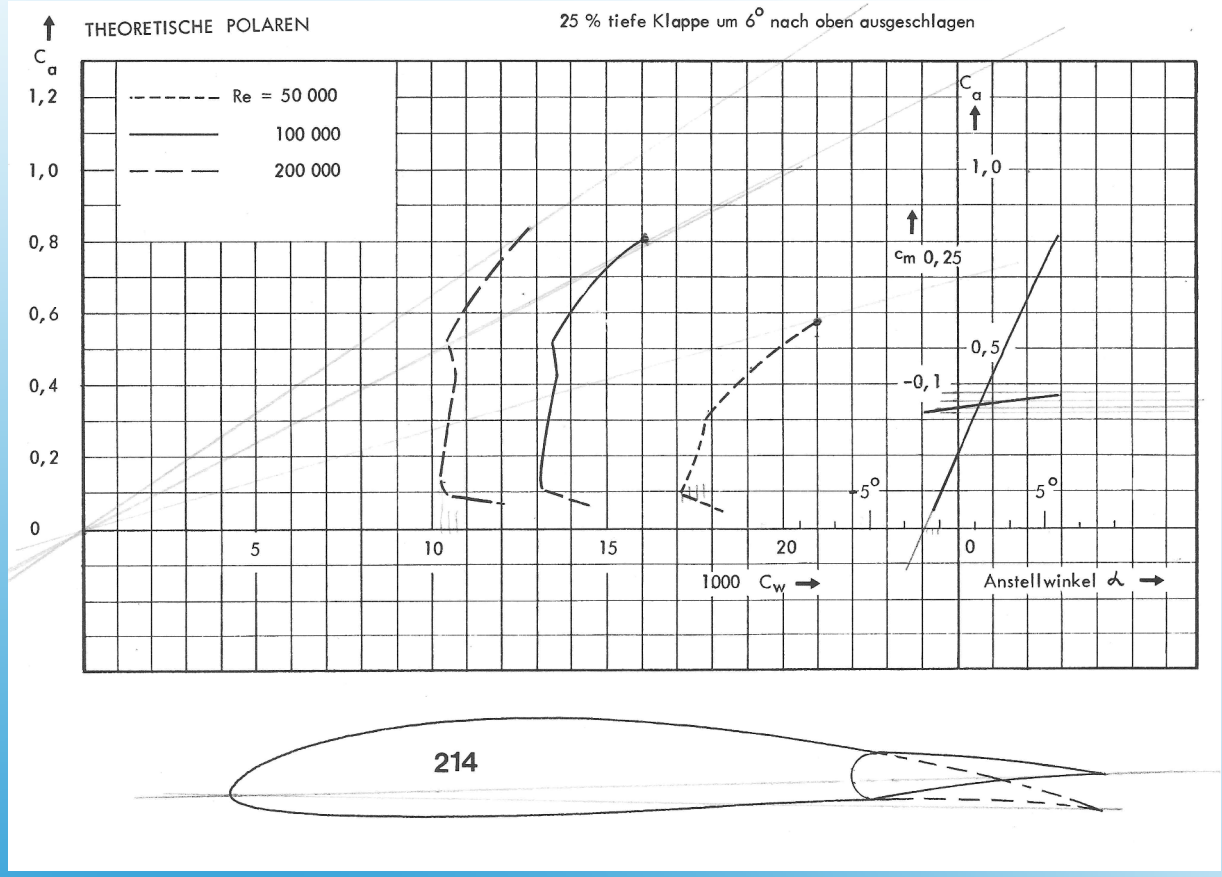

C_a
1,2
1,0
0,8
0,6
0,4
0,2
0
Re = 50 000
100 000
200 000
400 000
5
10
15
20
1000 C_w
C_a
1,0
c_m 0,25
0,5
-0,1
-5°
5°
0
Anstellwinkel α
180

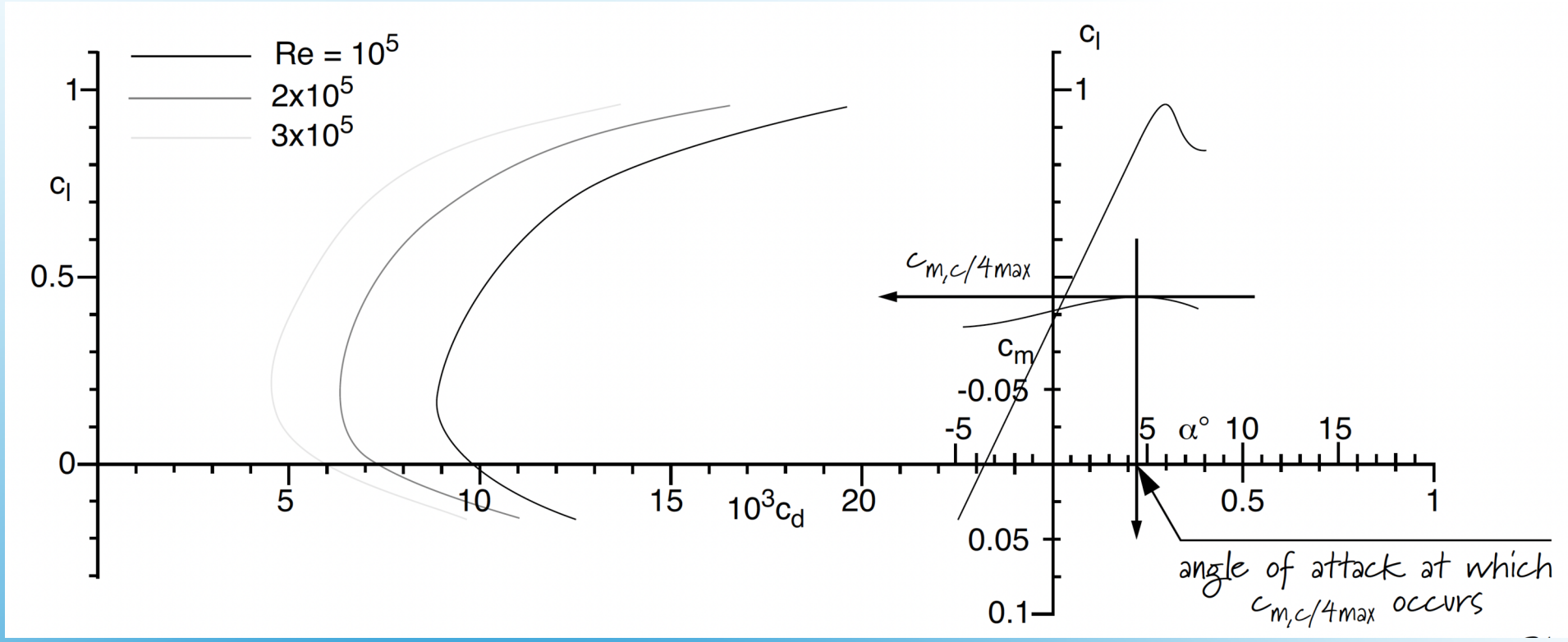

C_a
1,2
1,0
0,8
0,6
0,4
0,2
0
Re = 50 000
100 000
200 000
400 000
5
10
15
20
1000 C_w →
C_a
1,0
c_m 0,25
0,5
-0,1
-5°
5°
0
Anstellwinkel α →
182


C_a
1,2
1,0
0,8
0,6
0,4
0,2
0
Re = 50 000
100 000
200 000
400 000
5
10
15
20
1000 C_w →
C_a
1,0
c_m 0,25
0,5
-0,1
-5°
5°
0
Anstellwinkel α →
184


C_a
1,2
1,0
0,8
0,6
0,4
0,2
0
Re = 50 000
100 000
200 000
400 000
5
10
15
20
1000 C_w
C_a
1,0
c_m 0,25
0,5
-0,1
-5°
5°
0
Anstellwinkel α
186

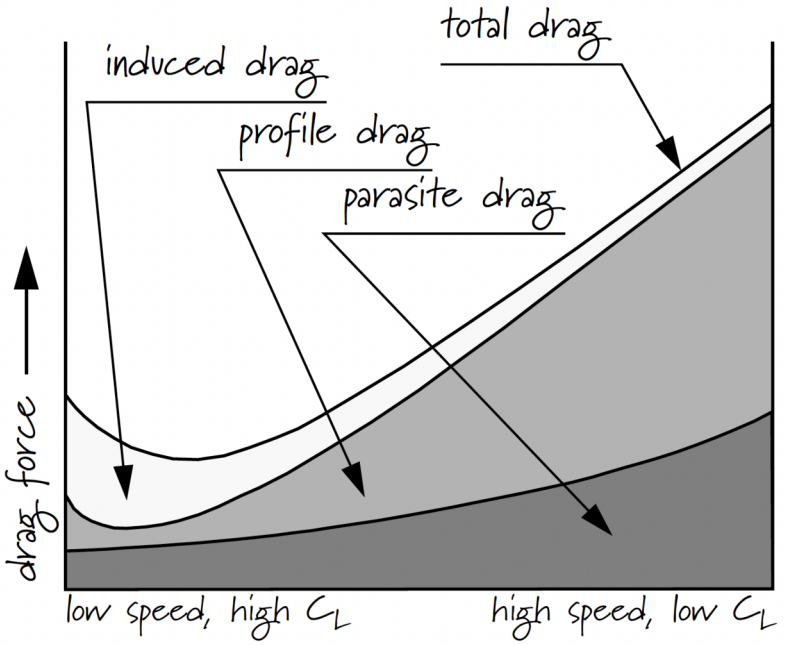

THEORETISCHE POLAREN
C_a
1,2
1,0
0,8
0,6
0,4
0,2
0
Re = 50 000
100 000
200 000
5
10
15
20
1000 C_w
C_a
1,0
c_m 0,25
0,5
-0,1
-5°
5°
0
Anstellwinkel α
214

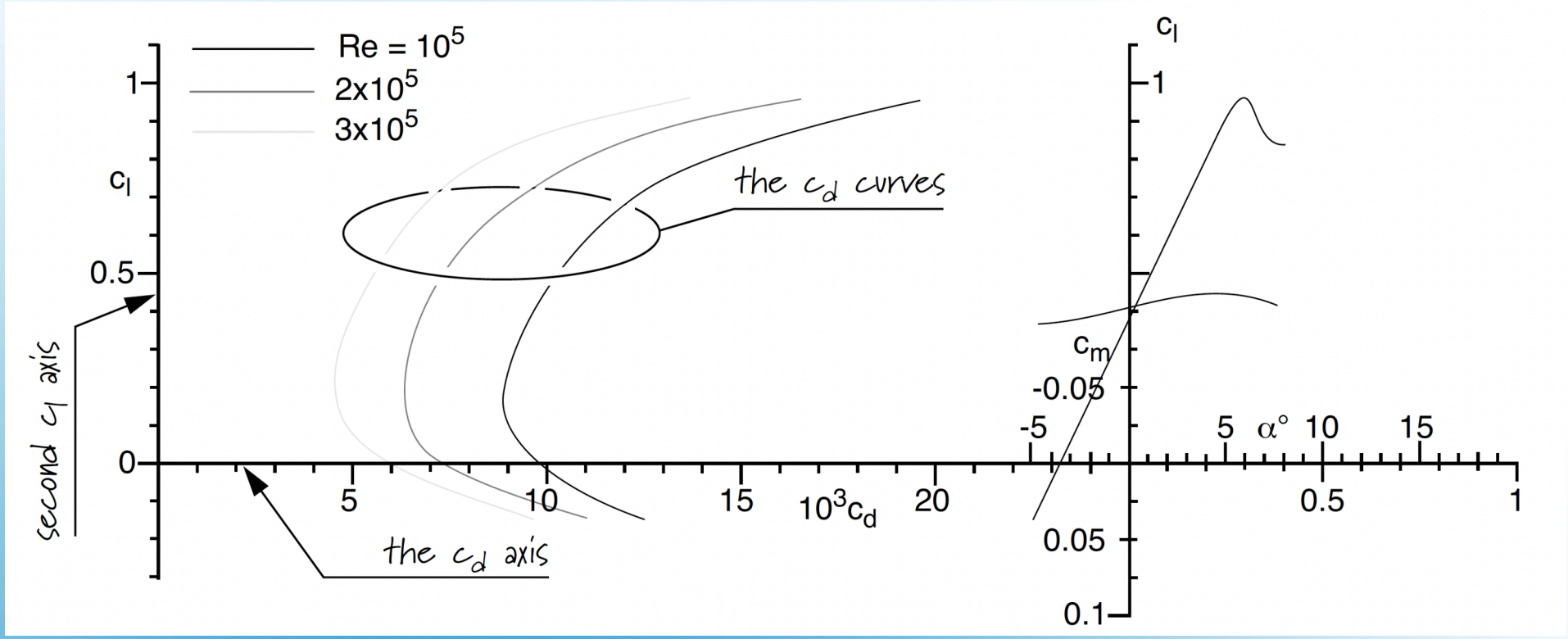

THEORETISCHE POLAREN
25 % tiefe Klappe um 6° nach oben ausgeschlagen
Ca
1,2
1,0
0,8
0,6
0,4
0,2
0
Re = 50 000
100 000
200 000
5
10
15
20
1000 Cw
Ca
1,0
cm 0,25
0,5
-0,1
-5°
5°
0
Anstellwinkel α
214

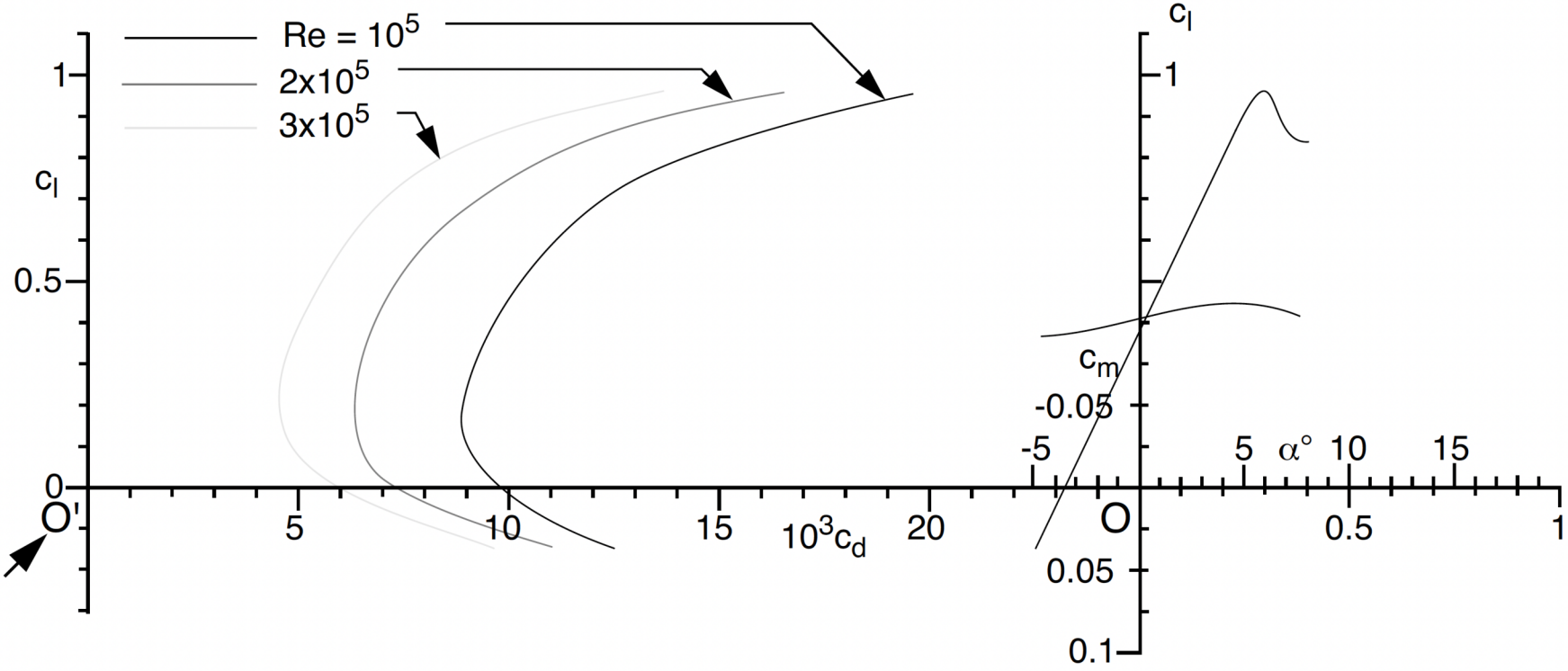

Re = 10^5
2x10^5
3x10^5
c_l
1
0.5
0
5
10
15
10^3c_d
20
c_l
1
c_m,c/4max
c_m
-0.05
-5
5
α°
10
15
0.5
1
0.05
0.1
angle of attack at which
c_m,c/4max occurs

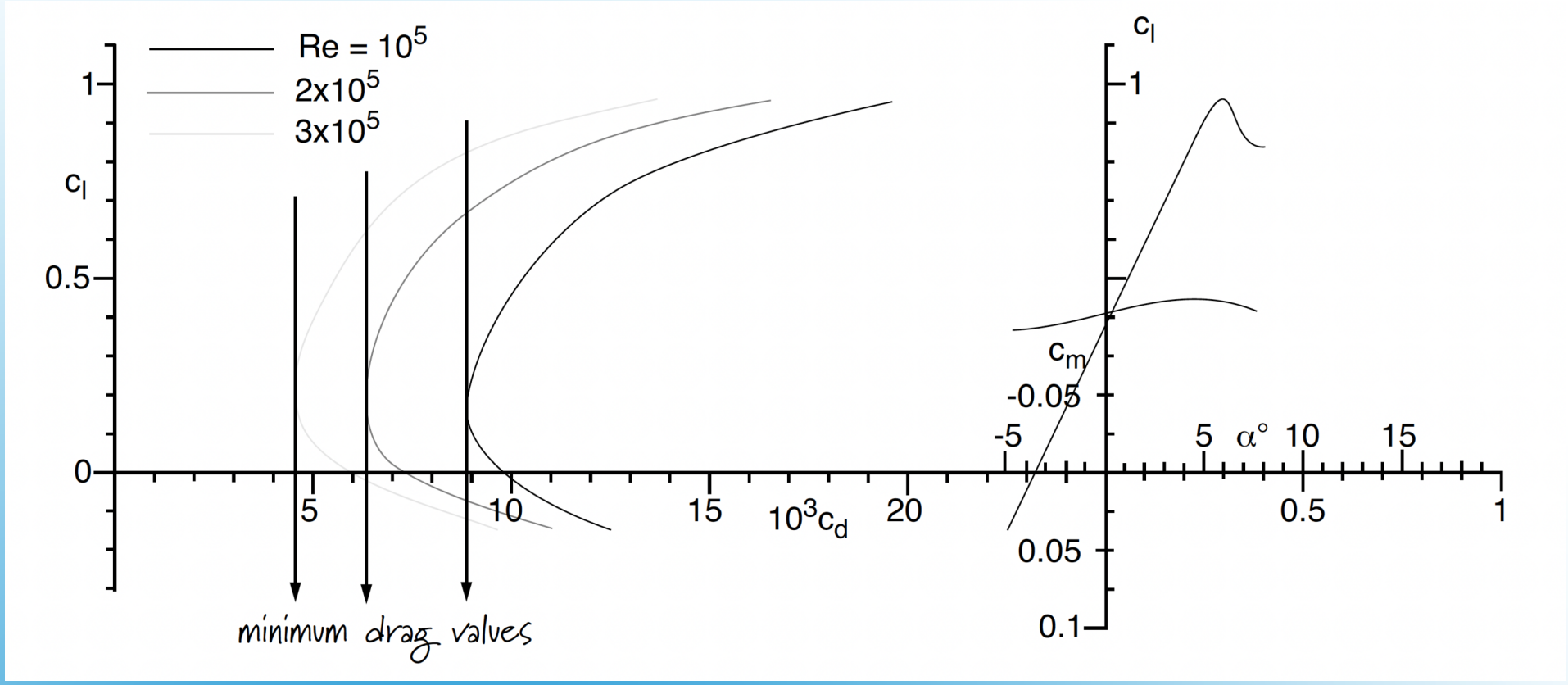
- The Drag Coefficient (c_d) Curve

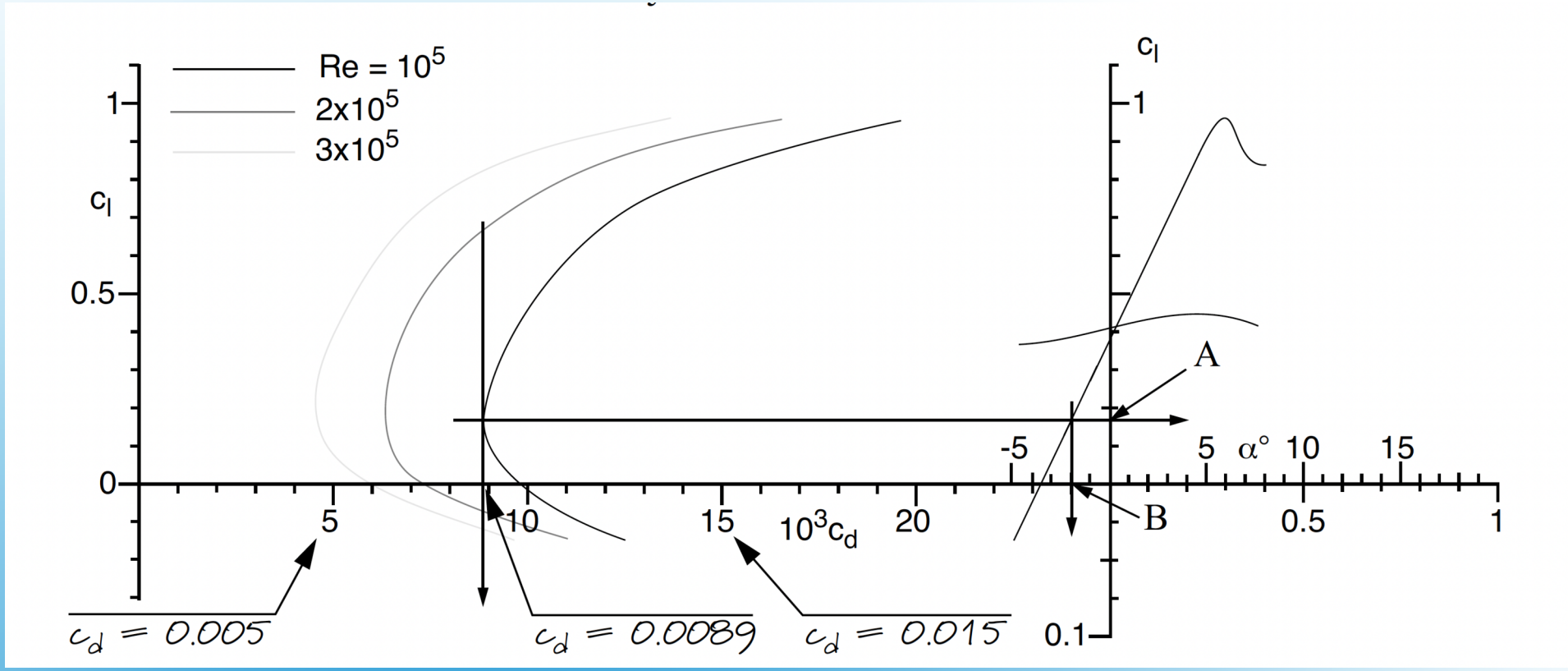
- The c_d as portrayed on a polar diagram normally consists of skin friction drag and pressure drag only; that is, it does not include induced drag.
- Skin friction drag is created by the viscosity ("stickiness") of the air and by surface roughness.
- Pressure drag is created by the object pushing the air away from its free stream flow.
- Induced drag, or more exactly vortex induced drag, is created by the rotating airflow that occurs around the tips of any lifting surface such as a wing or stabilizer.

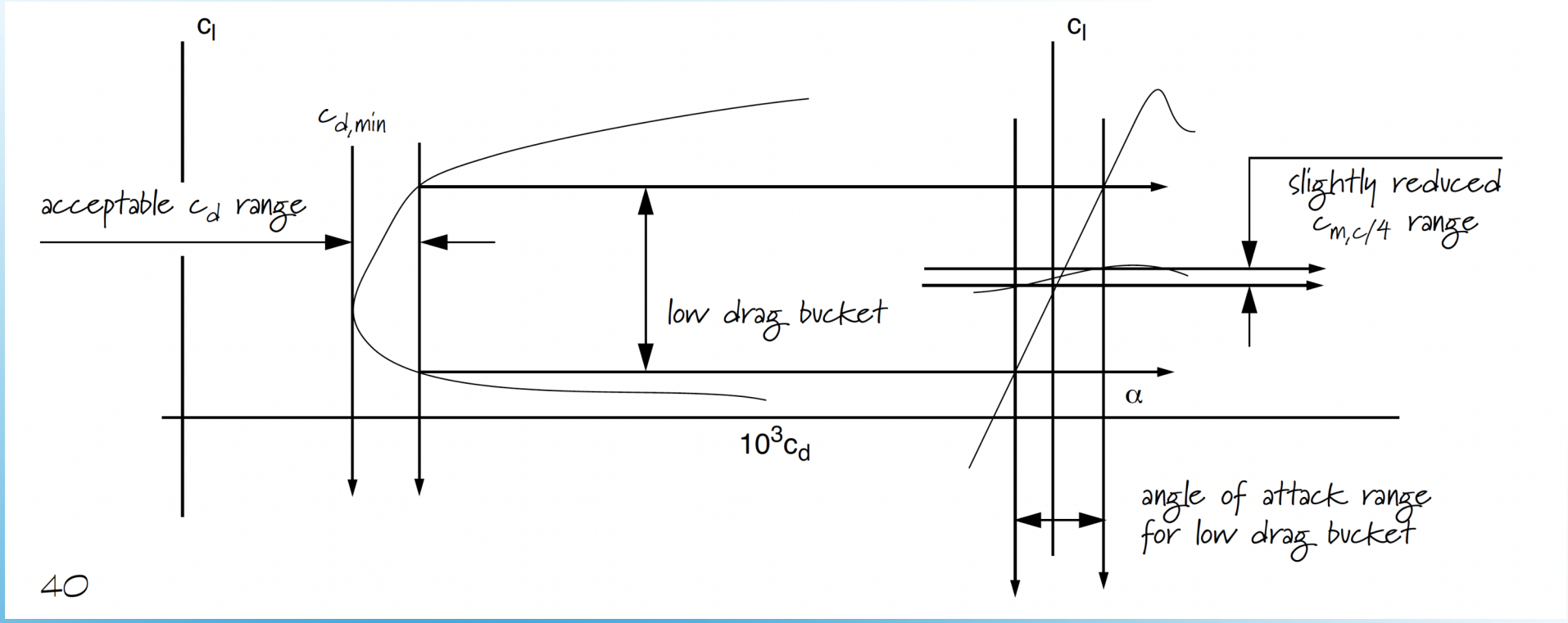


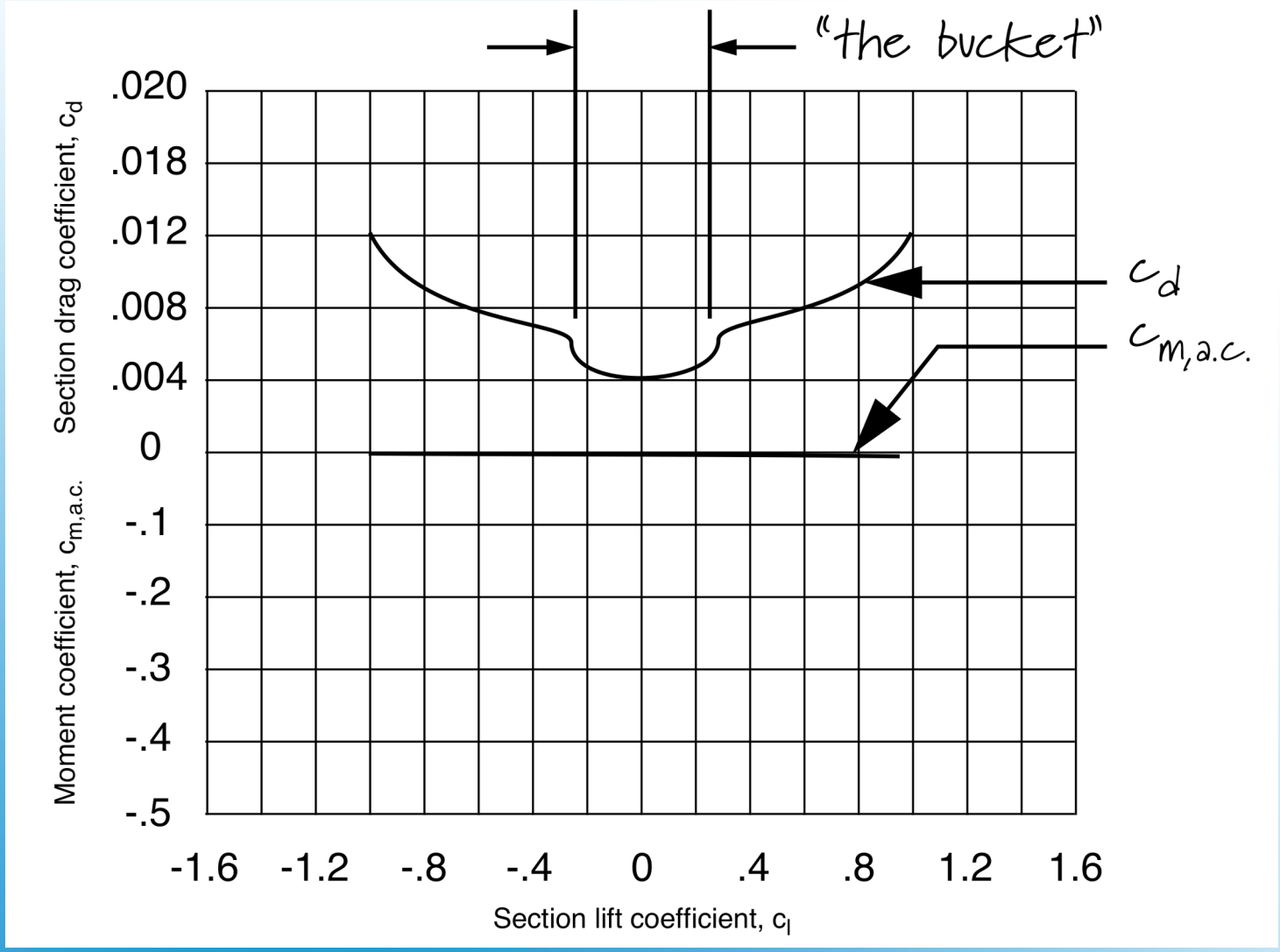

Re = 10^5
2x10^5
3x10^5
c_l
1
0.5
0
second c_l axis
the c_d curves
the c_d axis
5
10
15
10^3c_d
20
c_l
1
c_m
-0.05
0.05
0.1
-5
5
α°
10
15
0.5
1

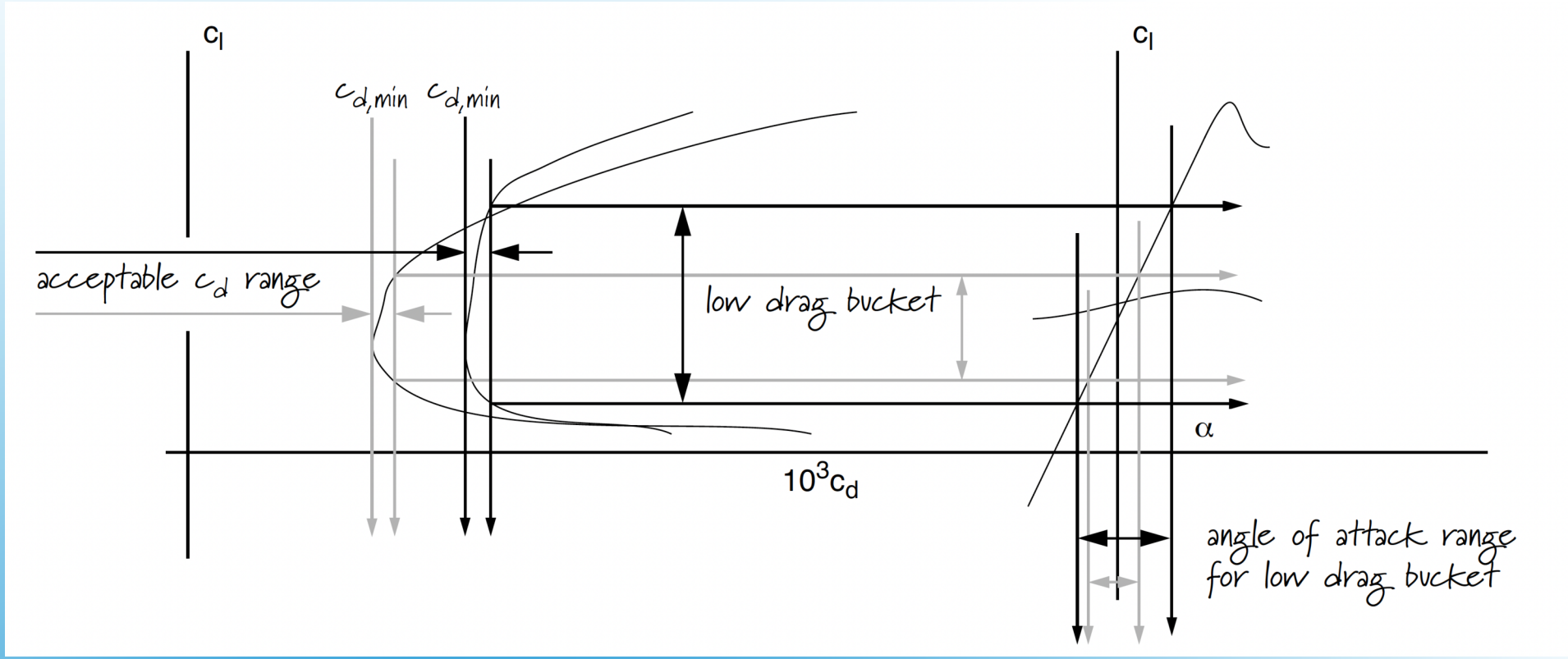

Re = 10^5
2x10^5
3x10^5
c_l
1
0.5
0
O'
5
10
15
10^3c_d
20
c_l
1
c_m
-0.05
-5
5
α°
10
15
O
0.5
1
0.05
0.1

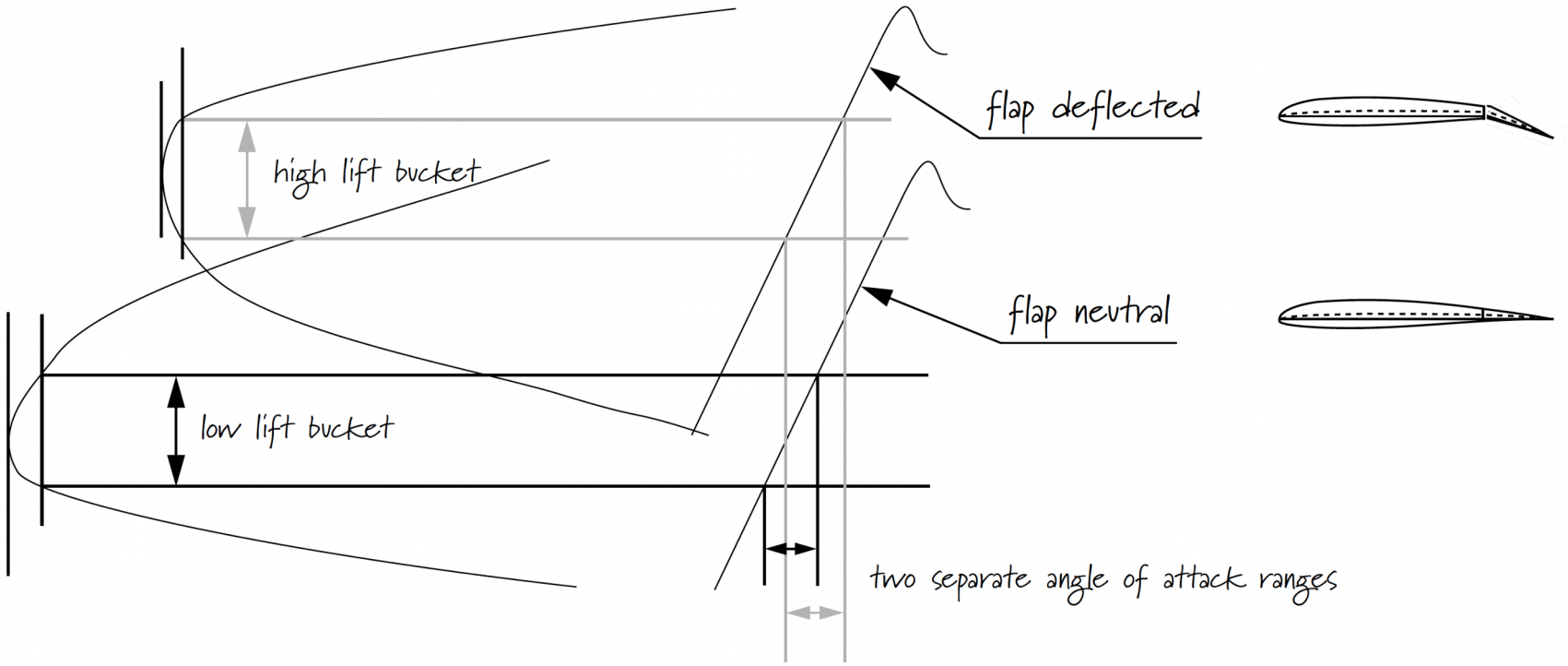

Re = 10^5
2x10^5
3x10^5
c_l
1
0.5
0
5
10
15
10^3c_d
20
minimum drag values
c_l
1
c_m
-0.05
-5
5
α°
10
15
0.5
1
0.05
0.1


Re = 10^5
2x10^5
3x10^5
c_l
1
0.5
0
5
10
15
10^3c_d
20
c_l
1
A
B
-5
5
α°
10
15
0.5
1
0.1
c_d = 0.005
c_d = 0.0089
c_d = 0.015

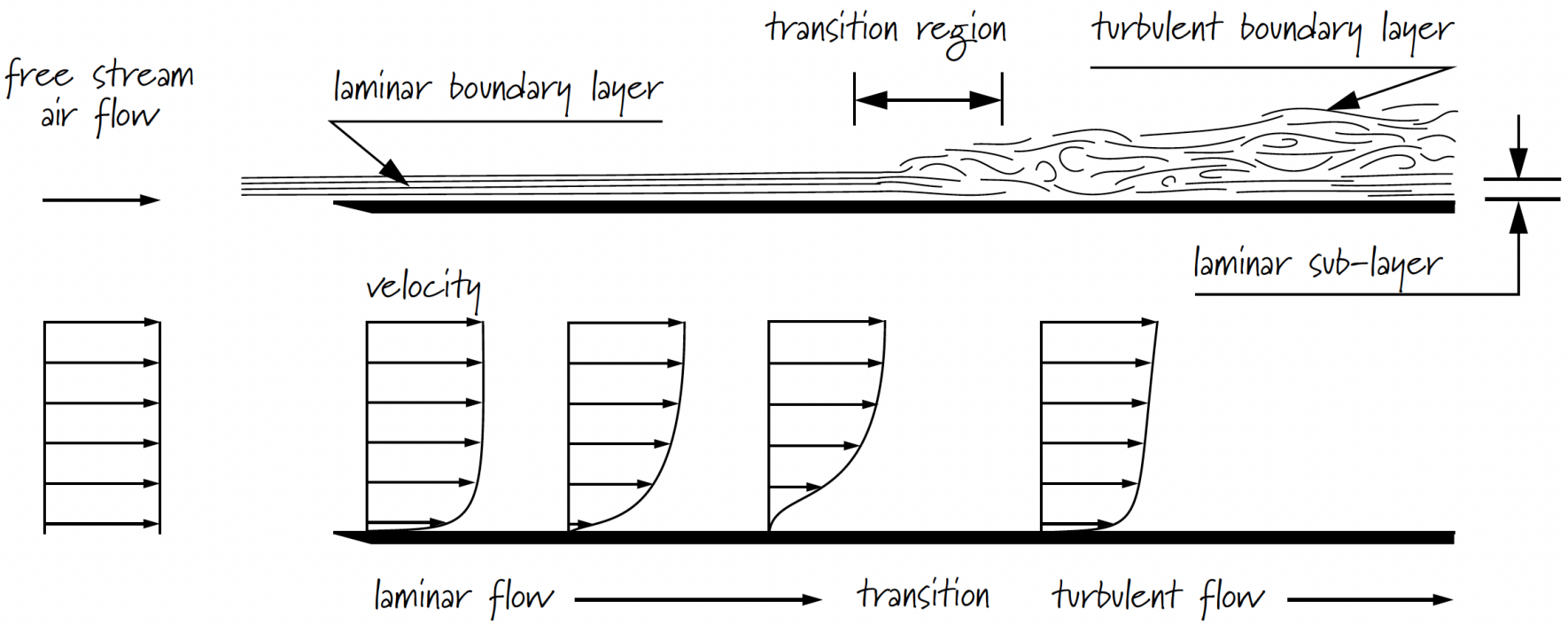

c_l
$c_{d,min}$
acceptable c_d range
low drag bucket
$10^3 c_d$
c_l
slightly reduced $c_{m,c/4}$ range
α
angle of attack range for low drag bucket
40


"the bucket"
.020
.018
.012
.008
.004
0
-.1
-.2
-.3
-.4
-.5
Section drag coefficient, c_d
Moment coefficient, $c_{m,a.c.}$
c_d
$c_{m,a.c.}$
-1.6
-1.2
-.8
-.4
0
.4
.8
1.2
1.6
Section lift coefficient, c_l


c_l
$c_{d,min}$ $c_{d,min}$
acceptable c_d range
low drag bucket
$10^3 c_d$
c_l
α
angle of attack range
for low drag bucket


high lift bucket
low lift bucket
flap deflected
flap neutral
two separate angle of attack ranges

- **Other Curves and Symbols**

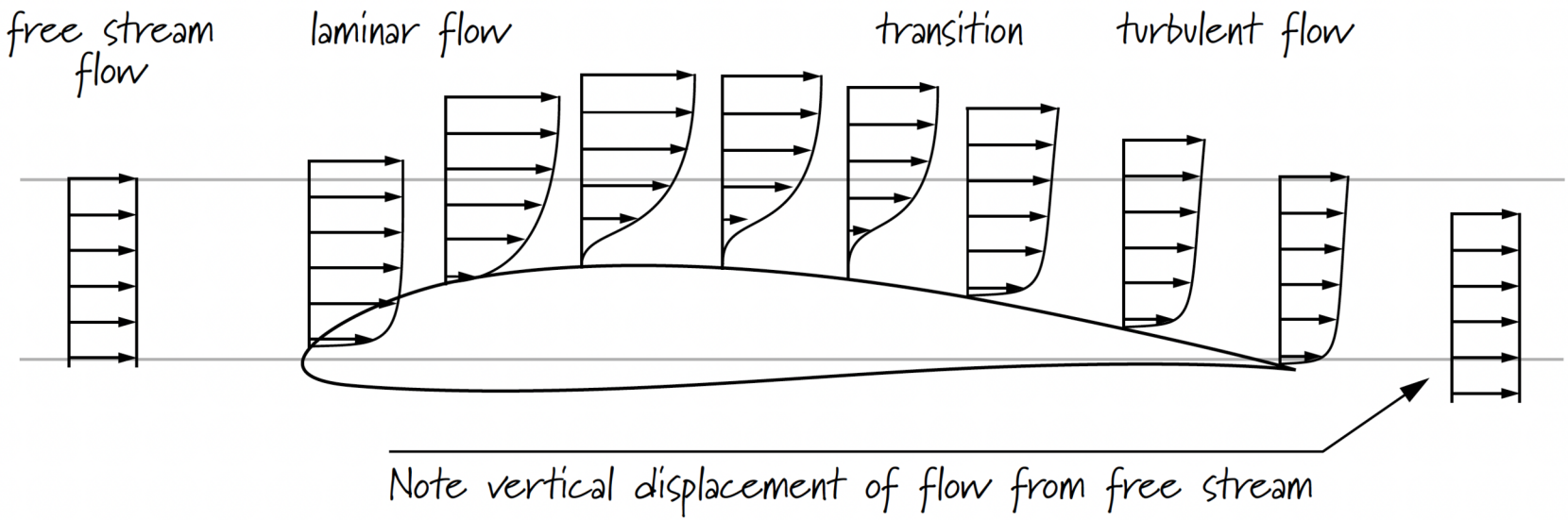

free stream
air flow
laminar boundary layer
transition region
turbulent boundary layer
laminar sub-layer
velocity
laminar flow
transition
turbulent flow

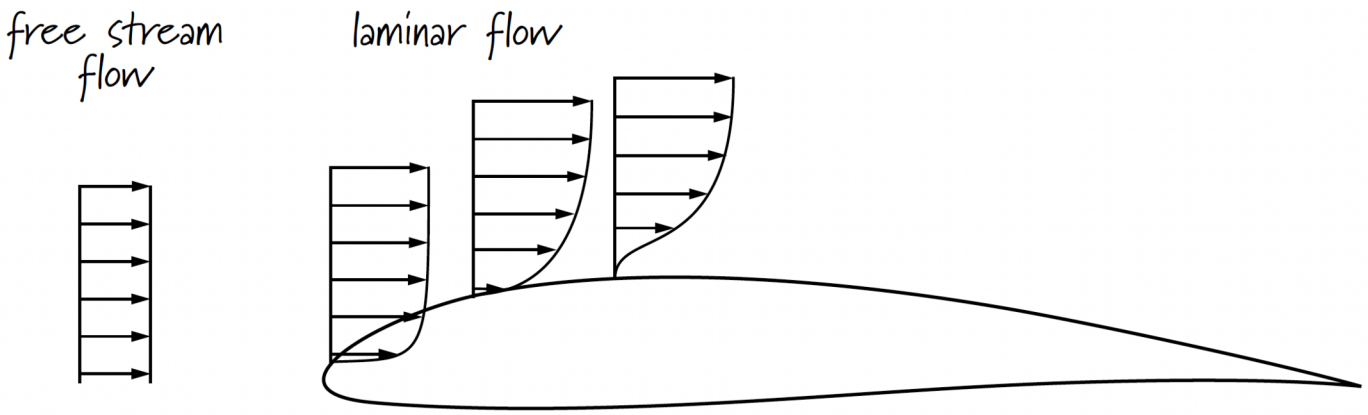

817
NASA
N817NA
ARMSTRONG FLIGHT RESEARCH CENTER
EARTH SCIENCE

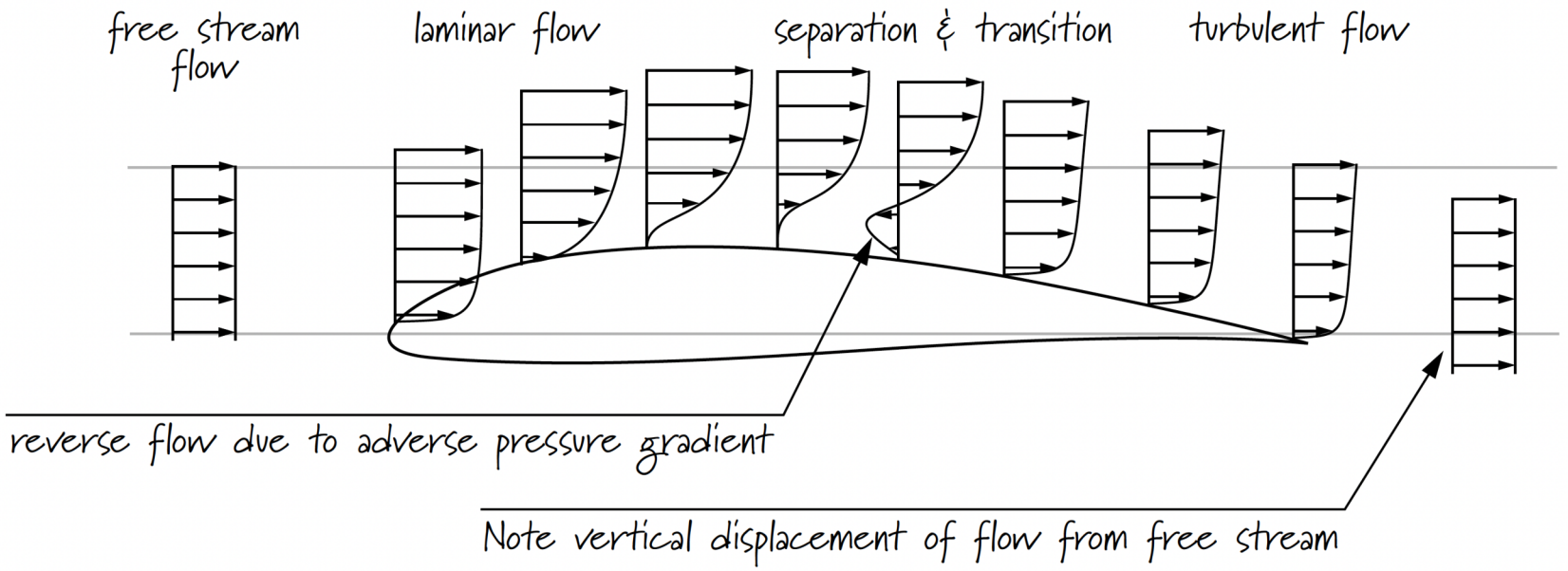

ARMSTRONG FLIGHT RESEARCH CENTER
EARTH SCIENCE

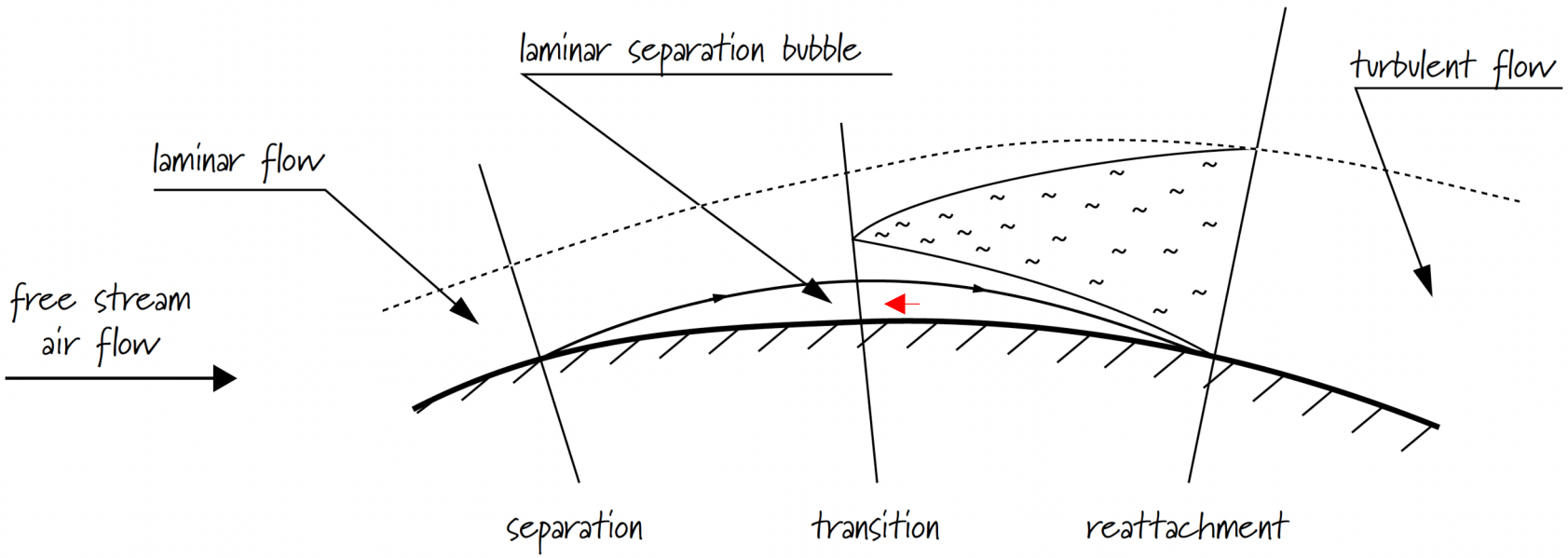

ARM

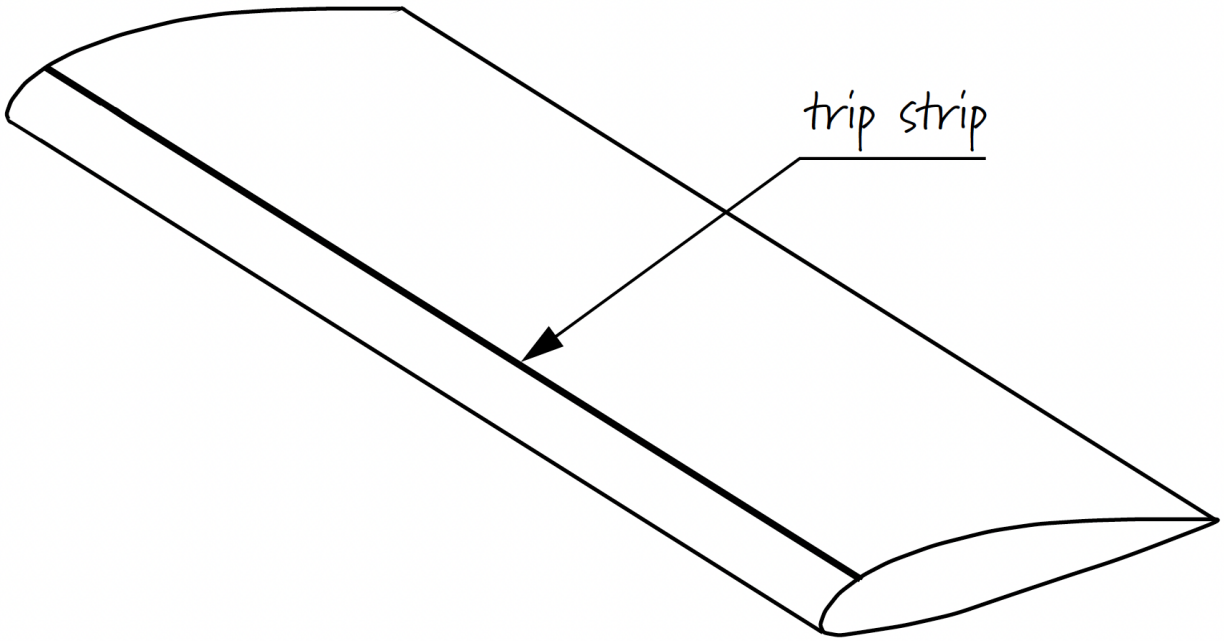

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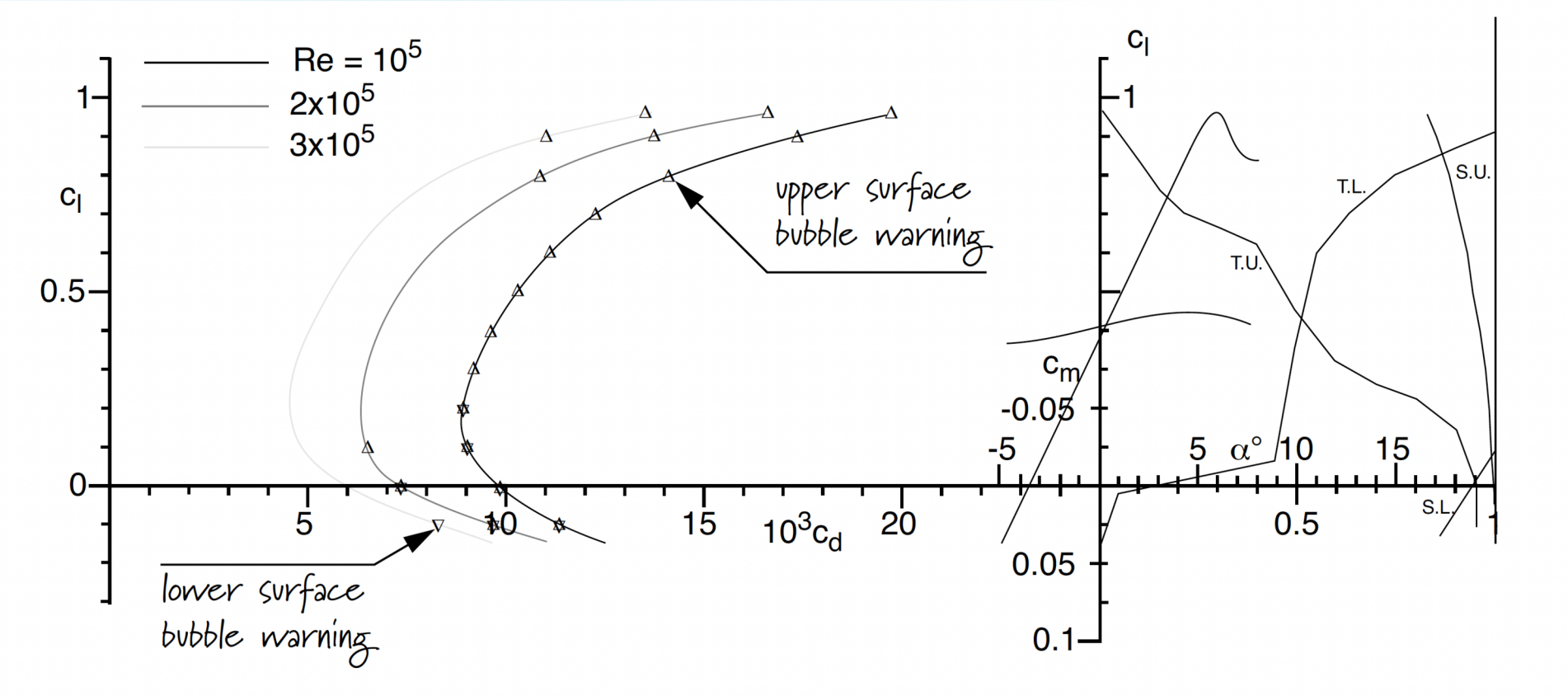

free stream flow
laminar flow
transition
turbulent flow
Note vertical displacement of flow from free stream

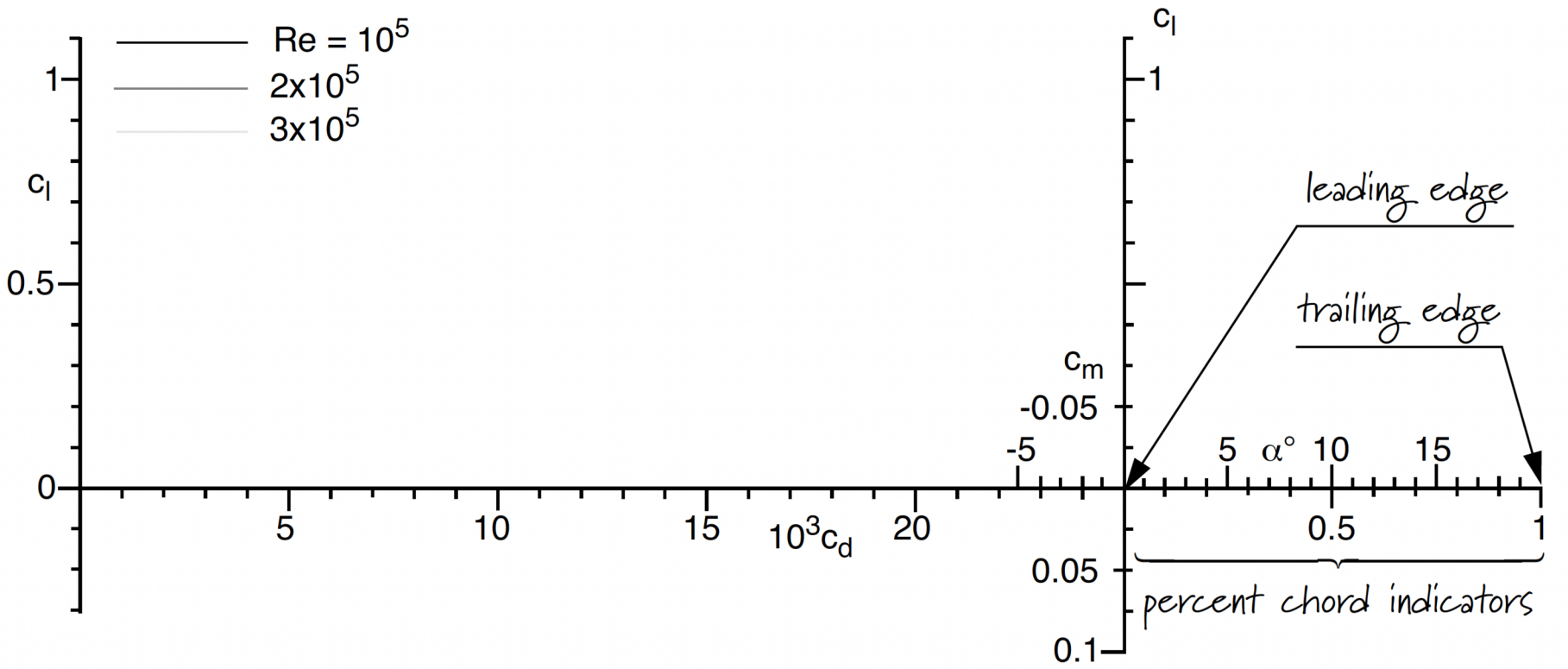

free stream flow
laminar flow

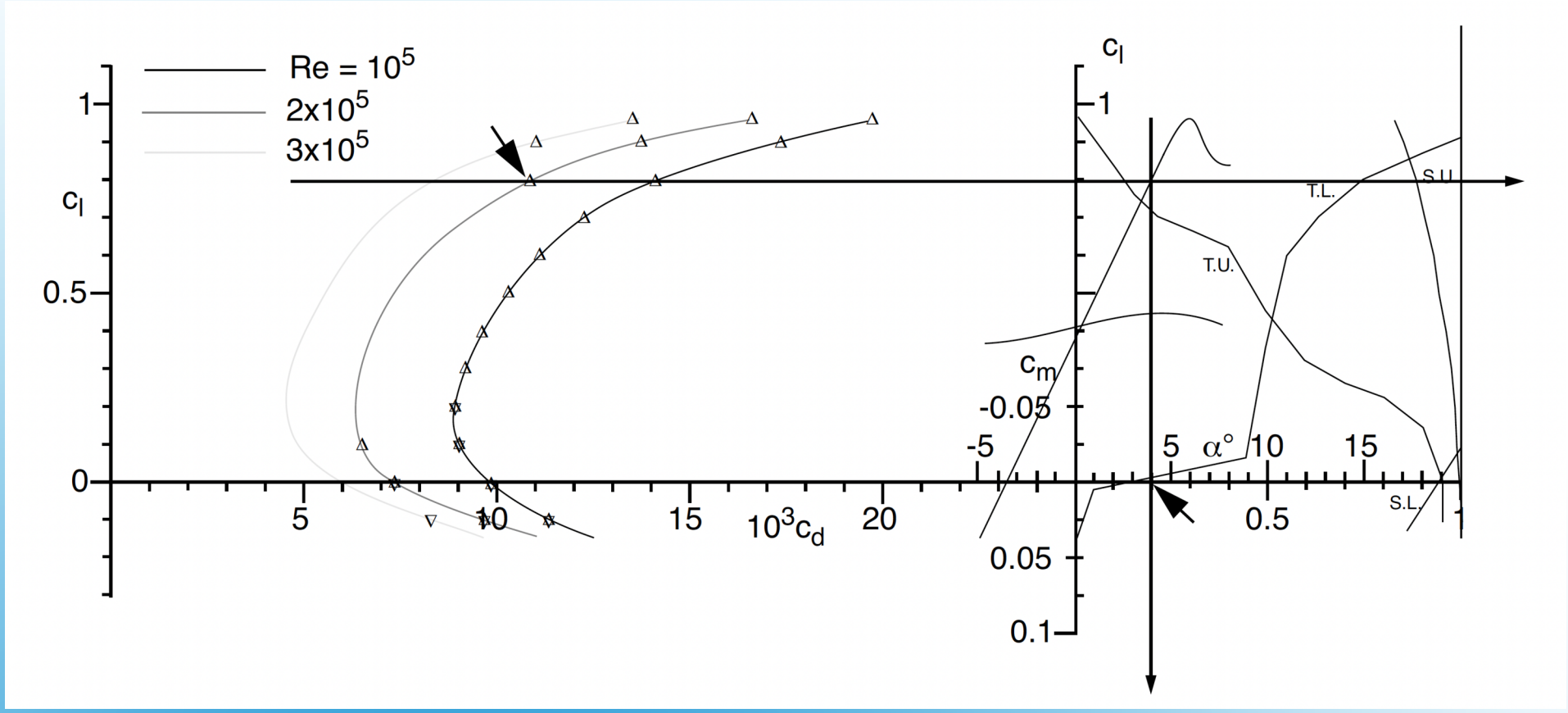

free stream flow
laminar flow
separation & transition
turbulent flow
reverse flow due to adverse pressure gradient
Note vertical displacement of flow from free stream

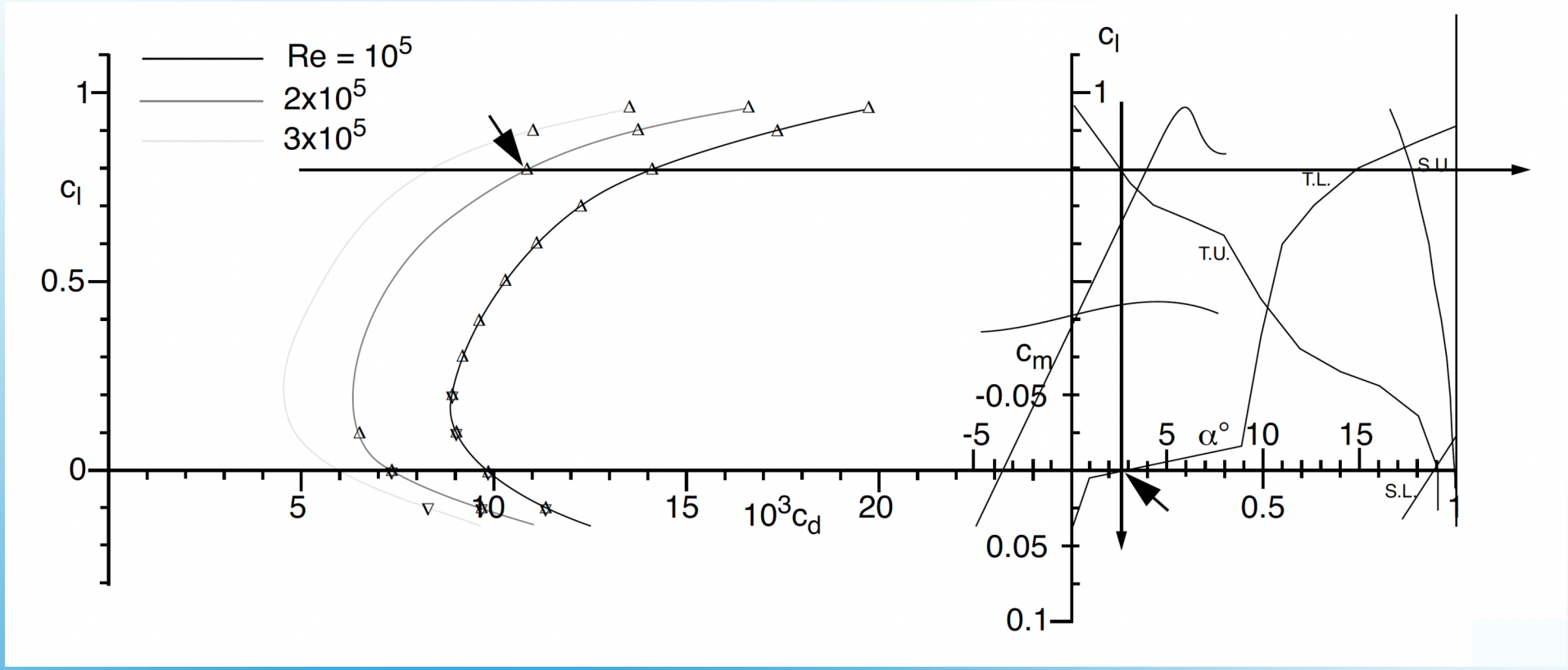

laminar separation bubble
turbulent flow
laminar flow
free stream
air flow
separation
transition
reattachment


trip strip


Re = 10^5
2x10^5
3x10^5
c_l
1
0.5
0
5
10
15
10^3c_d
20
upper surface
bubble warning
lower surface
bubble warning
c_l
1
T.L.
S.U.
T.U.
c_m
-0.05
-5
5
α°
10
15
0.5
S.L.
0.05
0.1


Re = 10^5
2x10^5
3x10^5
c_l
1
0.5
0
5
10
15
10^3c_d
20
c_l
1
c_m
-0.05
-5
5
α°
10
15
0.5
1
0.05
0.1
leading edge
trailing edge
percent chord indicators


Re = 10^5
2x10^5
3x10^5
c_l
1
0.5
0
5
10
15
10^3c_d
20
c_l
1
T.L.
S.U.
T.U.
c_m
-0.05
-5
5
α°
10
15
0.5
S.L.
1
0.05
0.1


Re = 10^5
2x10^5
3x10^5
c_l
1
0.5
0
5
10
15
10^3c_d
20
c_l
1
T.L.
S.U.
T.U.
c_m
-0.05
-5
5
α°
10
15
0.5
S.L.
0.05
0.1



The End
of Part 1

What’s going to be Part 2?

- Determining the Lift to Drag Ratio (c_l/c_d)
- Finding the Optimum Lift Coefficient ($c_{l,opt}$)
- Applying Section Characteristics to a Wing
- Putting it All Together

