University of Cincinnati
College of Engineering and Applied Science

Course Code : **MECH-3011**
Course Title : **Fluid Mechanics**

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1. Part A: Confirmation of Tutorial Completion

At the beginning of this project, a tutorial on the operation of FLUENT was conducted by each member of the group. The below signatures verify that each member has confirmed their completion of Part A as well as their contribution to the overall output of the project.

Name	Contribution %	Signature
Alex Braidich	25 %	Alex Braidich
Matt Stang	25%	Matt Stang
Zach Wells	25%	Zach Wells
Spencer Ellmaker	25%	[illegible]

2. Part B: Designing an Aircraft

2.1 Problem Definition

The type of plane that was designed for was a small medium altitude aircraft. It will be able to carry four passengers (including the pilot) and will cruise at a speed of 53.611 m/s (≈120mph) at an altitude of 1500m (≈4920 ft). In order to determine an estimate for the weight of the aircraft, the following calculation was completed and then verified to seem logical through literature analysis:

Weight = (# of people * kg/person) + (weight of frame)

= (4 people * 90.7185kg/person) + (640kg)

Weight ≈ 1,000 kg

Based on the literature review, the approximate sizing of the designed aircraft was 7.75m long with a 1.5m diameter fuselage shell. The design was assumed to be modelled as a cylinder with a hemispherical nose cap, wings (chord and span to be determined after further analysis), and three small rear flaps (see Appendix A page 1 or Figure 7 for a detailed sketch of the designed plane's sizing).

2.2 Airfoil Selection

For the airfoil analysis, the NACA 2412 airfoil was selected. This conclusion was reached after investigating the airfoil of a similarly sized small aircraft, the Cessna 170B, and its airfoil styling. The NACA 2412 airfoil possesses a relatively symmetric cross section (at 40% of the chord length a maximum camber of 2% exists) which will be functional for the low stress flying operation that the proposed airplane design will be regularly performing. Page 1 of Appendix B shows the geometry of the NACA 2412 airfoil and page 2 shows information for the lift and drag coefficients with respect to a Reynolds number in the magnitude of 10^6 plotted against varying angles of attack. In addition, a published journal on a wind tunnel analysis of the NACA 2412 airfoil at various Reynold's numbers was obtained that was also used for reference throughout this project (see Appendix B page 3).

2.3 Airfoil Simulation

2.3.1 At angle of attack = 0°

For the initial CFD analysis of the NACA 2412 airfoil, an angle of attack of zero degrees was used. As in the tutorial of part A, the airfoil was analyzed against inviscid flow for a large C-style boundary region. Figures 1 and 2 below illustrate the results of the initial CFD simulation.

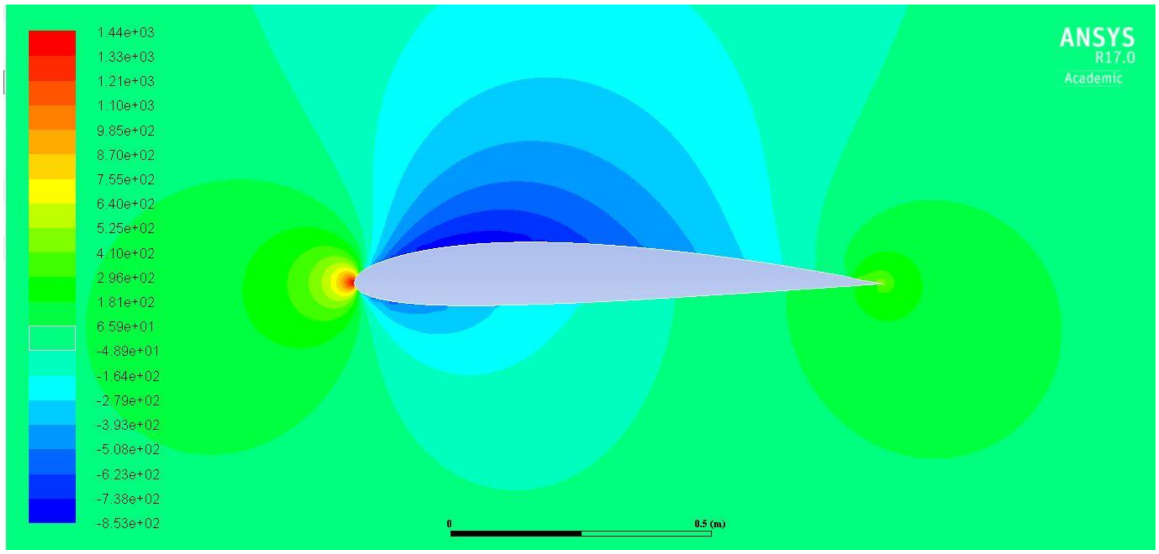


Figure 1. Countered Pressure at α = 0°

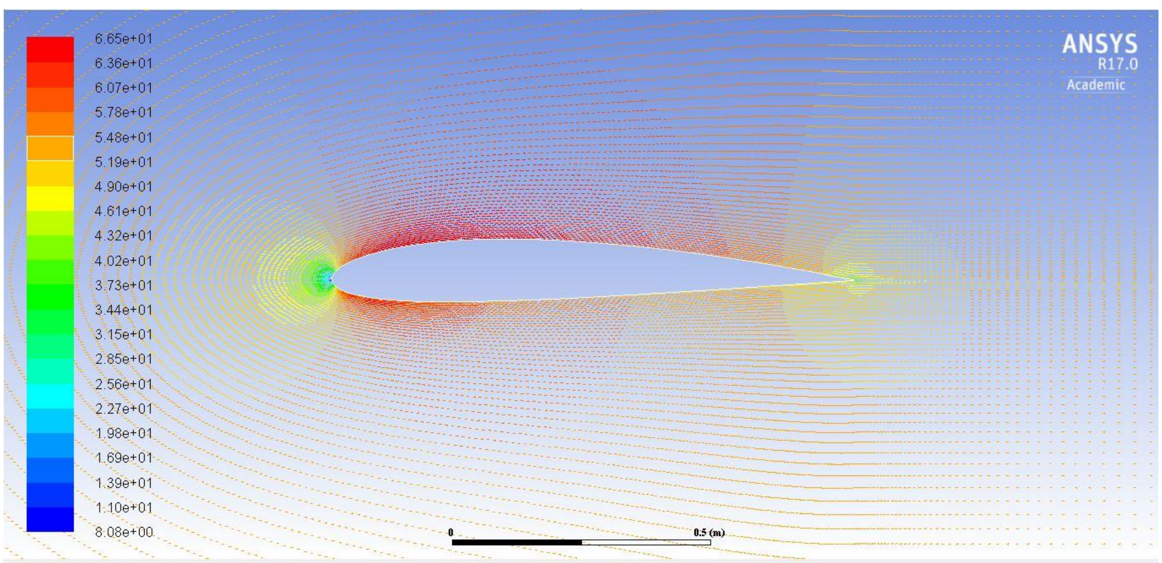


Figure 2. Velocity Magnitudes at α = 0°

Source	Lift Coefficient	Drag Coefficient	Lift to Drag Ratio
CFD Analysis (α = 0°)	0.20769527	0.00096447976	215.3443
Airfoiltools.com	0.25	0.005	50
IJERG journal	0.261	0.012	21.75

Table 1. Lift and Drag Data at α = 0°

2.3.2 At angle of attack = 5°

For the second CFD analysis of the NACA 2412 airfoil, an angle of attack of five degrees was used. Again, the airfoil was analyzed against inviscid flow for a large C-style boundary region. Figures 3 and 4 below illustrate the results of the second CFD simulation.

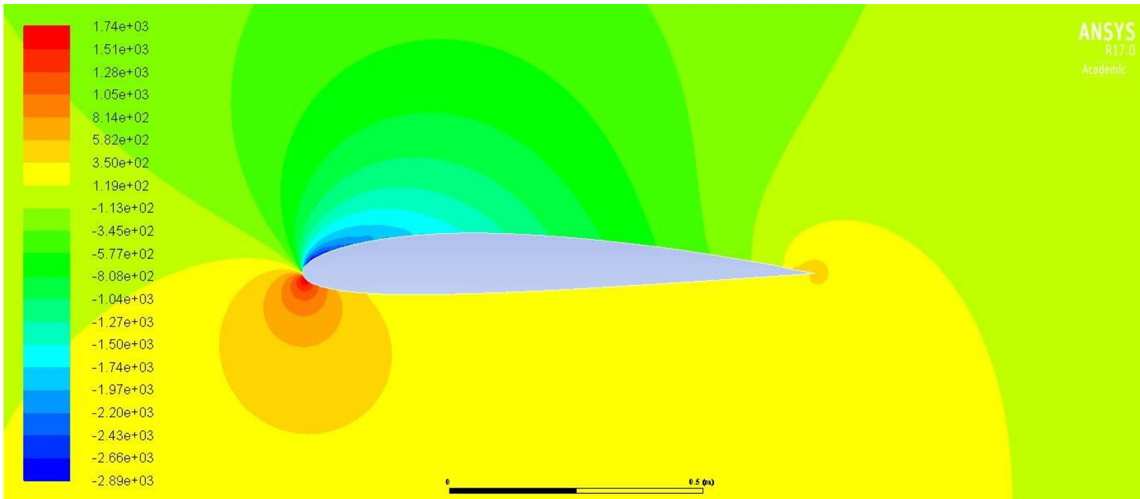


Figure 3. Countered Pressure at α = 5°

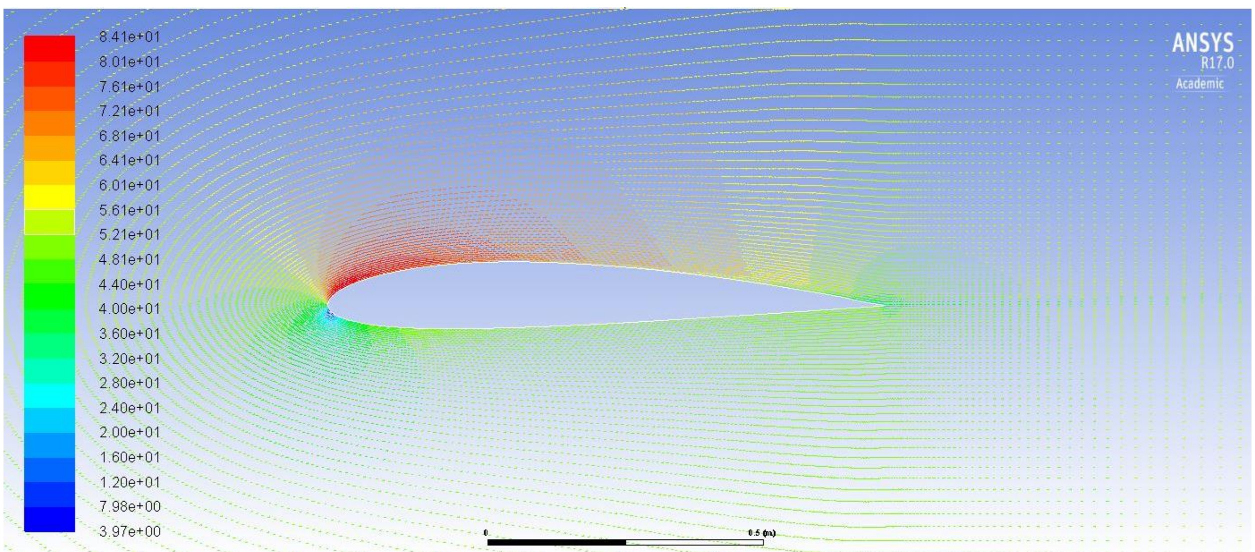


Figure 4. Velocity Magnitudes at α = 5°

Source	Lift Coefficient	Drag Coefficient	Lift to Drag Ratio
CFD Analysis (α = 5°)	0.59296891	0.003935708	150.6638
Airfoiltools.com	0.75	0.008	93.75
IJERG journal	0.733	0.0148	49.527

Table 2. Lift and Drag Data at α = 5°

2.3.3 At angle of attack = 10º

For the second CFD analysis of the NACA 2412 airfoil, an angle of attack of five degrees was used. Again, the airfoil was analyzed against inviscid flow for a large C-style boundary region. Figures 3 and 4 below illustrate the results of the second CFD simulation.

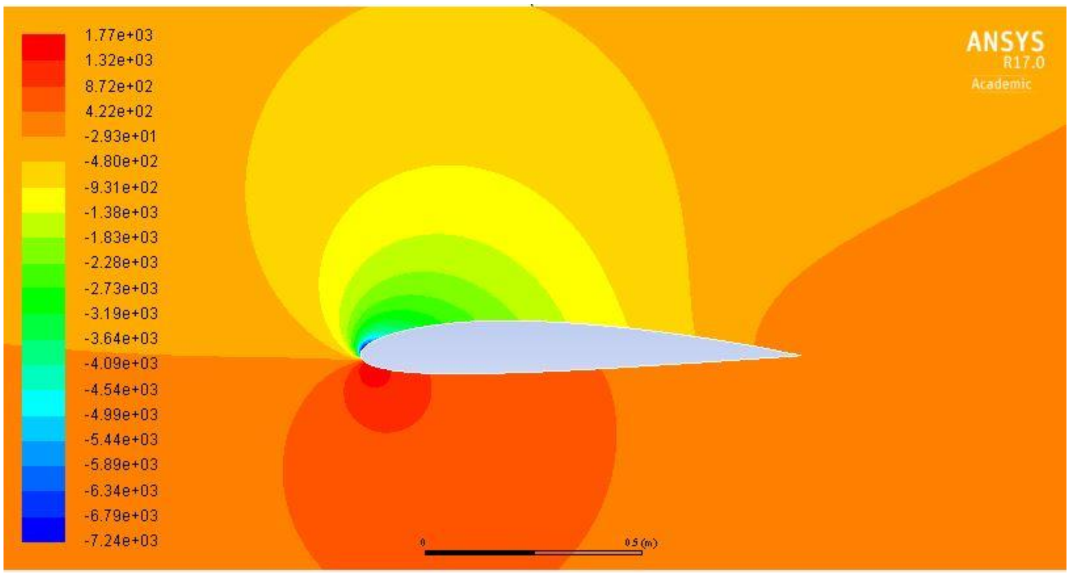


Figure 5. Countered Pressure at α = 10°

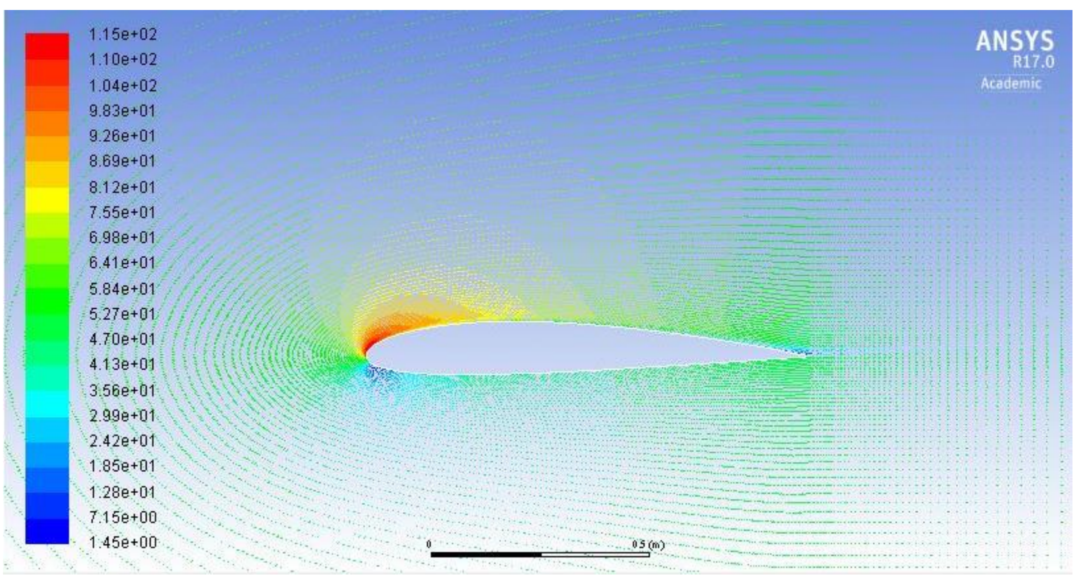


Figure 6. Velocity Magnitudes at α = 10°

Source	Lift Coefficient	Drag Coefficient	Lift to Drag Ratio
CFD Analysis (α = 10°)	1.2772977	0.012827562	99.5745
Airfoiltools.com	1.25	0.018	69.44
IJERG journal	1.139	0.024	47.458

Table 3. Lift and Drag Data at α = 10°

2.3.4 At angle of attack = 15°

For the final attempted CFD analysis of the NACA 2412 airfoil, an angle of attack of fifteen degrees was used. This value was originally desired to be analyzed as an interview that our group conducted with a professional pilot concerning the similarly styled Cessna 170B stated that an angle of attack of fifteen degrees was ideal. However, when the airfoil was analyzed against inviscid flow for a large C-style boundary region the simulation would not converge. The main assumption for why this simulation was unsuccessful is that due to the modelling of the flowing fluid as inviscid that at this angle of attack the airfoil was actually stalling.

- Reynolds number general for airplane design body
  - Re = $\frac{VD}{\nu} = \frac{53.611\frac{m}{s} * 1.5m}{1.5e{-5}\ \frac{m^2}{s}}$ = 5.36e6

2.3.5 General conclusions

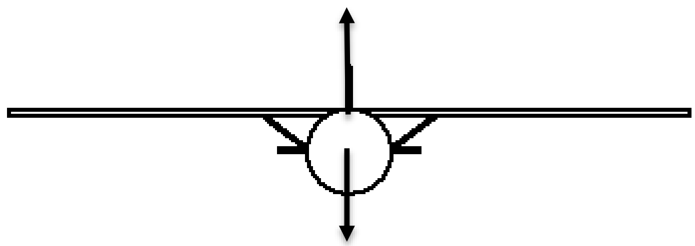
The optimized angle of attack used for this discussion was five degrees as the Reynolds number from Airfoiltools.com (1e6) was more representative of the Reynolds number for the proposed plane design (5.36e6 – see section 2.4 for calculation) than the International Journal of Engineering Research and General Science article (1e5). At the angle of five degrees is where the maximum lift to drag ratio occurred for the Airfoiltools.com data as seen in Table 2. As a general trend it was noted that the lift coefficients calculated from the CFD simulation were either close or slightly under (within twenty percent) the charted values from each of the literature sources. On the other hand, the drag coefficients from CFD were always significantly less than reported in the literature. To understand this phenomenon, the flow condition in each source of information was considered. As the CFD simulation was the only source where the flow was modeled as inviscid, this was investigated as the potential source for this discrepancy. It can be concluded that the inviscid flow model is in fact the source of error between the simulated and reported data by further considering the operation of an airfoil in general. An airfoil is designed to be streamlined, therefore implying that friction drag will dominate over pressure drag as the leading opposing force to the motion of the wing. By definition, inviscid flow neglects interactions due to viscous (friction) forces thus explaining why all of the CFD values were lower. Additionally, it can be concluded that as the drag coefficient was already a relatively small number compared to the lift coefficient, the modelling error in the drag coefficient resulted in the unrealistic lift to drag ratios reported for the CFD simulations.

Although not required by the questions in this project, running the CFD analysis at a range of angle of attacks allowed the observation of how the velocity magnitudes and pressure differentials changed as a function of the angle of attack. Comparing Figures 1, 3, and 5, it made logical sense that as the pressure differential increased (as seen by larger changes in color) the lift coefficient increased. This observation is also highlighted in the comparison of Figures 2, 4, and 6 with respect to velocity magnitude (again seen by larger changes in color). Finally, as the input velocity vector changed direction it was noted that the point of lowest velocity/highest pressure moved along the surface of the airfoil to remain perpendicular to the input velocity vector.

2.4 Wing Design

In order to determine the require chord length and span of each wing, an aspect ratio of 7.5 was assumed and the required lifting force at cruising velocity was calculated using Newton's second law.

- Free body diagram for airplane at cruising velocity



Cruising: $F_W = F_L$

$1{,}000 \text{ kg} * 9.81 \text{ m/s}^2 = F_L$

$F_L = 9810 \text{ N}$ or 9.81 kN

- Calculate the chord and span of the wing

$$F_L = \frac{1}{2}\rho v^2 A c_L$$

From Airfoiltools.com at an angle of attack of five degrees: $C_L = 0.75$ $C_D = 0.008$

$$0.5(9810\text{N}) = 0.5(1.225 \text{ kg/m}^3)(53.611 \text{ m/s})^2(c * 7.5c)(0.75)$$

$$4905 \text{ N} = 9902.3081c^2 \text{ N}$$

$$\therefore c = 0.7038\text{m}$$

$$\therefore s = 5.2785\text{m}$$

2.5 Drag Estimation

In order to determine the drag opposing the motion of the airplane, each section of the proposed airplane design was analyzed against both pressure and friction drag.

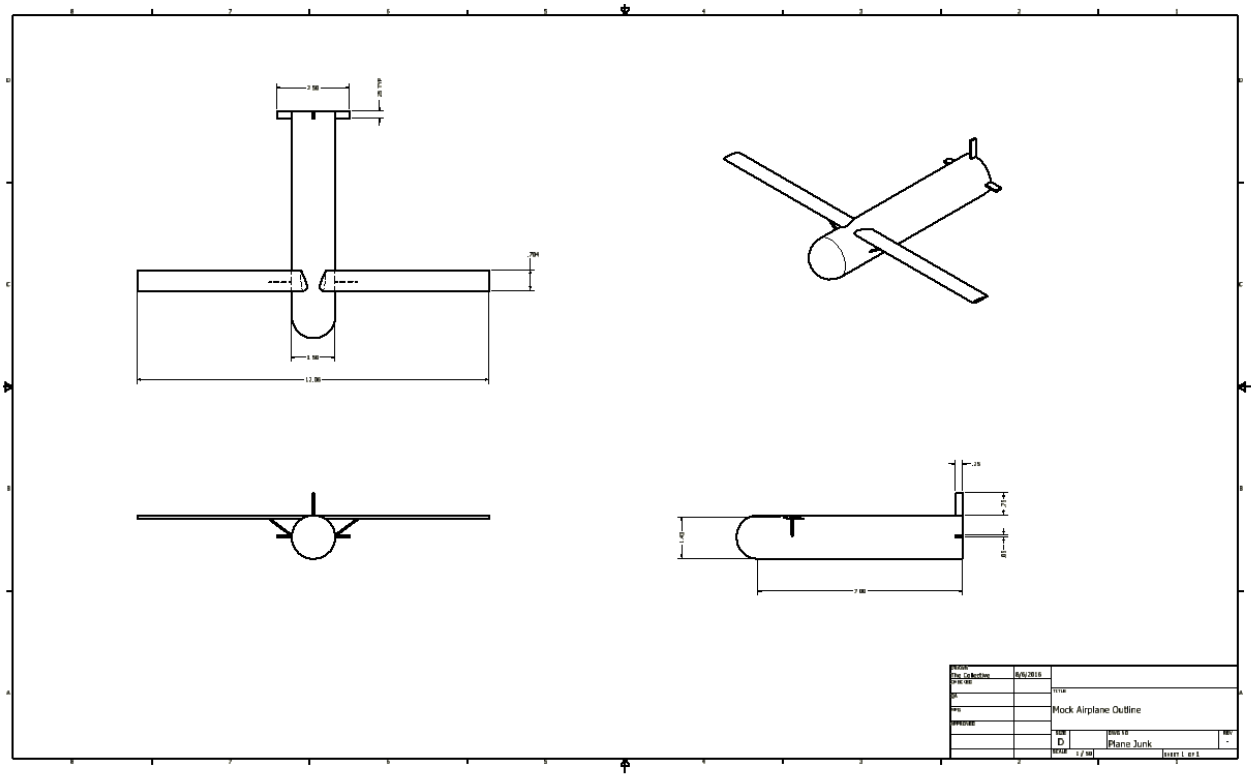


Figure 7. Assembly Outline of Proposed Airplane Design

- Hemispherical nose analysis

$$A = \pi(0.75\text{m})^2 = 1.7671 \text{ m}^2$$

$$F_D = \frac{1}{2}\rho v^2 A c_D = \frac{1}{2}(1.225 \text{ kg/m}^3)(53.611 \text{ m/s})^2(1.7671 \text{ m}^2)(0.38)$$

$$F_{D\ Nose} = 1182.1427 \text{ N (pressure)}$$

- Cylindrical body analysis

$$Re_L = \frac{\frac{53.611m}{s} * 7m}{1.5e{-5}\ m^2/s} = 2.502e7$$

$$C_D = \frac{0.455}{(\log Re_L)^{2.58}} - \frac{1610}{Re_L} = 0.0025$$

$$A = \pi(7\text{m})(1.5\text{m}) = 32.9867 \text{ m}^2$$

$$F_D = \frac{1}{2}\rho v^2 A c_D = \frac{1}{2}(1.225 \text{ kg/m}^3)(53.611 \text{ m/s})^2(32.9867 \text{ m}^2)(0.0025)$$

$$F_{D\ Body} = 147.4833 \text{ N (friction)}$$

- Side tail wing analysis

$$Re_L = \frac{\frac{53.611m}{s} * 0.25m}{1.5e{-5}\ m^2/s} = 8.935e5$$

$$C_D = \frac{0.455}{(\log Re_L)^{2.58}} - \frac{1610}{Re_L} = 0.0028$$

$$A = (0.25m)(0.5m) = 0.125\ m^2$$

$$F_D = \frac{1}{2}\rho v^2 A c_D = \frac{1}{2}(1.225\ kg/m^3)(53.611\ m/s)^2(0.125\ m^2)(0.0028)$$

$$F_{D\ Side\ Tail} = 0.6083\ N\ \ (friction)$$

$$b/h = (0.5m\ /\ 0.05m) = 10\ \ therefore\ C_D \approx 1.25$$

$$F_D = \frac{1}{2}\rho v^2 A c_D = \frac{1}{2}(1.225\ kg/m^3)(53.611\ m/s)^2(0.125\ m^2)(1.25)$$

$$F_{D\ Side\ Tail} = 55.0128\ N\ \ (pressure)$$

- Up tail wing analysis

$$Re_L = \frac{\frac{53.611m}{s} * 0.25m}{1.5e{-5}\ m^2/s} = 8.935e5$$

$$C_D = \frac{0.455}{(\log Re_L)^{2.58}} - \frac{1610}{Re_L} = 0.0028$$

$$A = (0.25m)(0.75m) = 0.1875\ m^2$$

$$F_D = \frac{1}{2}\rho v^2 A c_D = \frac{1}{2}(1.225\ kg/m^3)(53.611\ m/s)^2(0.1875\ m^2)(0.0028)$$

$$F_{D\ Up\ Tail} = 0.9124\ N\ \ (friction)$$

$$b/h = (0.75m\ /\ 0.05m) = 15\ \ therefore\ C_D \approx 1.4$$

$$F_D = \frac{1}{2}\rho v^2 A c_D = \frac{1}{2}(1.225\ kg/m^3)(53.611\ m/s)^2(0.1875\ m^2)(1.4)$$

$$F_{D\ Up\ Tail} = 92.4215\ N\ \ (pressure)$$

- Wing analysis

$$A = (0.7038m)(5.2785m) = 3.7150\ m^2$$

$$F_D = \frac{1}{2}\rho v^2 A c_D = \frac{1}{2}(1.225\ kg/m^3)(53.611\ m/s)^2(3.7150\ m^2)(0.008)$$

$$F_{D\ Wing} = 52.32\ N\ \ (friction)$$

$F_{D\ Wing}$ = 0 N (pressure – assumed zero as an airfoil is a streamlined surface)

- Total drag analysis

$$F_{D\ Totsl} = (147.4833N + 2(0.6083N) + 0.9124N + 2(52.32)N) +$$
$$(1182.1427N + 2(55.0128N) + 92.4215N)$$

$$F_{D\ Total} = 1638.8421\ N$$

2.6 Power Requirement and Propulsion Estimate

- Power requirement

$$P = FD\ Total * v = 1638.8421N * 53.611\ m/s = 87.86\ kW\ \text{ or }\ 117.82hp$$

- Propulsion and Fuel Consumption

  In order to propel the proposed airplane design a 130hp D-Motor LF39 gas engine has been selected that will also use the UL-1900 x 1100 propeller. The following assumptions and calculations were utilized to determine the final fuel consumption:

Average gasoline engine efficiency = 30%

Heating value of gasoline = 47.0 kJ/g

Propeller efficiency rule of thumb: $J = \frac{V}{ND} = \frac{53.611\,m/s}{(2800\,rev/s)(1.9\,m)} = 0.6046$

$\therefore$ Propeller efficiency $\approx$ 0.79 (See Appendix B "Propeller Efficiency")

Total efficiency = Engine efficiency * Propeller efficiency = 0.237

$P_{required} = P_{calculated}$ / Total efficiency = 87.86 kW / 0.237 = 370.72 kW

$\dot{m} = P_{reqiured}$ / Heating value = 370.72 kW / 47kJ/g = 0.007888 kg/s

$Q = \dot{m}$ / density of gasoline = 0.007888 kg/s / 719.7 kg/m^3 = 0.000001096 m^3/s

$\therefore$ Fuel consumption = 10.42 gallons/hour

3. Post-Project Analysis

3.1 Experience with ANSYS Workbench and FLUENT

This project served as a nice introduction into the applications of ANSYS Workbench and FLUENT. As our team chose to run the CFD simulations against a number of angle of attacks, we did become quite proficient at the basic program functions. However, due to ANSYS Workbench's inability to properly connect to and open FLUENT, we did not fully utilize this program's capabilities. Overall, our group enjoyed and appreciated this opportunity to work with a new software package.

3.2 General Conclusions

After completing this design project, it was concluded that our plane design would need revised before a more serious proposal would be considered. For example, the Cessna 170B illustrated a more feasible design concerning body proportions and streamlining considerations. However, for simplification of the drag estimates the approach shown is acceptable. The importance of modelling the flow correctly in CFD was heavily emphasized by the illogical drag ratios.

3.3 Team Contributions

Our team embraced a collaborative approach to this project and made a concerted effort to ensure that all members were involved, especially during the actual CFD simulations. At times, certain calculations were divided in order to increase group efficiency, but all members reviewed the results before submission.

APPENDICES

A – SCANNED HAND CALCULATIONS

#1

- Small aircraft → 4 person

4·200 lbs → 4(90.7185 kg) ≅ 360 kg
+ 640 kg --- plane
∴ 1000 kg

Weight -- 1000 kg

Angle of attack optimization

$@\ \alpha = 0° \longrightarrow C_{D_{net}} = 0.0009644\,7976$

*chart ratio – 50 $\left(\frac{.25}{.005}\right)$

*lift ratio – 21.75 $\left(\frac{.261}{.012}\right)$

$C_{L_{net}} = 0.20769527$

ratio = 215.3443

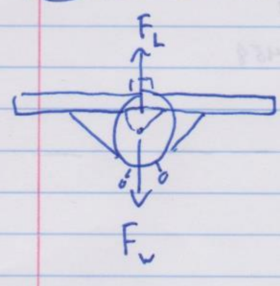
Goal --> highest lift to drag ratio $(\uparrow C_L \ \downarrow C_D)$

$$C_D = \frac{F_D}{\frac{1}{2}\rho V^2 A} \qquad C_L = \frac{F_L}{\frac{1}{2}\rho V^2 A}$$

$$\frac{C_L}{C_D} = \frac{\dfrac{F_L}{\cancel{\frac{1}{2}\rho V^2 A}}}{\dfrac{F_D}{\cancel{\frac{1}{2}\rho V^2 A}}} = \frac{F_L}{F_D}$$

$$Re = \frac{VD}{\nu} = \frac{53.611\,m/s \cdot (1.5\,m)_{ref}}{1.5 \cdot 10^{-5}\,m^2/s} = \boxed{5.3611 \cdot 10^6\ Re}$$

→ Table A-10 @ 20°C



Cruise then $F_L = F_w$

$$1000\,kg \cdot 9.81\,m/s^2 = F_L$$

$$F_L = 9810\ N$$

or $\boxed{9.81\ kN}$

*Needed at cruising

#12

www.skytamer.com/Cessna_170.html

→ chord length actual ≈ 1.6m

wikipedia

→ wing: 11m

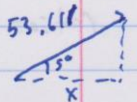
aspect = 7.46

∴ chord = 1.5m

@ $\alpha = 5°$

$C_{D_{net}} = 0.003935708$

ratio = 150.6638



$C_{L_{net}} = 0.59296891$

$X = 53.40699$ m/s

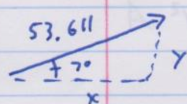
$Y = 4.6725$ m/s

$1\cdot10^6$ … * chart ratio – $\frac{.75}{.008} = 93.75$

$1\cdot10^5$ … * lit ratio – $\frac{.733}{.0148} = 49.527$

@ $\alpha = 7°$

$C_{D_{net}} =$



$C_{L_{net}} =$

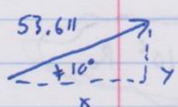
$X = 53.21139$ m/s

$Y = 6.5335$ m/s

@ $\alpha = 10°$

$C_{D_{net}} = .012827562$

ratio = 99.5745



$C_{L_{net}} = 1.2772977$

$X = 52.7965$ m/s

* chart ratio – $\frac{1.25}{.018} = 69.44$

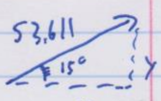
$Y = 9.3095$ m/s

* lit ratio – $\frac{1.139}{.024} = 47.458$

@ $\alpha = 15°$

$C_{D_{net}} =$ X

$C_{L_{net}} =$ X



*

$x = 51.7892$ m/s

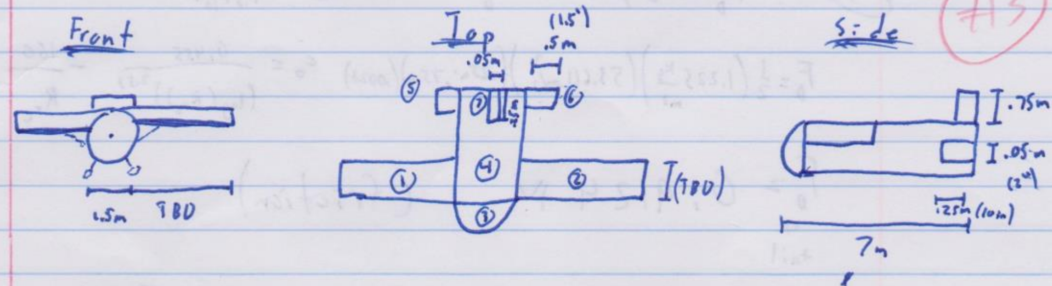
$y = 13.8755$ m/s

$\rightarrow$ ① $= C_D = .38$ (Table 9.3)

$C_D = \dfrac{F_D}{\frac{1}{2}\rho v^2 A}$ $\quad P = F_D \cdot v$ $\quad C_L = \dfrac{F_L}{\frac{1}{2}\rho v^2 A}$

Fuselage considerations

#13



(1+2) = return w/ CFD

(3) $\rightarrow$ $A = \pi r^2 = \pi\left(\frac{1.5}{2}\right)^2 = 1.7671\,m^2$

$$F_D = \frac{1}{2}\rho v^2 A C_D = \frac{1}{2}\left(1.225\,\frac{kg}{m^3}\right)\left(53.611\,m/s\right)^2\left(1.7671\,m^2\right)(.38)$$

$$F_{D_{nose}} = \cancel{23.0509}\ 1182.1427\ N \quad (\text{pressure})$$

(4) $F_D = \frac{1}{2}\rho v^2 A C_D$ $\quad$ 9.6 C_D chart $C_D \approx \dfrac{0.455}{(\log Re_L)^{2.58}} - \dfrac{1610}{Re_L}$ $\quad Re_L = \dfrac{53.611 \cdot 7}{1.5 \cdot 10^{-5}} \rightarrow 2.502 \cdot 10^{7}$

$$F_D = \frac{1}{2}\left(1.225\,\frac{kg}{m^3}\right)\left(53.611\,m/s\right)^2\left(7m \cdot \pi \cdot 1.5m\right)(.0025)$$

$$F_{D_{body}} = 147.4833\ N \quad (\text{friction})$$

#2* (5+6) $F_D = \frac{1}{2}\rho v^2 A C_D$ $\quad C_D = \dfrac{0.455}{(\log Re_L)^{2.58}} - \dfrac{1610}{Re_L}$ $\quad Re_L = \dfrac{53.611 \cdot .25}{1.5 \cdot 10^{-5}} \rightarrow 8.935 \cdot 10^{5}$

$$F_D = \frac{1}{2}\left(1.225\,\frac{kg}{m^3}\right)\left(53.611\,m/s\right)^2\left(.25m \cdot .5m\right)(.0028)$$

$$F_{D_{tail\,side}} = 0.6083\ N \quad (\text{friction})$$

$$F_D = \frac{1}{2}\left(1.225\,\frac{kg}{m^3}\right)\left(53.611\,m/s\right)^2\left(.5m \cdot .05m\right)(1.25) \qquad b/h = .5/.05 = 10 \quad \therefore C_D \approx 1.25$$

$$F_{D_{tail\,side}} = 55.0128\ N \quad (\text{pressure})$$

#4

7) $F_D = \frac{1}{2}\rho V^2 A c_D$ $R_{e_L} = \frac{53.611 \cdot .25}{1.5\cdot 10^{-5}} = 8.935\cdot 10^5$

$F_D = \frac{1}{2}(1.225\,\frac{kg}{m^3})(53.611\,m/s)^2(.25m\cdot .75m)(.0028)$ $c_D = \frac{0.455}{(\log(R_{e_L}))^{2.58}} - \frac{1610}{R_{e_L}} = .0028$

$F_{D_{vp\,tail}} = 0.9124\ N$ (friction)

$F_D = \frac{1}{2}\rho V^2 A c_D$ (*) $b/h = .05/.75 = .0667$

$F_D = \frac{1}{2}(1.225\,\frac{kg}{m^3})(53.611\,m/s)^2(.05\cdot .75)(1.4)$ $.75/.05 = 15$

$F_{D_{vp\,tail}} = 92.4215\ N$ (pressure) $\therefore c_d = 1.4$

Friction Pressure

$F_{D_{total}} = (147.4833N + 2(.6083N) + 0.9124N) + (1182.1427N + 2(55.0128N) + 92.4215N)$

$F_{D_{total}}$ (minus wings) $= 149.6123\ N + 1384.5898 = \boxed{1534.2021\ N}$

(1+2) $F_L = \frac{1}{2}\rho V^2 A c_L$ $s = 7.5c$ $\therefore 7.5c^2$

from chart lit
$c_L = 0.75$
$c_D = 0.008$
@ $\alpha = 5°$

$\frac{1}{2}(9810N) = \frac{1}{2}(1.225\,\frac{kg}{m^3})(53.611\,m/s)^2(c\cdot s)(.75)$

$4905 = 4902.3081\,c^2$ $\therefore 10.55m$ for wing total

$\therefore c = 0.7038m$ $\therefore$ span each wing $= 5.2785m$

#15

1+2

from chart lit

$$F_D = \frac{1}{2} \rho v^2 A c_D$$

$$F_{D_w} = \frac{1}{2}\left(1.225 \frac{kg}{m^3}\right)\left(53.611 m/s^2\right)\left(0.7038m \cdot 5.2785m\right)\left(0.008\right)$$

$$F_{D_w} = 52.32 \text{ N} \quad \text{(friction -- each)}$$

$$* \; F_{D_w} = 0 \text{ N} \quad \text{(pressure -- streamline assumption)}$$

$$F_{D_{total}} \text{ (w/ wings)} = 1534.2021 \text{ N} + 2(52.32 \text{ N}) = 1638.8421 \text{ N}$$

$$P = F_D v = 1638.8421 \text{N} \cdot 53.611 \text{ m/s}$$

$$P = 87{,}859.96382 \frac{Nm}{s} \text{ --- W}$$

or

87.86 KW

$$KW \cdot hp \rightsquigarrow 1 hp = 745.7 W$$

* use propeller propulsion

$$\therefore 117.82 \text{ hp}$$

#16

Power = 87.86 kW   Gasoline → 47.0 $\frac{kJ}{g}$

Average engine efficiency = 30%

Rule of thumb $J = \frac{V}{ND} = 0.6$ → efficiency of rotor = 79%

Total efficiency = $0.3 \cdot 0.79 = \eta = 0.237$

$P_{req} = 87.86\ kW / \eta$ $\frac{370.72\ kW}{47.0\ \frac{kJ}{g}} = 7.888\ \frac{g}{s} = 0.007888\ \frac{kg}{s}$

$P_{req} = 370.72\ kW$

density of gasoline = 719.7 $\frac{kg}{m^3}$

$\frac{0.007888\ \frac{kg}{s}}{719.7\ \frac{kg}{m^3}} = 0.000001096\ \frac{m^3}{s}$

convert to gallons per hour

10.42 $\frac{gal}{hr}$

B – EXTERNAL REFERENCES

8/3/2016 NACA 2412 (naca2412-il)

Airfoil Tools

Search 1636 airfoils Tweet Like 960

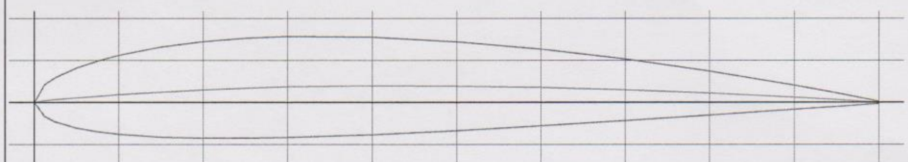
You have 0 airfoils loaded.
Your Reynold number range is 50,000 to 1,000,000. (**set**)

Search

#1

NACA 2412 (naca2412-il)

NACA 2412 - NACA 2412 airfoil



Details

(naca2412-il) NACA 2412
NACA 2412 airfoil
Max thickness 12% at 30% chord.
Max camber 2% at 40% chord
Source **UIUC Airfoil Coordinates Database**
Source dat file
The dat file is in Selig format

Dat file

```
NACA 2412
1.0000     0.0013
0.9500     0.0114
0.9000     0.0208
0.8000     0.0375
0.7000     0.0518
0.6000     0.0636
0.5000     0.0724
```

Parser

No parser warnings

Send to airfoil plotter
Add to comparison
Lednicer format dat file
Selig format dat file

Similar airfoils

E207 (12.04%)	Preview	Details
E220 (11.48%)	Preview	Details
S8055 (12%)	Preview	Details
NACA CYH	Preview	Details
RAF 38 AIRFOIL	Preview	Details
LDS-2 AIRFOIL	Preview	Details
MH 120 11.57%	Preview	Details
GOE 704 AIRFOIL	Preview	Details
ONERA OA212 AIRFOIL	Preview	Details
NACA 1412	Preview	Details

Polars for NACA 2412 (naca2412-il)

Plot	Airfoil	Reynolds #	Ncrit	Max Cl/Cd	Description	Source	
☐	naca2412-il	50,000	9	32.5 at α=7.25°	Mach=0 Ncrit=9	Xfoil prediction	Details
☐	naca2412-il	50,000	5	34.6 at α=6.5°	Mach=0 Ncrit=5	Xfoil prediction	Details
☐	naca2412-il	100,000	9	50 at α=6.75°	Mach=0 Ncrit=9	Xfoil prediction	Details
☐	naca2412-il	100,000	5	49.4 at α=6°	Mach=0 Ncrit=5	Xfoil prediction	Details
☐	naca2412-il	200,000	9	66.6 at α=6°	Mach=0 Ncrit=9	Xfoil prediction	Details
☐	naca2412-il	200,000	5	62.6 at α=5.25°	Mach=0 Ncrit=5	Xfoil prediction	Details
☐	naca2412-il	500,000	9	87.3 at α=5°	Mach=0 Ncrit=9	Xfoil prediction	Details
☐	naca2412-il	500,000	5	78.3 at α=4°	Mach=0 Ncrit=5	Xfoil prediction	Details
☑	naca2412-il	1,000,000	9	101.4 at α=4.5°	Mach=0 Ncrit=9	Xfoil prediction	Details
☑	naca2412-il	1,000,000	5	87 at α=4.5°	Mach=0 Ncrit=5	Xfoil prediction	Details

Update plots **Reynolds number calculator**

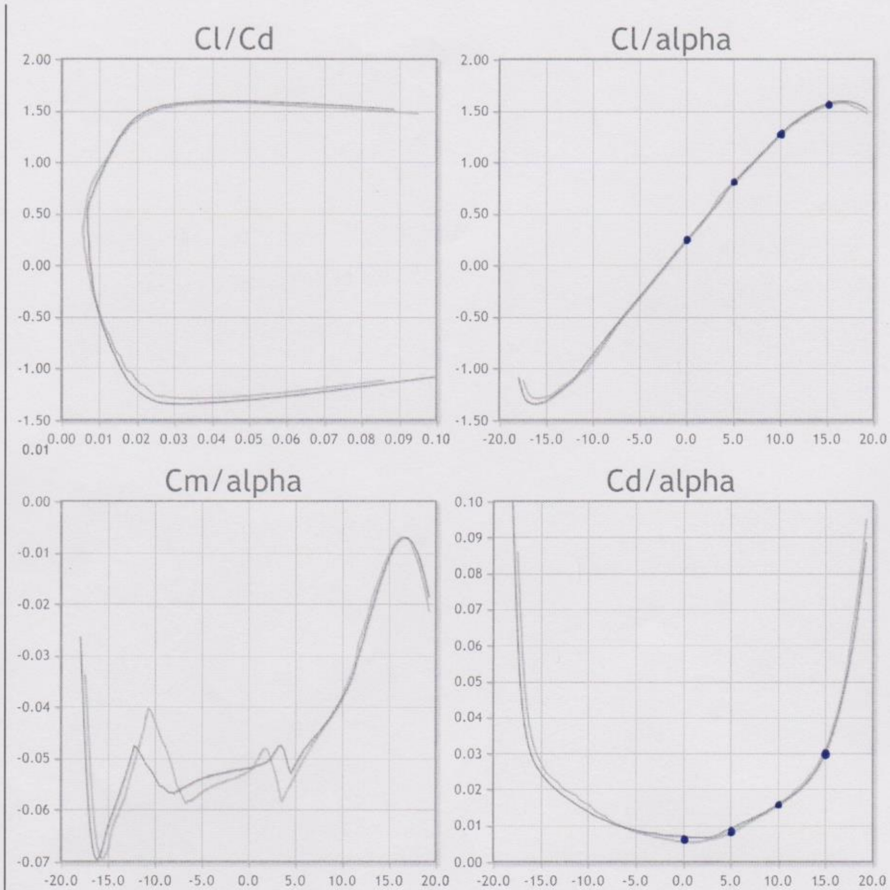
Set Reynolds number and Ncrit range

Update Range		Low	High
	Reynolds Number	50,000	1,000,000
	NCrit	7	9

http://airfoiltools.com/airfoil/details?airfoil=naca2412-il 1/2

8/3/2016 NACA 2412 (naca2412-il)

#12



AdChoices ▶ DAT File ▶ Airfoil ▶ Database Tools ▶ NACA

http://airfoiltools.com/airfoil/details?airfoil=naca2412-il 2/2

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#1

Design of NACA 2412 and its Analysis at Different Angle of Attacks, Reynolds Numbers, and a wind tunnel test

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Abstract—The purpose of this project is to analyze airfoil at different Reynolds numbers using Gambit and Fluent, and wind tunnel experiment. One model is prepared for wind tunnel analysis and 2D and 3D models are created and drawn in solid work and they were meshed in Gambit using geometry data gathered by Airfoil database available on internet. These models were read into Fluent where flow boundary conditions were applied and the discretized Navier-Stokes equations were solved numerically. Tests also run in wind tunnel to find out the general aerodynamic characteristics of the Airfoil (NACA 2412).

Keywords— airfoil, NACA 2412, analysis of airfoil, design of airfoil, 3D analysis of airfoil, four digit airfoil, angle of attacks

INTRODUCTION

In this project, computational Fluid Mechanist analysis of airfoil has been done to understand the aerodynamic airfoil concepts

Airfoil taken is NACA 2412, this is cambered airfoil belongs to the four digit series of the NACA airfoil classification, the general characteristics of this airfoil are:-

NACA FOUR DIGIT SERIES

The NACA four-digit wing sections define the profile by:

1. First digit describing maximum camber as percentage of the chord.
2. Second digit describing the distance of maximum camber from the airfoil leading edge in tens of percents of the chord.
3. Last two digits describing maximum thickness of the airfoil as percent of the chord.

NACA 2412 is the airfoil of NACA 4 digit series. From its designation we get the NACA 2412 airfoil has a maximum camber of 2% located 40% (0.4 chords) from the leading edge with a maximum thickness of 12% of the chord. Four-digit series airfoils by default have maximum thickness at 30% of the chord (0.3 chords) from the leading edge. NACA 2412 is slow speed airfoil; this airfoil is used in single engine Cessna 152, 172 and 182 airplanes

SOME PARAMETERS

Reynolds number

The Reynolds number relates the density, viscosity, speed and size of typical flow in a dimensionless equation which is involve in many fluid dynamics problems. This dimensionless numbers or combination appears in many cases related to the fact that laminar flow can be seen or turbulent. From a mathematical point of view the Reynolds number of a problem or situation is defined by the following equation.[3]

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H2

$$Re = (\rho \times V \times L)/\mu$$

Table No.1/ Aerodynamic forces		
For lift coefficient	For Drag Coefficient	For moment coefficient
$C_L = 2 f(Re, M, \alpha)$	$C_D = 2 f(Re, M, \alpha)$	$C_M = 2 f(Re, M, \alpha)$
$L = C_L \frac{1}{2} \rho V^2 c$	$D = C_D \frac{1}{2} \rho V^2 c$	$M = C_M \frac{1}{2} \rho V^2 C^2$

PROCESS OF AIRFOIL DESIGN

Coordinates of NACA 2412 is taken from Javafoil software and its Reynolds no. characteristics are also taken [11]

Table No. 2/ Coordinates of NACA 2412

Upper surface			lower surface		
1	**0**	**0**	**0**	**0**	**0**
0.989259	0.002267	0	0.012606	-0.01662	0
0.957222	0.008773	0	0.04613	-0.02921	0
0.905298	0.018704	0	0.098928	-0.03756	0
0.835653	0.030889	0	0.168624	-0.04171	0
0.751234	0.043993	0	0.25226	-0.0421	0
0.655658	0.056642	0	0.346406	-0.03963	0
0.553071	0.067493	0	0.447493	-0.03544	0
0.447978	0.075277	0	0.551457	-0.02982	0
0.344577	0.078639	0	0.653359	-0.02351	0
0.24774	0.076012	0	0.748766	-0.01728	0
0.162245	0.067489	0	0.833478	-0.01161	0
0.092055	0.054036	0	0.903719	-0.00681	0
0.040324	0.037207	0	0.956323	-0.00313	0
0.009246	0.01873	0	0.988889	-0.0008	0
0	0	0	1	0	0

NACA 2412 airfoil is analyzed on JAVA FOIL. JAVAFOIL is the analysis software which gives analysis data of various airfoils its coordinates, parameters for various Reynolds number, coefficient of lift and drag graphs, coefficient of moment and angle of attack graphs etc.

Modeling of airfoil

The airfoil model is easily designed in solid work. In order to do that airfoil coordinates are plotted and the airfoil 3D model is created.

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#13

Figure No. 1/ Design of airfoil on solidwork

Gambit is meshing software that is capable of creating meshed geometries that can be read into Fluent and other analysis software. Making a meshed file, it is done in both 2D and 3D these files are imported in fluent.

We have done meshing of Airfoil NACA 2412 and of its domain and then the simulation of flow variables over this control volume is done in case of 2D of Control Line.

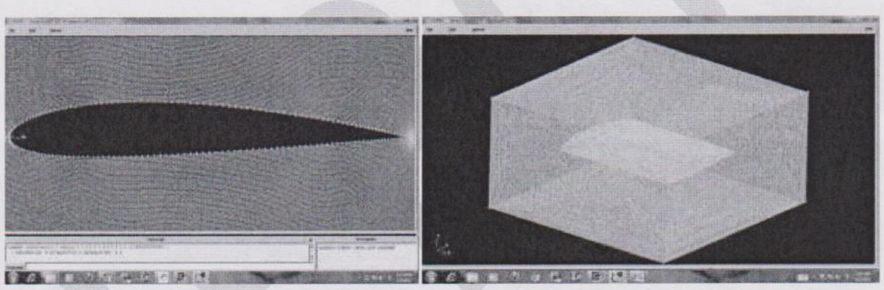

Figure No. 2/ 2D meshing of Airfoil 2412 Figure No. 3/ 3D meshing of Airfoil 2412

The desired mesh can now be read into FLUENT which will then run the geometry through the numerical analysis. Different angles of attack will be analyzed in FLUENT 6.3.26. Airfoil and angle of attacks 4, 8, and 12 degrees are analyzed. Fluent gives results.

#14

2D ANALYSIS DATA

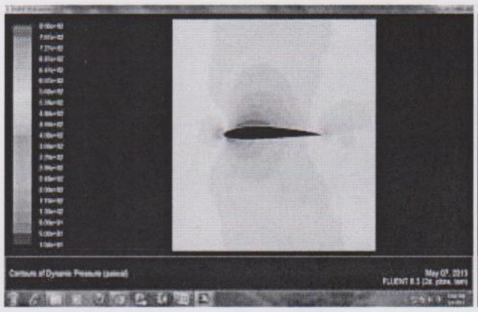
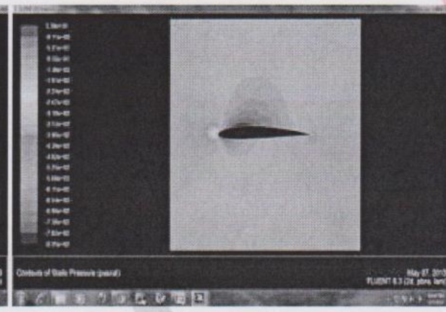

Figure No. 5/Contour of dynamic pressure   Figure No. 6/ Contour of static pressure

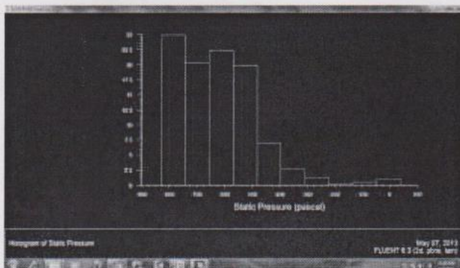


Figure No. 7/ Variation of static pressure   Figure No. 8/ Contours of velocity magnitudes

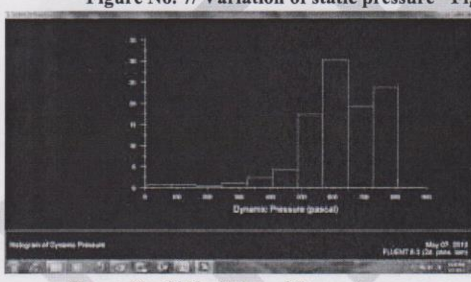
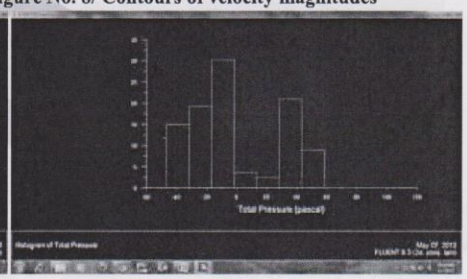

Figure No. 9/ Variation of Dynamic pressure   Figure No. 10/ Variation of Total pressure

3D ANALYSIS RESULTS

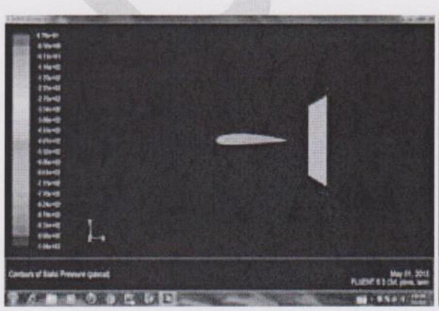
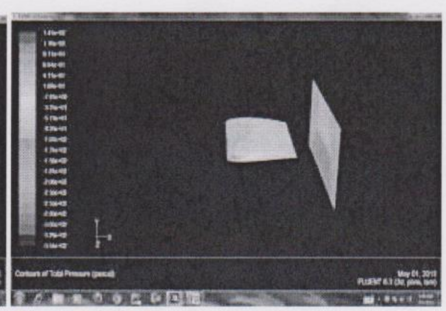

Figure No. 11/ Contour of total pressure   Figure No. 12/ Contour of static pressure

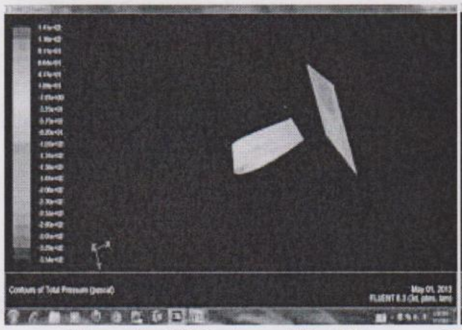
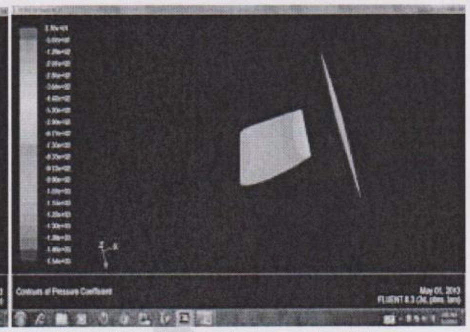

Figure No. 13/ Contour of Total Pressure **Figure No. 14/ Contour of Pressure Coefficient**

Results at different angle of attacks and at different Reynolds numbers For this three Reynolds numbers are chosen 60000, 100000, and 140000 with four different angle of attacks 0, 4, 8, and 12 degrees. (Table No. 4, 5, & 6)

Table No.3/Velocity characteristics

α	Cl	Cd	Cm 0.25	Cp*	M cr.
[°]	[-]	[-]	[-]	[-]	[-]
0	0.261	0.01197	-0.051	-0.572	0.77
4	0.733	0.01483	-0.055	-1.458	0.603
8	1.139	0.02418	-0.059	-3.935	0.418
12	1.144	0.09473	-0.029	-7.403	0.318

Table No. 4/ **Coefficient of lift, drag and moments at different angles of attack, at Reynolds number 60000.**

A	Cl	Cd	Cm 0.25
[°]	[-]	[-]	[-]
0	0.261	0.01532	-0.05
4	0.73	0.01841	-0.055
8	1.128	0.02794	-0.059
12	1.142	0.10236	-0.027

#16

Table No. 5/ **Coefficient of lift, drag and moment at different angles of attack, at Reynolds number 140000.**

A	Cl	Cd	Cm 0.25
[°]	[-]	[-]	[-]
0	0.261	0.01126	-0.051
4	0.734	0.0131	-0.055
8	1.143	0.02226	-0.06
12	1.146	0.0905	-0.031

Table No. 6/ **Coefficient of lift, drag, pressure and moments at different angles of attack, at Reynolds number 100000.**

α	Cl	Cd	Cm 0.25
[°]	[-]	[-]	[-]
0	0.261	0.01197	-0.051
4	0.733	0.01483	-0.055
8	1.139	0.02418	-0.059
12	1.144	0.09473	-0.029

After a century of theoretical research on the subject of airfoil and wing theory, the final word on the performance of an airfoil must still come from wind tunnel testing. The reason for this state of affairs is that the flow field about a wing is extremely complicated. The simplifying assumptions that are frequently introduced in order to treat the problem theoretically are much too severe to fail to influence the final results. Many of these assumptions ignore the effects of viscosity, nonlinearities in the equations of motion, three-dimensional effects, non steady flow, free stream turbulence, and wing surface roughness. Nevertheless the theoretical prediction of lift produced by a wing has been reasonably successful (not quite so true for drag) and serves as an effective basis with which to study the experimental results.

RESULTS OF THE 3D ANALYSIS DATA

1) Static pressure varies form -4.97e+02 to -1.03e+03 Pascal from trailing to leading edge.
2) Dynamic pressure at upper most part and lower most part is of order 8.06e + 02 Pascal while at leading edge it is of order 1.70e+02 and at trailing edge it is of order 4.53e+02.
3) Total pressure is maximum at the leading edge 1.41e+02 Pascal and decreases along the length.
4) Coefficient of pressure is maximum at leading edge and trailing edge while lower at thick surfaces.
5) Absolute pressure is also higher at leading and trailing edge while it has smaller values at thick surfaces of order 1.01e + 05 Pascal.
6) Velocity magnitude is seems to be constant over the whole airfoil surface 1.81e m/s.
7) X-Velocity is constant.
8) Y-Velocity is nearly constant -4.84e+01 m/s.
9) Z-Velocity is also nearly constant with magnitude 6.05e+02 m/s.

#7

10) Relative tangential velocity magnitude is lower at upper surface with magnitude -6.37e+00 and at lower surface it is changing from tip to end from 9.33e to 3.05e m/s.
11) Vorticity is irregularly changing at the upper surface of the airfoil while at tip and ends it is of magnitude 2.305e+02 1/s.
12) Molecular viscosity is changing irregularly different at different locations about 1.79e+05 kg-m/s.
13) Wall shear stress is maximum at few locations of the most thicken areas of the airfoil with magnitude 1.7e-01 Pascal.

RESULTS OF THE 2D ANALYSIS DATA

- Static pressure is constant at the thick surfaces of the airfoil.
- Dynamic pressure is constant at the lower ends of the airfoil.
- Density is seems to be constant with magnitude 1.23 kg-m/s.
- Velocity magnitude is also constant whether it is in x, y, or z direction

WIND TUNNEL DATA

1. Coefficient Lift coefficient is maximum at 15 X/C with magnitude 1.65. it is increasing from -15 to 15 X/C then sudden drop in Cl and from 17.5 it is constant up to 30 X/C. (Figure No. 16)

2. Drag coefficient is minimum at 0 with value0.034 and making a irregular parabolic curve. (Figure No. 15)

3. This drag polar is a irregular parabola C_d has its minimum value at 0.034 at 0.75 C_L, and C_L has its maximum value 1.68 at 0.05 to 0.055 of the C_d. (Figure No. 17)

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#18

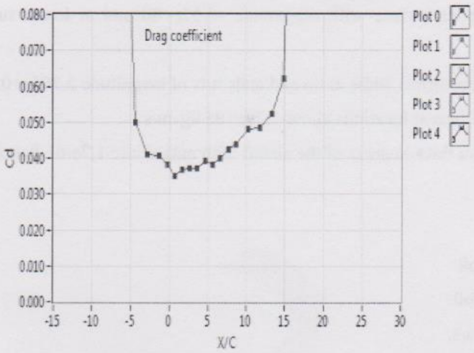

Figure No. 15/ C_d vs X/C

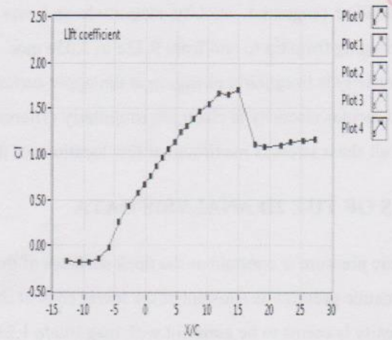

Figure No. 16/ C_L vs X/C

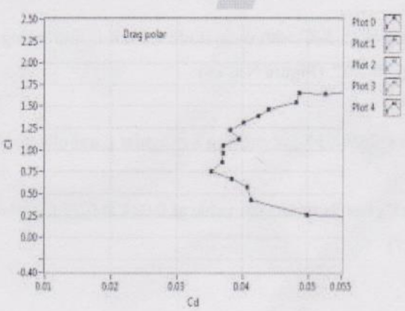

Figure No. 17/ C_L Vs C_D

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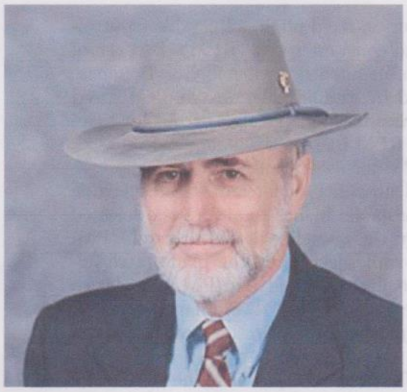
[13] ***www.airfoil****tools.com/search/index*
[14] www.aerodesign.de
[15] www.designfoil.com
[16] www.wind.nrel.gov

#1

Propeller Efficiency
Rule of Thumb



David F. Rogers, PhD, ATP

Theoretically the most efficient propeller is a large diameter, slowly turning single blade propeller. Here, think the Osprey or helicopters. In both cases, large diameter, slowly turning, compared to typical fixed wing aircraft, propellers are used. Generally, single bladed propellers are not used because of dynamic imbalance - think vibration. As a result, the general wisdom is that better propeller efficiency results from decreasing RPM. However, propeller efficiency is not only a function of RPM. It is also a function of propeller diameter and true airspeed. Generally these parameters are combined into a nondimensional parameter called the advance ratio ($J = V/ND$), where V is the true airspeed in feet per second, N is the propeller rotational speed in revolutions per second and D is the propeller diameter in feet.†

Propeller efficiency also depends on the power coefficient, which is a function of, again, N and D and also density as well as the brake horsepower. Specifically, the power coefficient, C_p, is another nondimensional parameter defined by

$$C_p = \frac{\text{BHP}}{\rho N^3 D^5}$$

where BHP is the brake horsepower and ρ (rho) is the local air density. From this, you can see that simply saying lower RPMs give better propeller efficiency is a bit simplistic.

† What is meant by a nondimensional parameter? Well, it is a parameter which, upon substituting the dimensions into the expression for each of the physical parameters, results in all the dimensions cancelling out, e.g.,

$$J = \frac{V}{ND} = V\frac{1}{N}\frac{1}{D} = \frac{\text{ft}}{\text{sec}}\,\frac{1}{\frac{\text{rev}}{\text{sec}}}\,\frac{1}{\text{ft}} = \frac{\text{ft}}{\text{sec}}\,\frac{\text{sec}}{\text{rev}}\,\frac{1}{\text{ft}} = \frac{\text{ft}}{\text{sec}}\,\frac{\text{sec}}{}\,\frac{1}{\text{ft}}$$

Because revolutions (rev) is not a physical dimension, the denominator in the second term is replaced with a blank. Finally, we have

$$J = \frac{V}{ND} = \frac{\text{ft}}{\text{sec}}\,\frac{\text{sec}}{\text{ft}} = \frac{\cancel{\text{ft}}}{\cancel{\text{sec}}}\,\frac{\cancel{\text{sec}}}{\cancel{\text{ft}}}$$

and each of the physical dimensions cancels out, i.e., J is dimensionless.

Furthermore, as with any aircraft, the designer has a design goal in mind. For the Bonanza, the design goal was high speed cruise coupled with all around good handling and performance. The design goal influences propeller design and selection.

Propellers

Typically propellers are divided into three main categories: fixed pitch, adjustable (controllable) pitch, both ground and in flight adjustable, and constant speed (RPM). Because of wartime experience, Beech originally chose a controllable pitch propeller for the Bonanza. Maximum propeller diameter is principally influenced by ground clearance and tip speed (Mach number). Bonanza propellers started at 88 inches in diameter and, except for takeoff, a maximum RPM of 2050. As maximum engine RPM increased, diameter decreased, because of tip Mach number, to 80 inches at 2700 RPM for a constant speed propeller.

The basic design philosophy for a constant speed propeller is, for any selected engine power, or torque, to change the pitch (angle) of the propeller blades to absorb the selected engine power, provided there is enough torque to turn the propeller at the selected RPM. Increasing the blade pitch increases the blade drag, while decreasing the blade pitch decreases the blade drag. Hence, a larger (coarser) blade angle, for a given RPM, will absorb more power and require more torque to turn it at the requested RPM. Similarly a smaller (finer) blade angle, for a given RPM, will absorb less power and require less torque to turn it at the requested RPM.

Propeller blades are twisted from root to tip. The amount by which the blades are twisted, along with the variation in chord, airfoil section and sweepback of the blade leading edge, are design decisions. Those design decisions are significantly influenced by the design goal. Even with a controllable pitch or constant speed propeller, optimum design throughout the flight regime is not achievable. The design goal for the Bonanza is high speed cruise. Hence, propeller design and selection is optimum or near optimum for high speed flight. Thus, performance for takeoff and climb is suboptimal.

A Simplified Rule of Thumb

From the propeller maps for both the common two and three blade McCauley propellers fitted to the later Bonanzas, a *generic* propeller efficiency curve, η, (eta) as a function of the advance ratio, J, can be estimated as shown in Figure 1. For full power and 2700 RPM at sea level and for 65% power at 6000 ft at 2300 RPM, the maximum propeller efficiency occurs for between $J = 0.95\pm$ and $J = 1.05\pm$ as shown by the gray band in Figure 1.

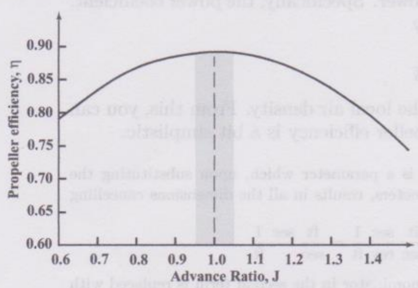


Figure 1. Generic propeller efficiency vs advance ratio.

Outside of these values the propeller efficiency decreases. From this, we can see that, for a fixed diameter propeller, it is the ratio of the true airspeed to the propeller RPM that is important in achieving maximum efficiency.

For this generic propeller efficiency curve the ratio of RPM/TAS should be maintained at approximately 15 to give an advance ratio of $J = 1.0\pm$, i.e.,

$$J = \frac{V}{ND} = \frac{1.69\,\mathrm{KTAS}}{(\mathrm{RPM}/60)(\mathrm{D}/12)} = \frac{(60)\,(1.69)\,(12)\,\mathrm{KTAS}}{(\mathrm{RPM})\,(\mathrm{D})} = 1.0$$

where the 1.69 converts KTAS (knots true airspeed) to ft/sec TAS, the 60 converts RPM to RPS (revolutions per second) and the 12 converts inches to feet. For a propeller diameter of 80 inches, after rearranging and inverting this equation, we have

$$\frac{RPM}{KTAS} = 15.2$$

as a rule of thumb to maintain maximum propeller efficiency. However, this is a rule of thumb so let's use 15.0 for A Simplified Rule Of Thumb (ASROT).† It is easier to remember and close enough.

Fine Tuning the Rule of Thumb

The propeller efficiency curve shown in Figure 1 is a composite of both the typical Bonanza two and three blade propellers at two different conditions. Is there a way to fine tune the rule of thumb for a specific aircraft and propeller? Because the Bonanza design goal was high speed cruise, it is reasonable to assume that the factory propeller is optimized for that condition.

As a practical matter, for a normally aspirated engine the cruise true airspeed increases with altitude until the 'critical' altitude for a given power setting is reached. Let's call the critical altitude the 'knee' in the curve. Above the knee the engine can no longer produce the requested percentage of power. High speed cruise conditions for various altitudes and power settings are represented by the altitude vs cruise airspeed graph (see Figure 2) in the performance section of the Pilot Operating Handbook (POH). As examples to test the ASROT let's use the knee in the altitude vs cruise true airspeed graph from the POH.

Table 1 Cruise Airspeed Efficiencies

Altitude	%BHP	BHP	RPM	KTAS	J	C_p	η	Symbol	ASROT
6000 ft	75	213.8	2500	173	1.059	0.0621	0.902	Red dot	14.5
7500 ft	65	185.3	2300	164	1.085	0.0724	0.905	Blue dot	14.0
8700 ft	55	156.8	2100	152	1.101	0.0722	0.905	Green dot	13.8
14000 ft	45	128.3	2100	141	1.021	0.0807	0.896	Black dot	14.9
11000 ft	45	128.3	2100	140	1.010	0.0722	0.900	Red circle	15.0

Weight = 3100 lbs, standard day

Table 1 illustrates the results for the power settings, altitudes and airspeeds corresponding to the knee in the cruise true airspeeds vs altitude curves represented by the dashed line in Figure 2 obtained using the bare propeller efficiency map‡ for the C76 McCauley 3-blade 80 inch propeller, as shown in Figure 3. Table 1 shows that the advance ratio, J, lies on the plus side of the 0.95 to 1.05 maximum efficiency curve at the higher true airspeeds and

† Pronounced AS ROT

‡ Bare propeller efficiencies do not consider blockage effects of the nose or nacelles. For a properly designed propeller blockage effects can *decrease* propeller efficiencies by 1-3%.

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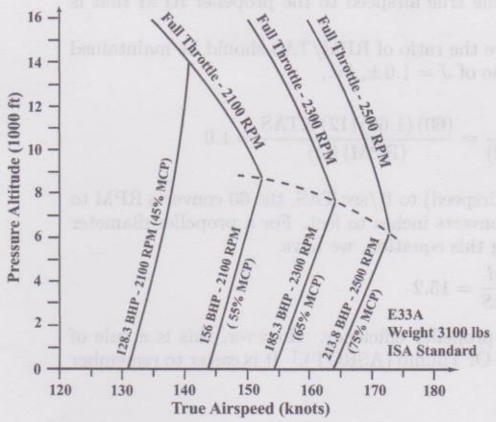

Figure 2. Cruise airspeeds.

RPMs. This implies that an ASROT factor of 15 over-estimates the required RPM. The last column in Table 1 shows the ASROT factor required to obtain the stated RPM at the stated KTAS, i.e., the RPM in column 4 divided by the KTAS in column 5. The result shows that the ASROT factor for this propeller should be adjusted downward. A reasonable adjusted ASROT factor of 14.5 is suggested.

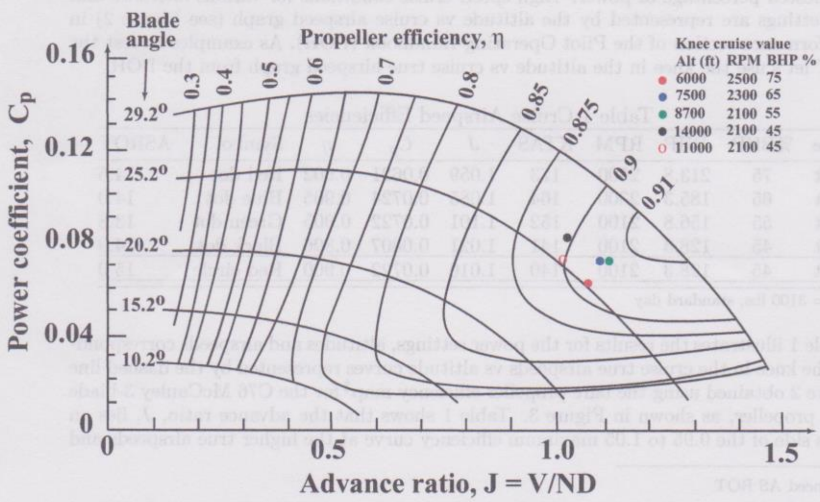

Figure 3. McCauley C76 propeller map.

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Table 1 also shows that typically for higher KTAS higher RPM is required to maintain maximum propeller efficiency, while for lower KTAS lower RPM is required to maintain maximum propeller efficiency. This result confirms the current wisdom that lower RPM for lower true airspeeds increases propeller efficiency.

Turning now to the C76 propeller map shown in Figure 3, notice the clustering of all of the high speed cruise data from 6000 to 14,000 ft. In fact, Figure 3 and Table 1 illustrate that the difference in propeller efficiency is at most a little over 1/2%. This suggests that using the knee values from Figure 2 is a reasonable way to fine tune the ASROT value for a particular propeller. It also indicates that being a little off in either RPM or J is not serious.

Finally, be aware that determining the specific RPM and manifold pressure that will give near optimal propeller efficiency for any given flight condition is an iterative process.

Hang on to Figure 3, we'll be coming back to it when we look at take-off, climb and turbonormalized operations.