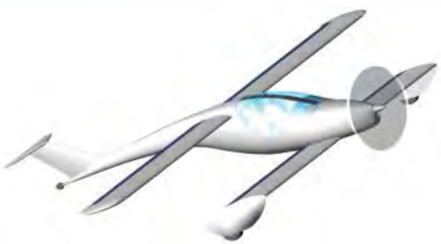
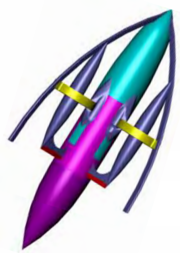
AE 521 Aerospace Design I







Dr. Ron Barrett-Gonzalez
Professor of Aerospace Engineering
The University of Kansas, Lawrence

Schrenk's Approximation

Good news: Newly developed shear panel analysis arrives in Douglas Aircraft Corporation, February 1930… but there's a problem.

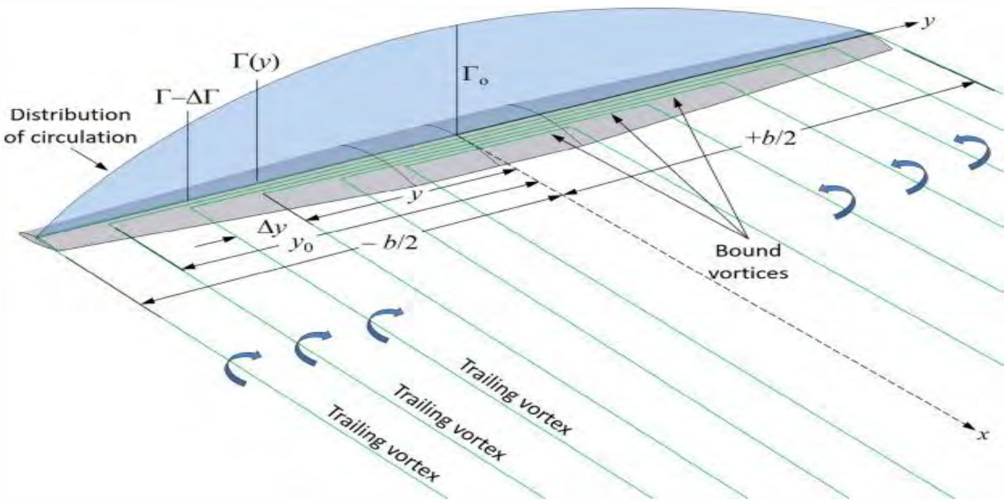
How to estimate loads?

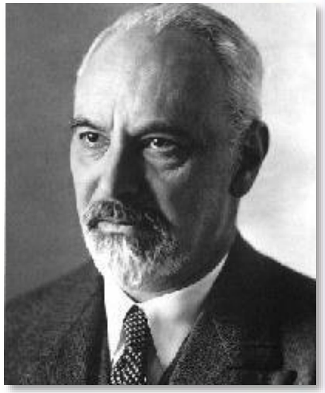
Aero. Engineering Veggies

Hmmm… in 1914 Prandtl figured out that an elliptical wing loading was ideal, but…

Few people really know what the ideal lift distribution should look like…

And fewer know how to model it.





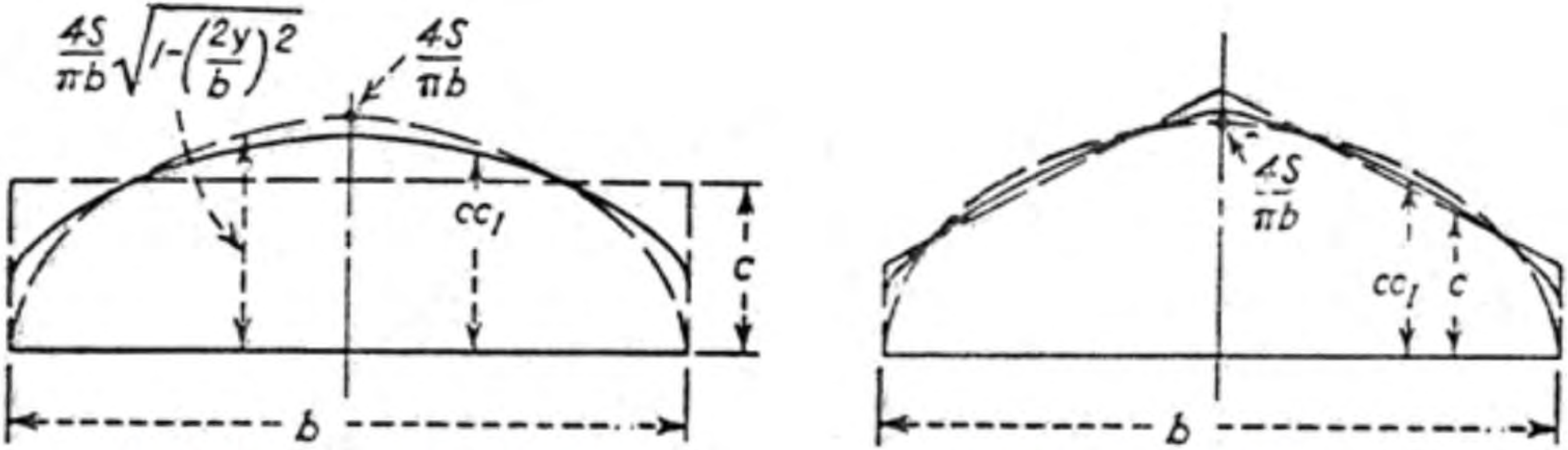
Aero. Engineering Veggies

Order to Chaos:

Dr. Ing. Oster Schrenk

Schrenk's Method:

Assume the span load is the average of geometric loading and elliptical loading:

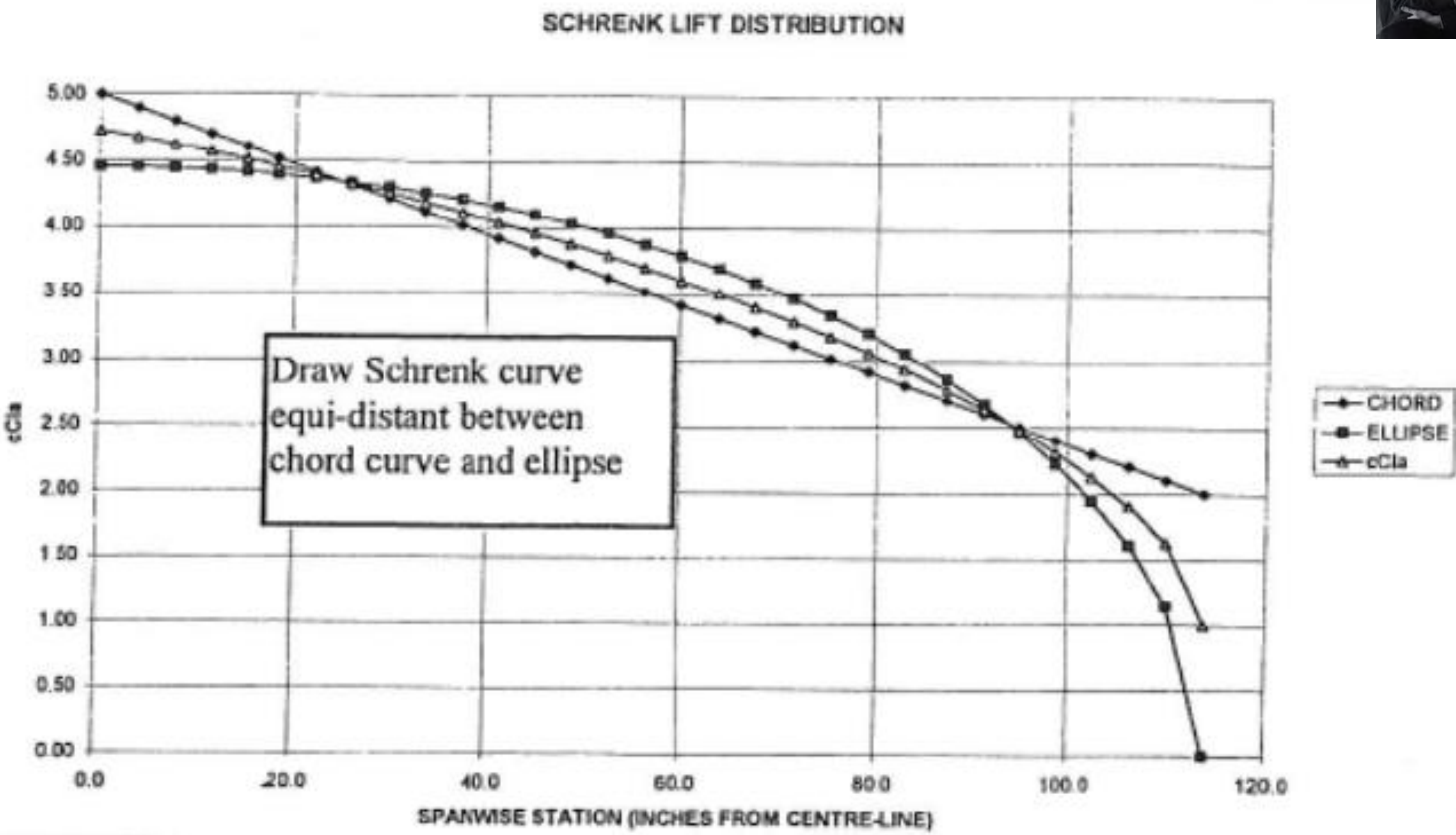


(a) Rectangular planform　　*(b) Trapezoidal planform*

Aero. Engineering Veggies

Schrenk's Method:

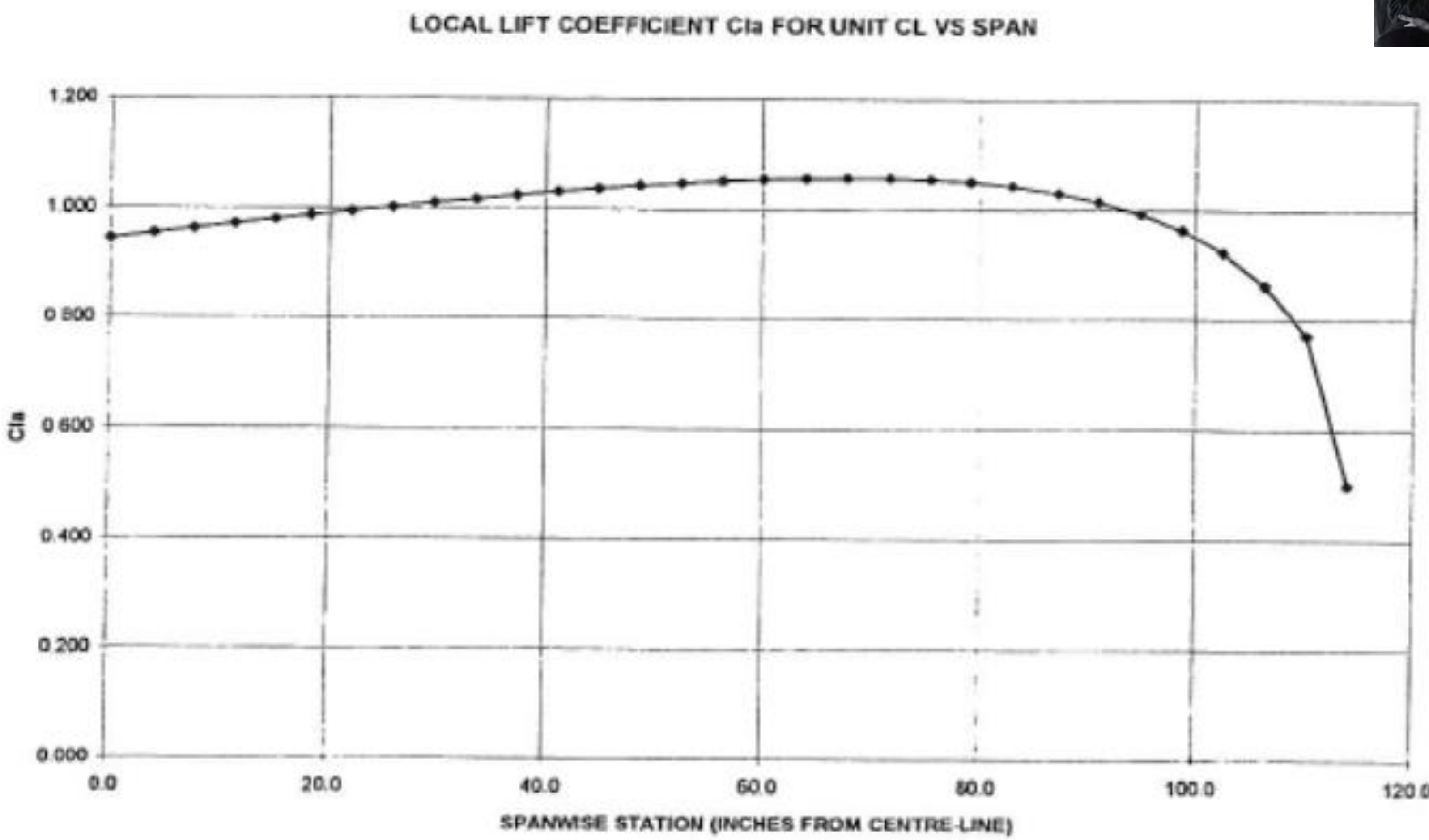
Assume the span load is the average of geometric loading and elliptical loading:



Aero. Engineering Veggies

Schrenk's Method:

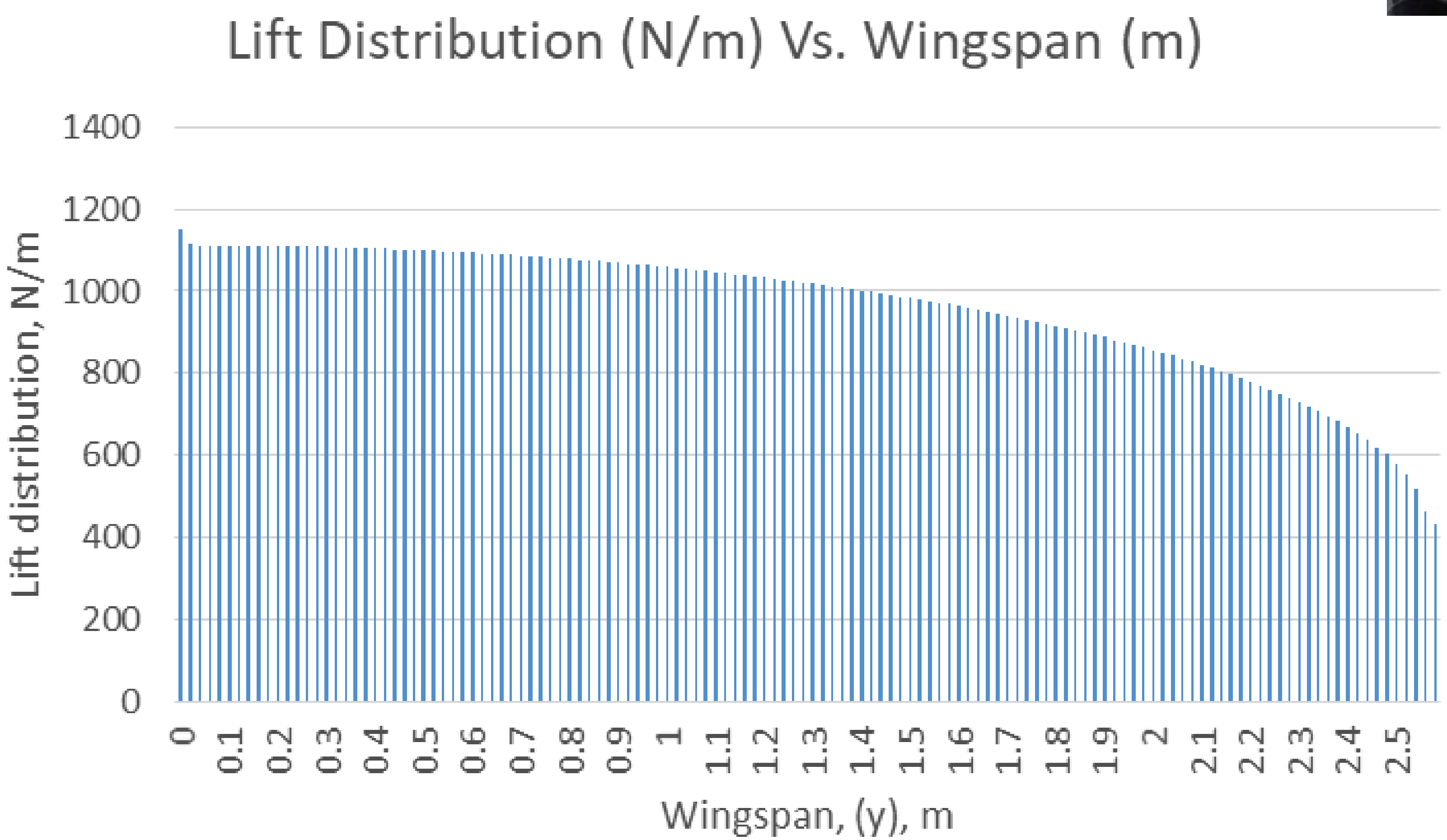
Assume the span load is the average of geometric loading and elliptical loading:

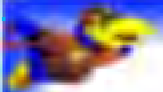


Aero. Engineering Veggies

Schrenk's Method:

Assume the span load is the average of geometric loading and elliptical loading:

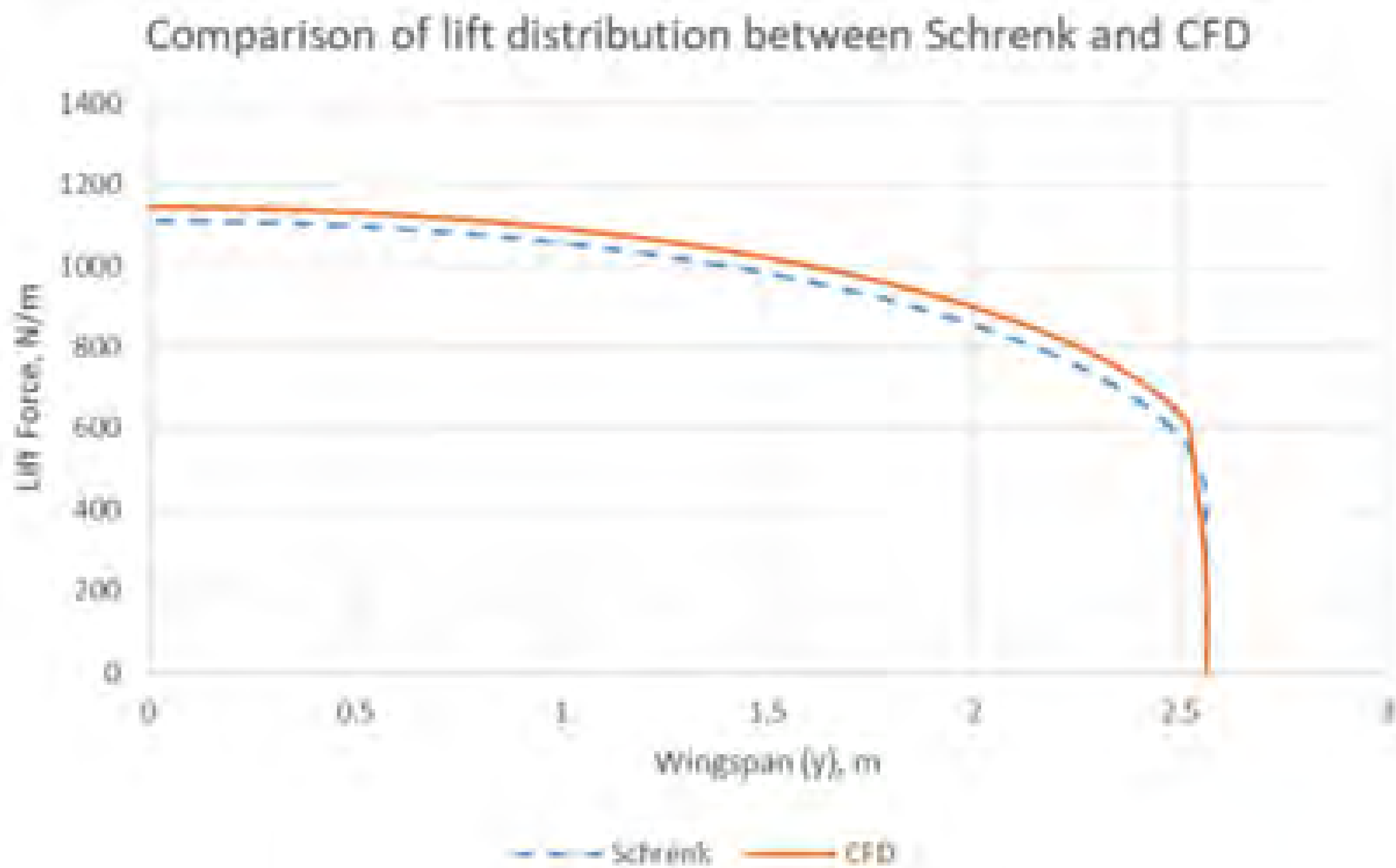




Aero. Engineering Veggies

Schrenk's Method:

Assume the span load is the average of geometric loading and elliptical loading:



Aero. Engineering Veggies

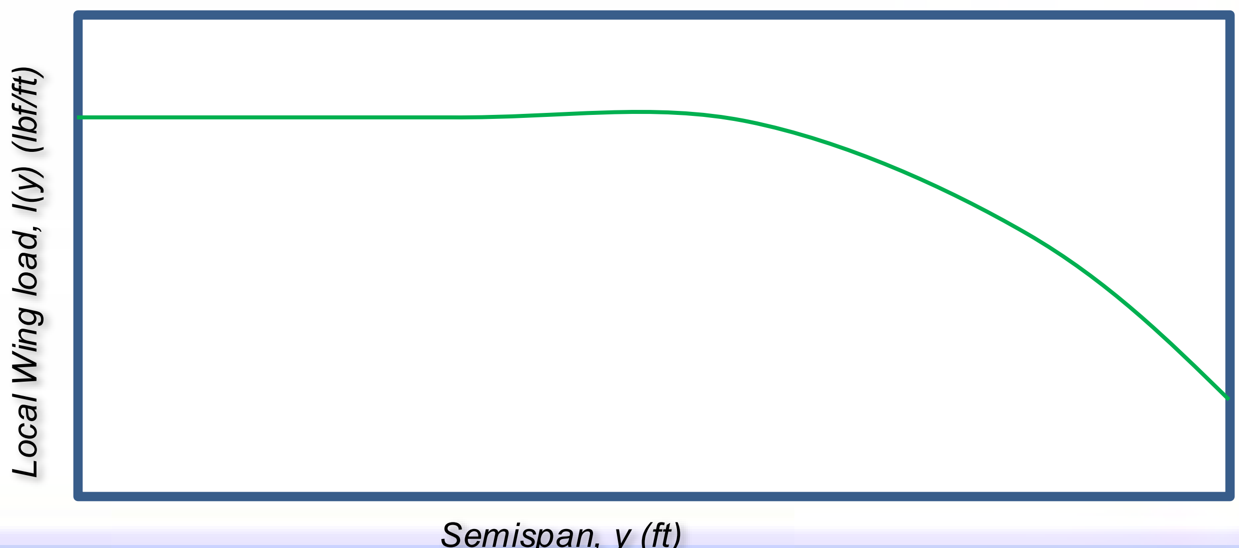
Good news: Newly developed shear panel analysis arrives in Douglas Aircraft Corporation, February 1930… but there's a problem.

$$Load = \frac{Load_{elliptic} + Load_{trapezoidal}}{2}$$

$$Load_{elliptic} = \frac{4 * MaxLoad}{\pi * Wingspan}\sqrt{\left(1 - \left(\frac{2 * y}{Wingspan}\right)^2\right)}$$

$$Load_{trapezoidal} = \frac{2 * MaxLoad}{Wingspan(1 + TaperRatio)}\left(1 - \left(\frac{2 * y}{Wingspan}\right)(1 - TaperRatio)\right)$$

Step 2: Plot load(y) (lb/ft)



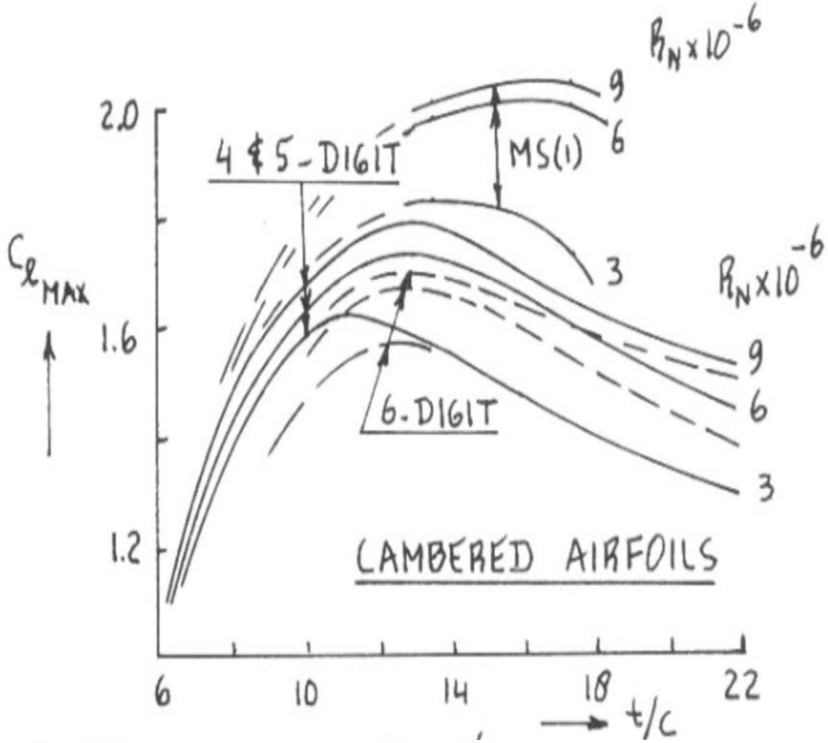
Aero. Engineering Veggies

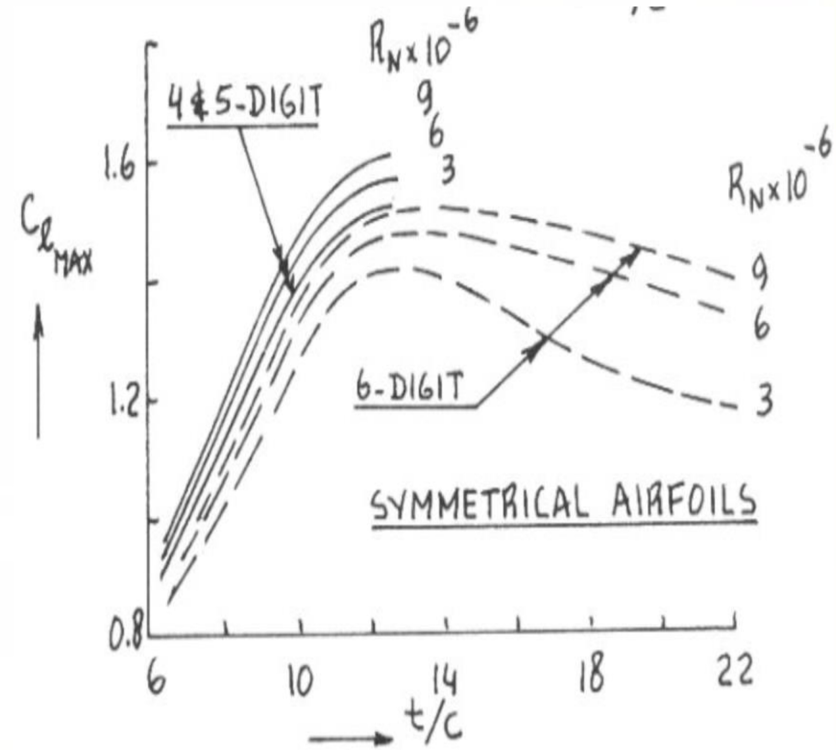
Good news: Newly developed shear panel analysis arrives in Douglas Aircraft Corporation, February 1930… but there's a problem.

Step 4: Get $C_{lmaxr}(Rn)$ and $C_{lmaxt}(Rn)$

$$R_{nr} = \frac{r V c_r}{m}$$

$$R_{nt} = \frac{r V c_t}{m}$$

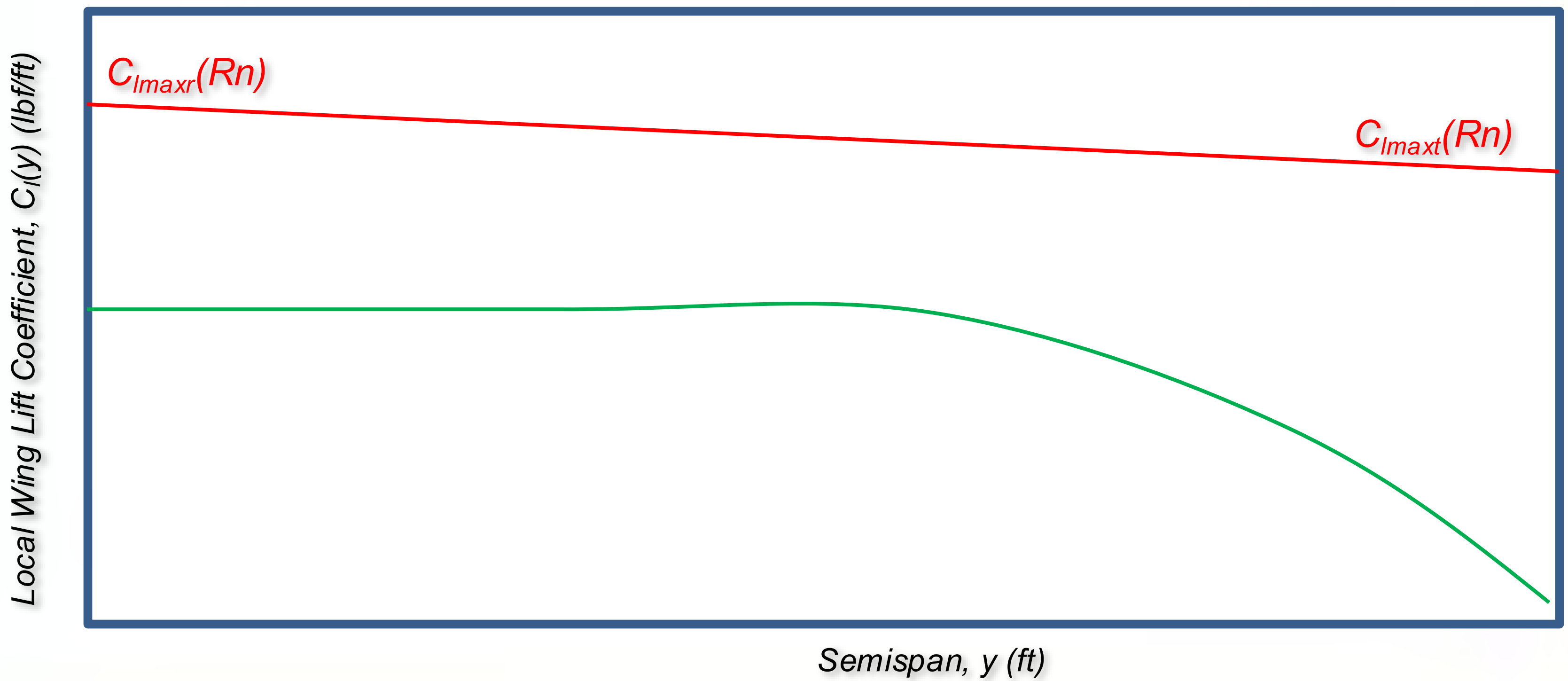




Aero. Engineering Veggies

Good news: Newly developed shear panel analysis arrives in Douglas Aircraft Corporation, February 1930… but there's a problem.

Step 4: Get $C_{lmaxr}(Rn)$ and $C_{lmaxt}(Rn)$ and *connect*



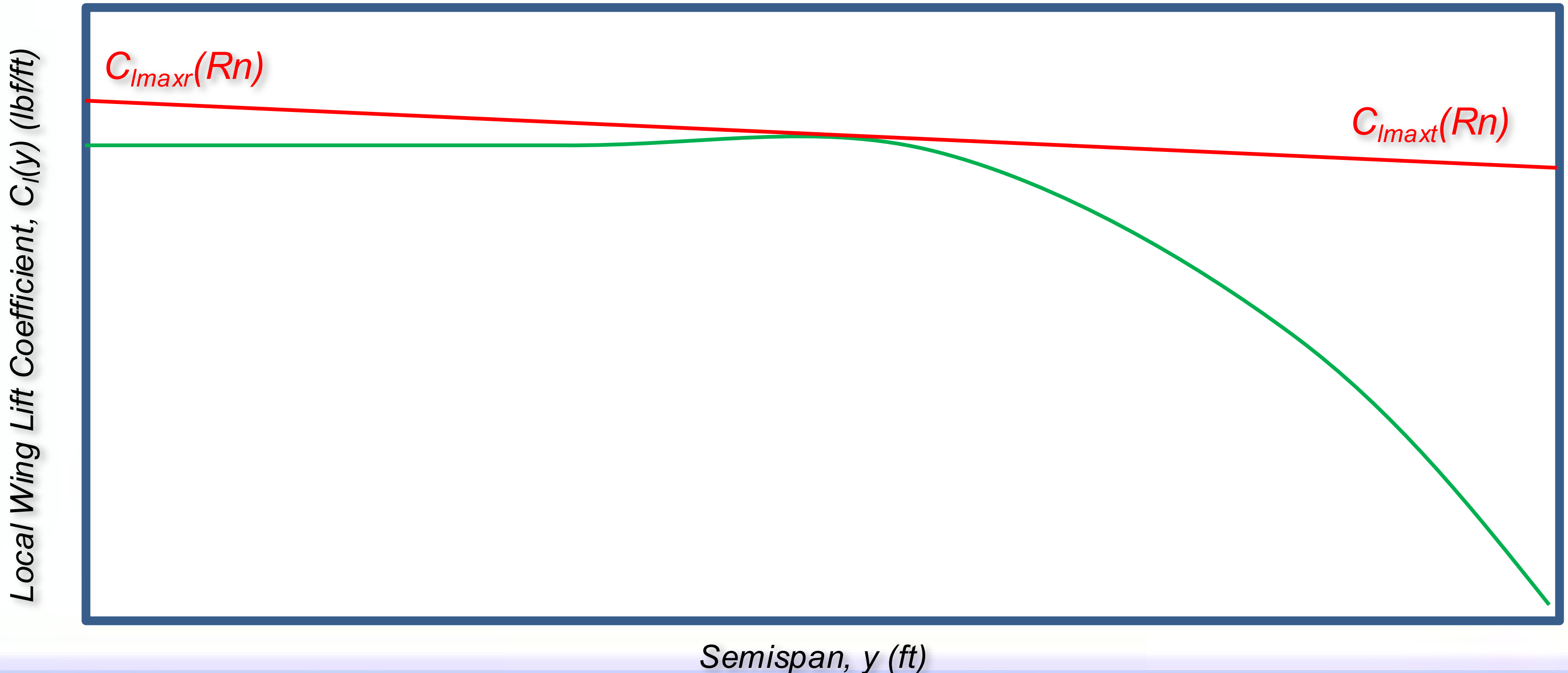
Aero. Engineering Veggies

Good news: Newly developed shear panel analysis arrives in Douglas Aircraft Corporation, February 1930… but there's a problem.

Step 5: Increase total wing lift till wing lift coefficient trend line ($C_l(y)$) touches the C_{lmax} line.

Step 6: Note the total load at that condition – that's $L_{maxwing}(Rn,Mach)$

Step 7: Divide by qS to get $C_{Lmaxclean}$ (or other flight condition)

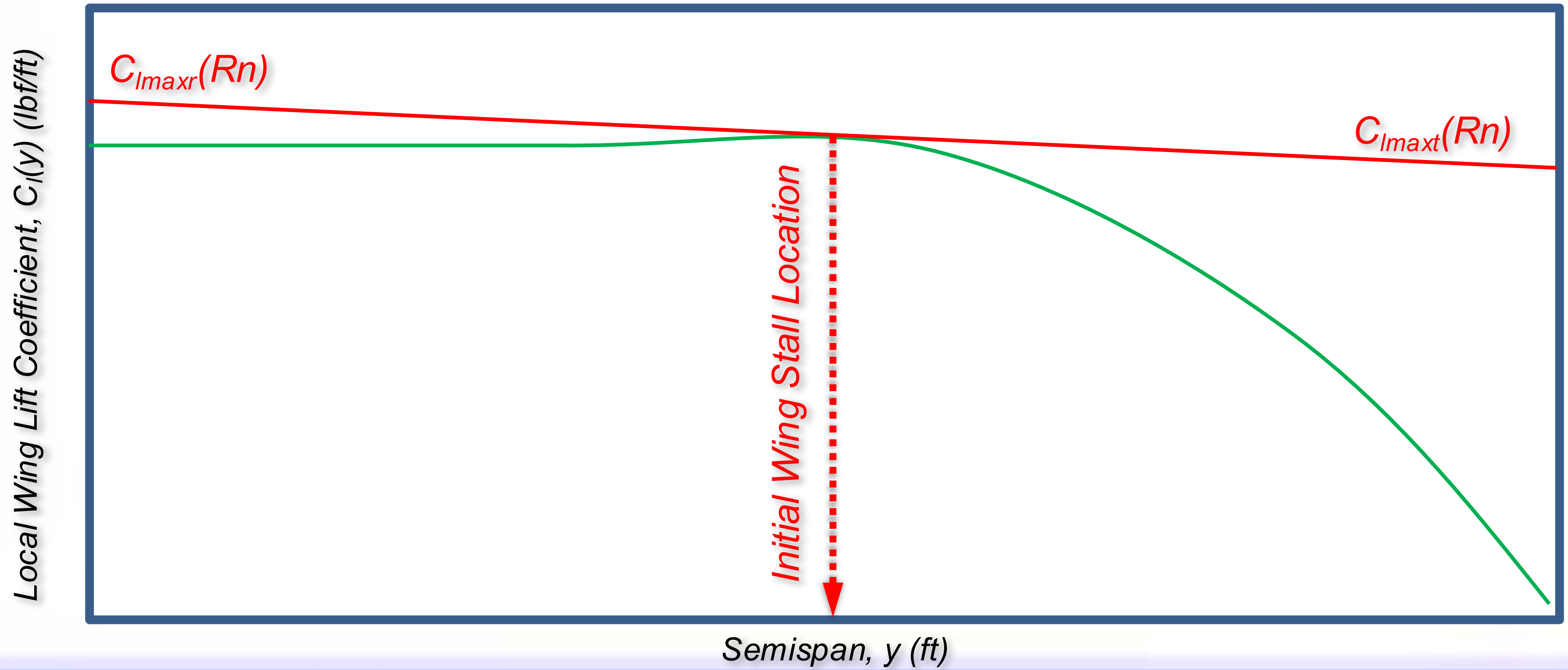


Aero. Engineering Veggies

Good news: Newly developed shear panel analysis arrives in Douglas Aircraft Corporation, February 1930… but there's a problem.

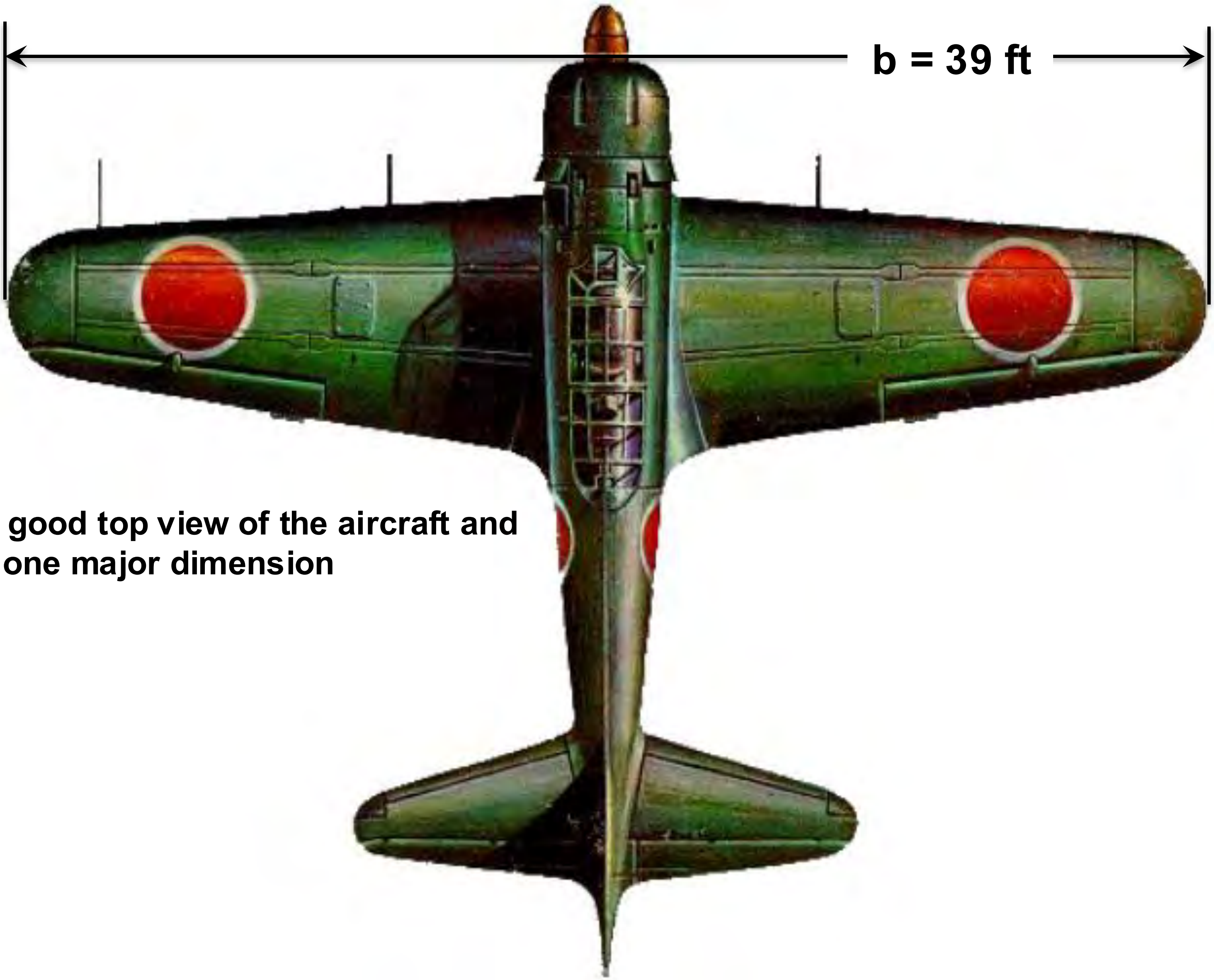
Locate position where lift tangency occurs. That is where stall will first occur. If near ailerons, change wing design:

a. Include washout
b. Change airfoil section type distribution
c. Droop leading edge



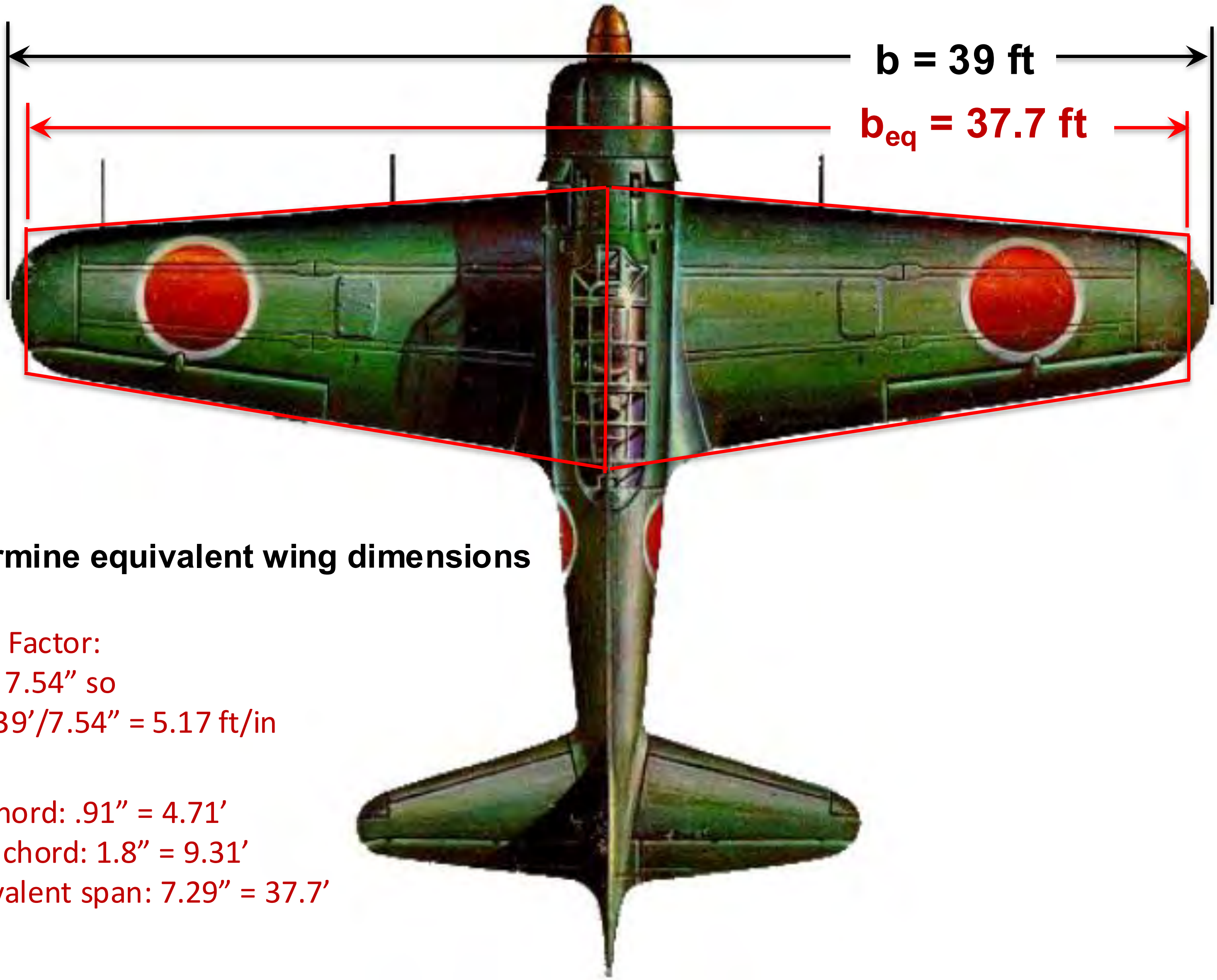
Estimate Wing Load Distribution on a Mitsubishi A6M Zero & Stall

Estimate Wing Load Distribution on a Mitsubishi A6M Zero & Stall



1. Get a good top view of the aircraft and at least one major dimension

Estimate Wing Load Distribution on a Mitsubishi A6M Zero & Stall

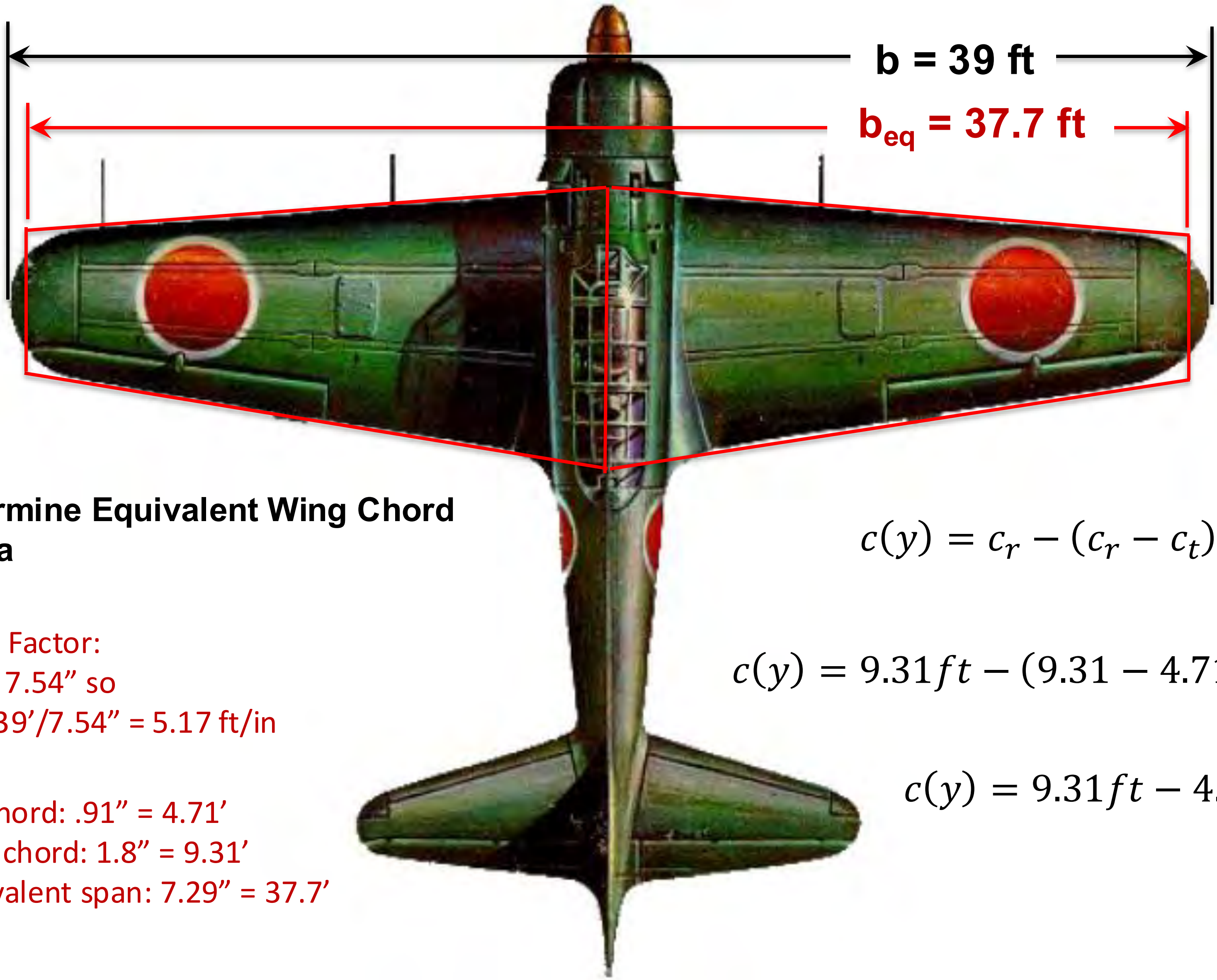


2. Determine equivalent wing dimensions

Scale Factor:
39’ = 7.54” so
SF = 39’/7.54” = 5.17 ft/in

Tip chord: .91” = 4.71’
Root chord: 1.8” = 9.31’
Equivalent span: 7.29” = 37.7’

Estimate Wing Load Distribution on a Mitsubishi A6M Zero & Stall



3. Determine Equivalent Wing Chord Formula

Scale Factor:
39’ = 7.54” so
SF = 39’/7.54” = 5.17 ft/in

Tip chord: .91” = 4.71’
Root chord: 1.8” = 9.31’
Equivalent span: 7.29” = 37.7’

$$c(y) = c_r - (c_r - c_t)\frac{y}{b/2}$$

$$c(y) = 9.31ft - (9.31 - 4.71ft)\frac{y}{b/2}$$

$$c(y) = 9.31ft - 4.6\frac{y}{b/2}ft$$

Estimate Wing Load Distribution on a Mitsubishi A6M Zero & Stall

4. Estimate nominal combat weight:

W_{to} = 6,331lb

W_f = 150gal * 6.01lb/gal = 902lb

$W_{nom} = W_{to} – 0.4W_f$ = 6,331lb = 0.4*902lb = 5,970lb

5. Estimate the wing lift:

$L = nW_{nom}$ = n*5,970lb

6. Choose nondimensional semispan slices (1 – 2% is typically good)

7. Calculate local dimensional span slice width (ft)

8. Calculate the trapezoidal equivalent chord distribution:

$$c_{trapeq}(y) = c_r - (c_r - c_t)\frac{y}{b/2}$$

9. Calculate the elliptical equivalent chord distribution:

$$c_{ellipeq} = \left(\frac{c_r + c_t}{2}\right)\left(\frac{4}{\pi}\right)\sqrt{1 - \left(\frac{y}{b/_2}\right)^2}$$

10. Calculate the nominal equivalent chord distribution:

$$c_{nom} = \frac{c_{ellipeq} + c_{trapeq}}{2}$$

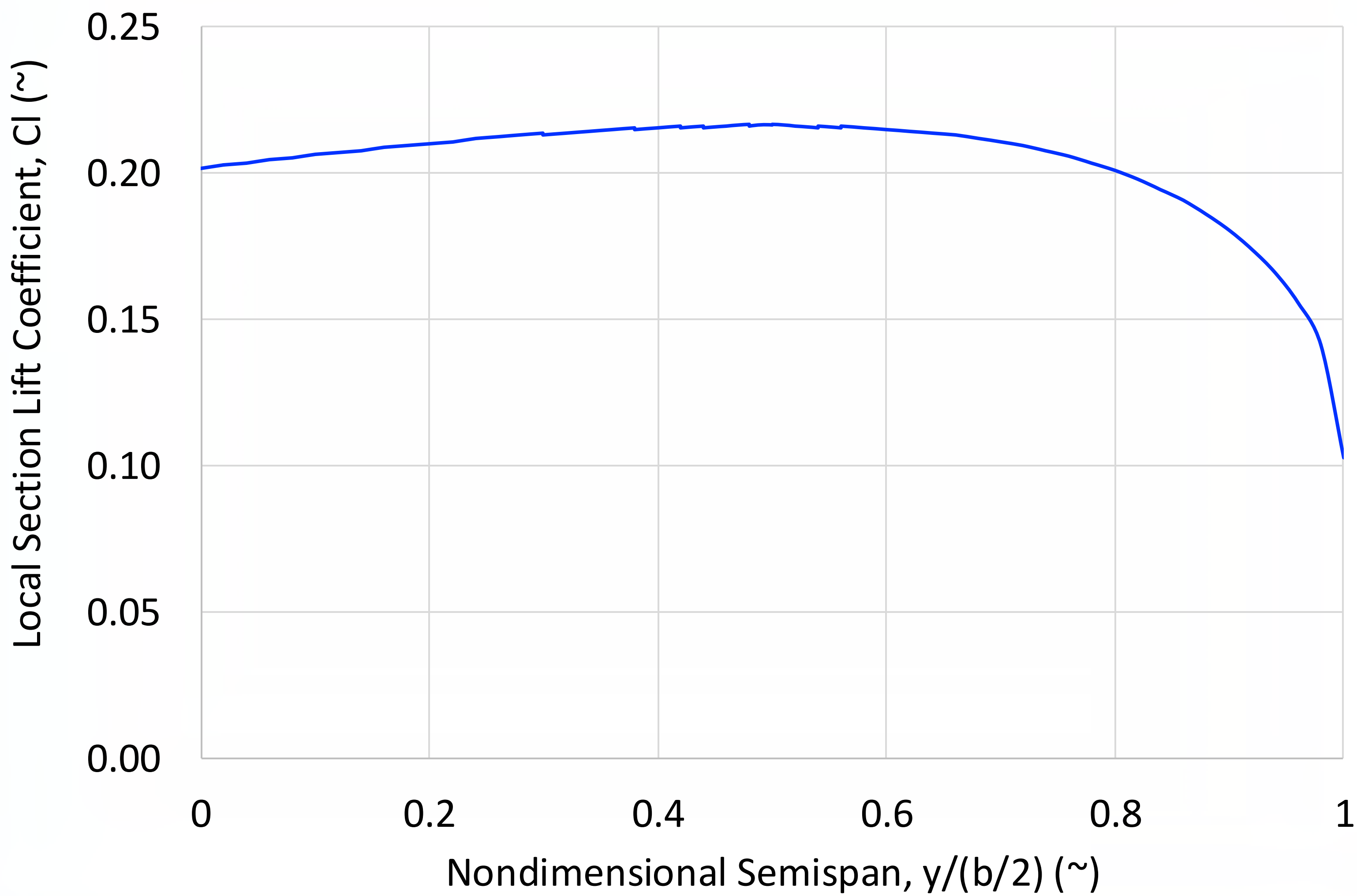
11. Calculate local lift for each strip:

$$\mathrm{L}'(\mathrm{y})\ (\mathrm{lbf}) = \left(\frac{\mathrm{L}}{\mathrm{S}}\right) \mathrm{c}_{\mathrm{nom}} \Delta\, \mathrm{y}$$

12: Calculate local c_l, normalized for local chord

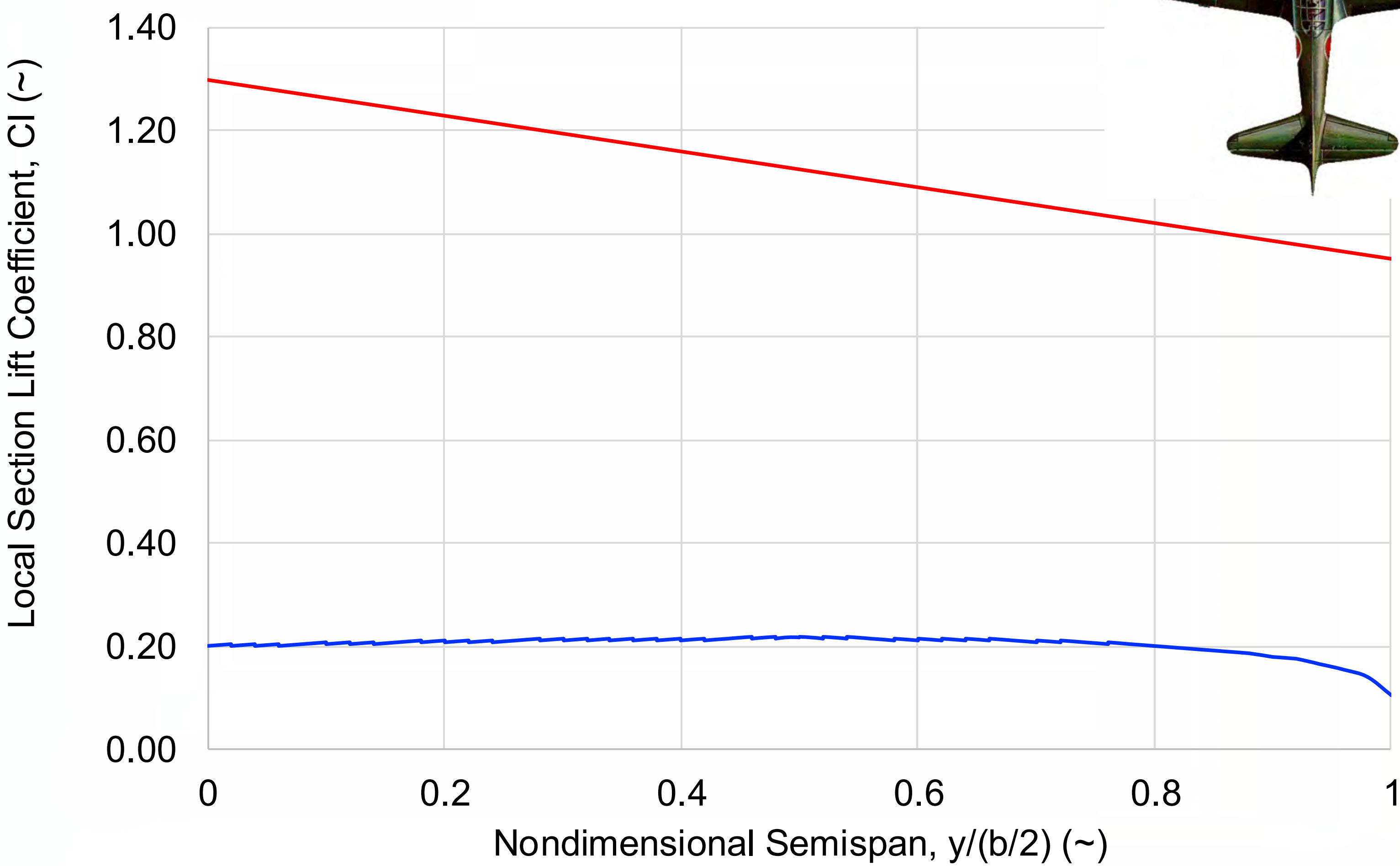
$$c_l(y) = \frac{L'(y)}{q c_{trapeq}(y)\Delta y}$$

13. Plot local $C_l(y)$



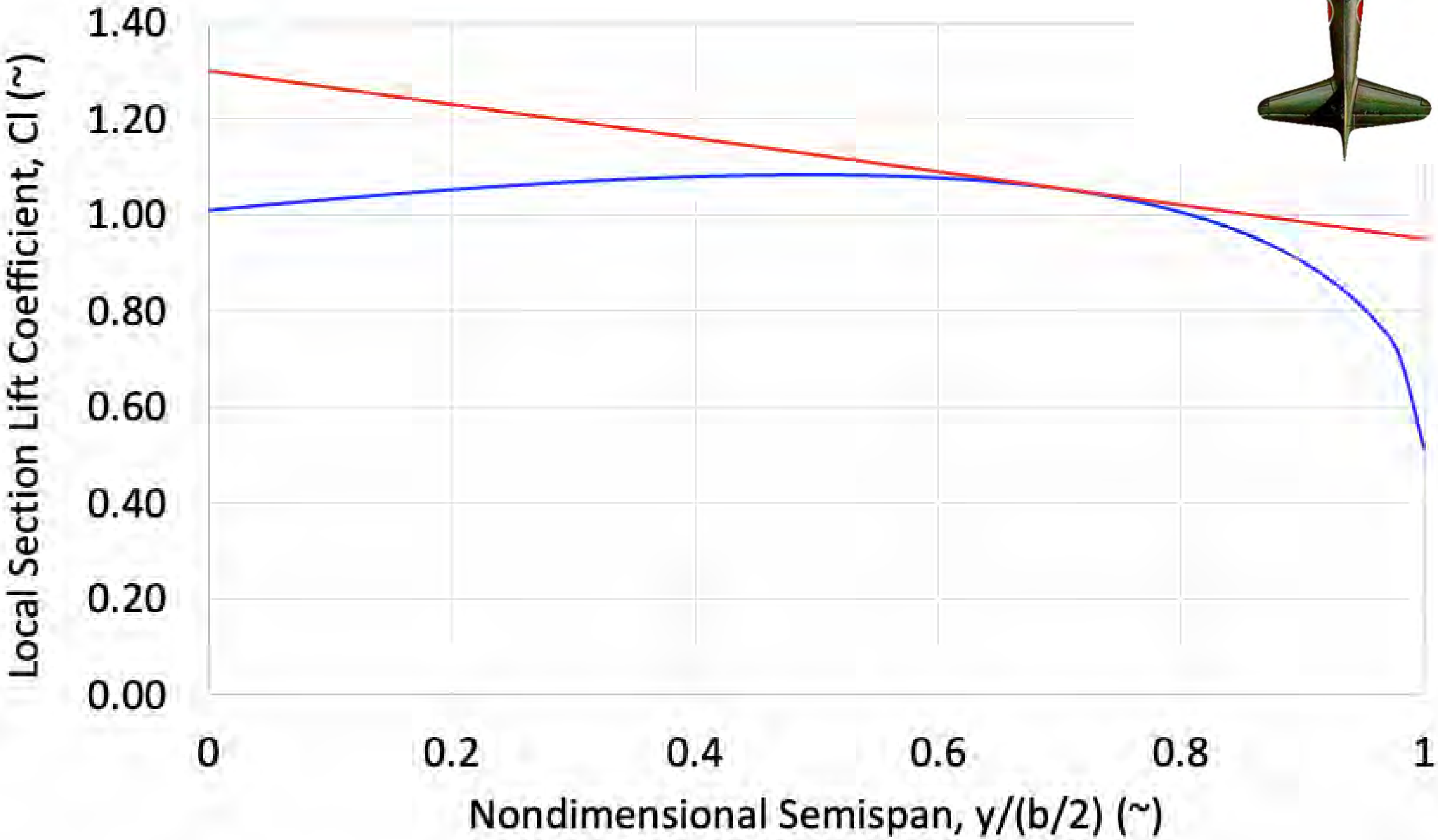
Estimate Wing Load Distribution on a Mitsubishi A6M Zero & Stall

14. Get root and tip C_{lmax} values C_{lmaxr} & C_{lmaxt} and plot



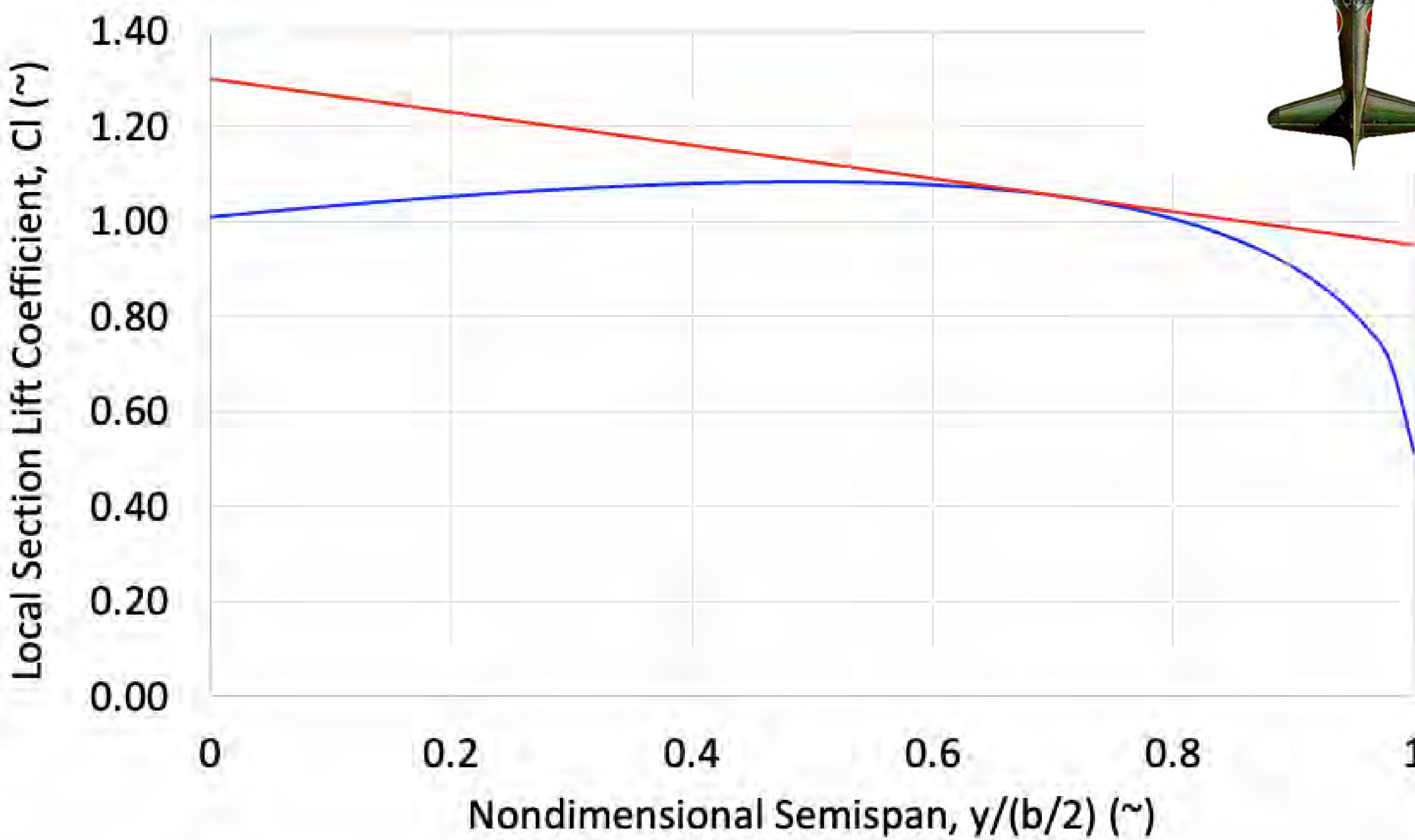
Estimate Wing Load Distribution on a Mitsubishi A6M Zero & Stall

15. Increase n at Vcr till the two lines touch (in this case, n = 5g's)



Estimate Wing Load Distribution on a Mitsubishi A6M Zero & Stall

16. Hold n = 1, decrease speed till lines touch (in this case 80.5kts)



Estimate Wing Load Distribution on a Mitsubishi A6M Zero & Stall

17. Calculate local bending moment increment: $M'(y) = yL'(y)$

18. Integrate local bending moments from tip to root & plot: