ASEN 2004 Aero Lab Part 1:
Computational Aerodynamic Performance of the Tempest UAS

Assigned: Wednesday 16 January
Lab Presentations Due: 4 Feb 19 (Times will be scheduled)
Collaboration Guidance: Team Assignment
Software Requirement:

- Any computational/programming tool (MATLAB, Excel, Python, etc). Must provide code or file with submission
- Any presentation software

1. Learning Objectives

- Reinforce understanding of the importance of the basic aerodynamic coefficients and the drag polar for estimating aerodynamic performance
- Benchmark computational lift, drag, and performance equations against computational fluid dynamics (CFD) data for Tempest UAS
- Understand the advantages and limitations of theoretically and empirically derived conceptual design approximations for estimating drag polar and performance in the design of new aircraft
- Become familiar with CFD data and understand value and limitations of CFD
- Enable the informed use of conceptual design computational methods for the subsequent experimental Aero Lab Part 2: Aero Glider Design

2. Required Deliverables

2.1. Attendance at every lab period is required. Instructions for weekly tasks and the group report will be presented during the scheduled lab time.

2.2. Team presentation with the following guidelines:

- Maximum 10 min + 5 min questions
- At a minimum, presentation must answer all the questions & show all the required figures posed in section 6.
- Every group member is required to brief a portion of the presentation.

2.3. Upload presentation and source calculations file (Excel, Matlab, Hand Calcs, etc) to Canvas lab assignment

3. Background

3.1 – Finite Wing Lift

Through ASEN 2002, you were introduced to airfoil data that represented the lift characteristics of a theoretically "infinite" wing where experimental data for the coefficient of lift were obtained for airfoils using a wing section that spanned the entire width of a wind tunnel test section.

The MH 32 airfoil is a 8.7% thick airfoil at approximately the 30% chord with 2.3% camber. The following profile and 2-D lift characteristics generated via the XFOIL program[1] and Airfoil Tools[2] website:

Figure 1: MH-32 Airfoil Profile (Source: UIUC Airfoil Database[3])

Table 1: MH 32 Lift Curve Data for Re = 10^6 (Source: XFOIL via Airfoil Tools website)

Alpha	C_l	C_d
-5.0000	-0.2446	0.0140
-4.0000	-0.1465	0.0091
-3.0000	-0.0401	0.0073
-2.0000	0.0658	0.0059
-1.0000	0.1717	0.0049
0.0000	0.2737	0.0043
1.0000	0.4058	0.0045
2.0000	0.5143	0.0050
3.0000	0.6167	0.0057
4.0000	0.7194	0.0066
5.0000	0.8201	0.0078
6.0000	0.9193	0.0092
7.0000	1.0129	0.0112
8.0000	1.1027	0.0134

[1] Drela, Mark, MIT, https://web.mit.edu/drela/Public/web/xfoil/
[2] http://airfoiltools.com/index
[3] https://m-selig.ae.illinois.edu/index.html

9.0000	1.1844	0.0165
10.0000	1.2533	0.0201
11.0000	1.2865	0.0252
12.0000	1.2763	0.0332
13.0000	1.2329	0.0475
14.0000	1.1635	0.0720
15.0000	1.0951	0.1052

Clearly, infinite wings do not exist on real aircraft, and the impacts of having a finite (or 3-D) wing have a significant effect on the performance of a wing in generating lift. The presence of a wingtip results in downwash which reduce the effective angle of attack "seen" by the wing, which in turn reduces the coefficient of lift generated by the wing relative to the 2-D airfoil data.

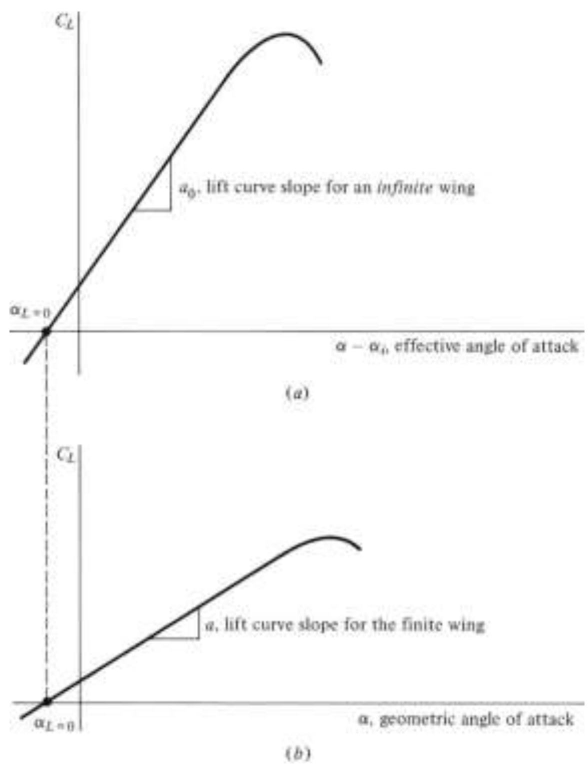


Figure 2: Impacts on lift curve slope for infinite and finite wings from Anderson[1]

Anderson also provides the following formulation of the impacts of downwash on both the lift curve slope (a) and coefficient of lift:

$$a = \frac{dC_L}{d\alpha} = \frac{a_o}{1+\frac{57.3 \cdot a_o}{\pi \cdot e \cdot AR}} \qquad (3.1)$$

[1] Anderson, J. D., Introduction to Flight, 8th Ed., McGraw Hill (2012)

$$C_L = a \cdot (\alpha - \alpha_{L=0}) \qquad (3.2)$$

Where (a_o) is the 2-D airfoil lift curve slope in (1/deg) and (*e*) is the span efficiency factor. For whole aircraft, lift is not just generated by the wing along, but can also be generated by the fuselage and tail surfaces; however, for the purposes of this lab, we will be assuming that the Tempest's wing generates significantly more lift relative to the fuselage and tail and treat the finite wing lift as the total aircraft lift. Additionally, we will assume a span efficiency factor of e = 0.9. Also note that the formulation for the 3-D wing lift coefficient CL only models the linear portion of the lift curve slope and not the nonlinear behavior near stall.

From this finite wing value of C_L, the 3-D wing drag polar can be calculated by adding the induced drag to the profile drag of the 2-D airfoil

$$C_{Dwing} = C_d + \frac{C_L^2}{\pi \cdot e \cdot AR} \qquad (3.3)$$

3.2 - The Whole Aircraft Drag Polar

An investigation of the aerodynamic performance of a whole aircraft generally begins with determining the relevant aerodynamic coefficients, either in the wind tunnel, in flight tests, or using computational fluid dynamics (CFD). Of the relevant aerodynamic coefficients, an aircraft's drag polar (relationship between and aircraft's C_D vs C_L) is especially important as many performance characteristics of an aircraft can be determined from this relationship. Unlike the 3-D wing drag polar developed in section 2, a whole aircraft drag polar must take into account all components of an aircraft which will significantly increase drag as compared to the simple streamlined shape of a wing. Anderson discusses the drag polar of a complete airplane that shows how drag varies with respect to lift, where the most general form is written as:

$$C_D = \underbrace{C_{Do}}_{\text{Parasite Drag}} + \underbrace{k_1 \cdot C_L^2}_{\text{Drag due to Lift}} \qquad (3.3a)$$

$$\text{Where } k_1 = \frac{1}{\pi \cdot e_o \cdot AR} \qquad (3.3b)$$

However, this formulation assumes that an aircraft's minimum drag occurs at zero lift which, for many aircraft, is not true as most have been designed to generate a small value of lift when most streamlined to the relative wind. This can also be seen in the drag polar for cambered airfoils as a shift in the drag polar where the minimum value of drag occurs at a non-zero value of lift. Raymer and Anderson both provide a formulation of the drag polar which accounts for this difference.

$$C_D = C_{Dmin} + k_1 \cdot (C_L - C_{LminD})^2 \qquad (3.4a)$$

Another common formulation which is just an expansion of (3.4a) and utilized in many sources is:

$$C_D = C_{Do} + k_1 C_L^2 + k_2 C_L \qquad (3.4b)$$

$$k_2 = -2k_1 C_{LminD} \qquad (3.4c)$$

$$C_{Do} = C_{Dmin} + k_1 C_{LminD}^2 \qquad (3.4d)$$

The entire parasite drag or zero lift drag coefficient (C_{Do}) is typically determined experimentally or via flight test; however, many methods have been developed in support of conceptual aircraft design that can provide an estimation of this value. The problem at this point is the determination of both C_{Dmin} and the associated C_{LminD} when you do not yet have what C_D is for the entire aircraft is yet. At the conceptual design stage, it is sufficient to approximate these values by assuming that the aircraft designer would attempt to align the aircraft minimum drag to the angle of attack where the wing has minimum drag. Therefore, C_{LminD} for a whole aircraft can be approximated by the following where α_{wing_minD} and $\alpha_{L=0}$ are determined from your wing drag polar and wing lift curve:

$$C_{LminD} = a \cdot \left(\alpha_{wing\_minD} - \alpha_{L=0}\right) \qquad (3.5)$$

The C_{Dmin} coefficient represents the skin friction drag of the entire aircraft. Raymer[1] outlines one such empirically-based method utilizing a equivalent skin friction coefficient (C_{fe}) and the ratio of wetted area (S_{wet}) to the reference wing planform area of the aircraft (S_{ref}).

$$C_{Dmin} = C_{fe} \frac{S_{wet}}{S_{ref}} \qquad (3.6)$$

The equivalent skin friction coefficient is a empirically derived value based on classes of aircraft that has shown to be fairly consistent and accounts for the skin friction drag and the small pressure drag due to separation that occurs when the aircraft is in subsonic cruise conditions. Raymer provides an equivalent skin friction coefficient for a light, single engine propeller plane as 0.0055; however, this is likely high as the Tempest UAS form is much more in line with a powered glider. Roskam provides a typical value for a sailplane to be on the order of 0.003 which is likely low as a sailplane doesn't have a propeller. As a rough approximation for this lab, the value we will utilize for the equivalent skin friction drag is $C_{fe} = 0.004$ which is bound by our two empirical values.

The wetted area of an aircraft is, like it suggests, the surface area of the aircraft that would get wet if it were left outside in the rain. The most accurate method would be to utilize a CAD model of the aircraft to precisely calculate the wetted area; however, there are also many methods that have been developed over time to more quickly approximate this area. Most of them basically reduce the form of an aircraft to more easily analyzed geometric shapes to estimate the wetted area. For this lab, you will conduct either a CAD estimate or an

[1] Raymer, D. P., Aircraft Design: A Conceptual Approach, 2nd Ed., AIAA Inc., Chap. 12 (1992).

approximate method. The following approximation method was developed by Roskam[1] and provides a good estimation.

1. Split the aircraft into logical components (fuselage, wing, tail section) and sum up their individual wetted areas.
2. For fuselages, break them up at points where there is significant cross-sectional area changes and model each section with an equivalent geometric shape (cone, cylinder, etc.)
3. For straight-tapered planforms (wings, horizontal & vertical tails, canards, pylons), utilize the following formula:

$$S_{wet\_plf} = 2 \cdot S_{\exp\_plf} \cdot \left(1 + 0.25 \cdot \left(\frac{t}{c}\right)_r \cdot \frac{1+\tau \cdot \lambda}{1+\lambda}\right) \quad (3.7)$$

   a. Where $S_{\exp\_plf}$ is the exposed planform area of the planform not including the area within the fuselage
   b. $\left(\frac{t}{c}\right)_r$ is the thickness to chord ratio at the planform root at the centerline of the aircraft
   c. $\tau = \frac{\left(\frac{t}{c}\right)_r}{\left(\frac{t}{c}\right)_t}$ is the ratio of the thickness to chord ratios at the root and the tip of the planform
   d. $\lambda = \frac{c_t}{c_r}$ is the taper ratio which is the ratio of the root chord to the tip chord of the planform

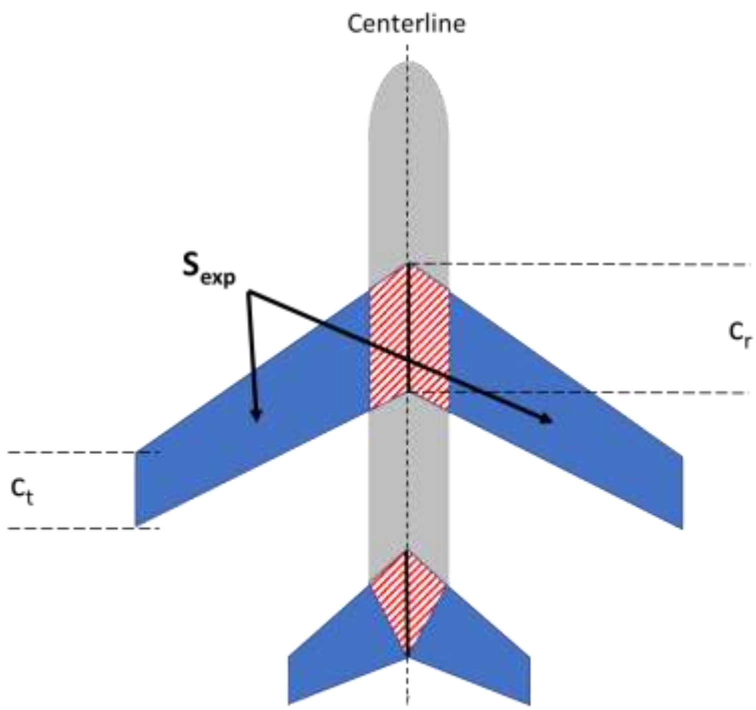


Figure 3: Planform References for Wetted Area Calculation

[1] Roskam, J., "Airplane Design Part II: Preliminary Configuration Design and Integration of the Propulsion System," Chapter 12, DARCorporation, 1997.

The value of Oswald's efficiency number (e_o) likewise has many approximated formulations many of which are based on empirical industry data within each aerospace company. For the purposes of this lab, we will utilize a formula from Raymer which is based on an obscure reference[1] (that does not appear with an internet search). Raymer gives the following formula for the Oswald efficiency that is based on empirical data from actual straight wing aircraft:

$$e_o = 1.78(1 - 0.045 \cdot AR^{0.68}) - 0.64 \qquad (3.8)$$

4. Performance Flight Conditions: Range and Endurance

For this lab, you will not be required to calculate the actual range and endurance of the Tempest UAV, but the required flight conditions in terms of velocity and associated angle of attack to achieve maximum glide and powered range and endurance based on the drag polar and lift curve slope plots you calculated. The gross takeoff weight (GTOW) provided in section 5 will be utilized to determine velocity requirements. Cruise altitude will be 1.8 km standard atmosphere.

From Anderson, the conditions for maximizing range and endurance for both gliding flight and propeller powered flight were derived to yield the following relationships between CL and CD:

For max range, an aircraft should fly where $C_{Do} = kC_L^2$
For max endurance, an aircraft should fly where $3C_{Do} = kC_L^2$

From this relationship and your calculated drag polar and lift curve slope, the velocity required and associated angle of attack for these conditions can be determined for steady unaccelerated flight.

5. The Tempest UAS

5.1 -- Physical Characteristics

The following schematic and table from Roadman et al. (2012) describe the geometry and physical characteristics of the Tempest UAS airframe (dimensions are in meters):

[1] Cavallo, B., "Subsonic Drag Estimation Methods," U.S. Naval Air Development Center, Rept. NADC-AW-6604, 1966.

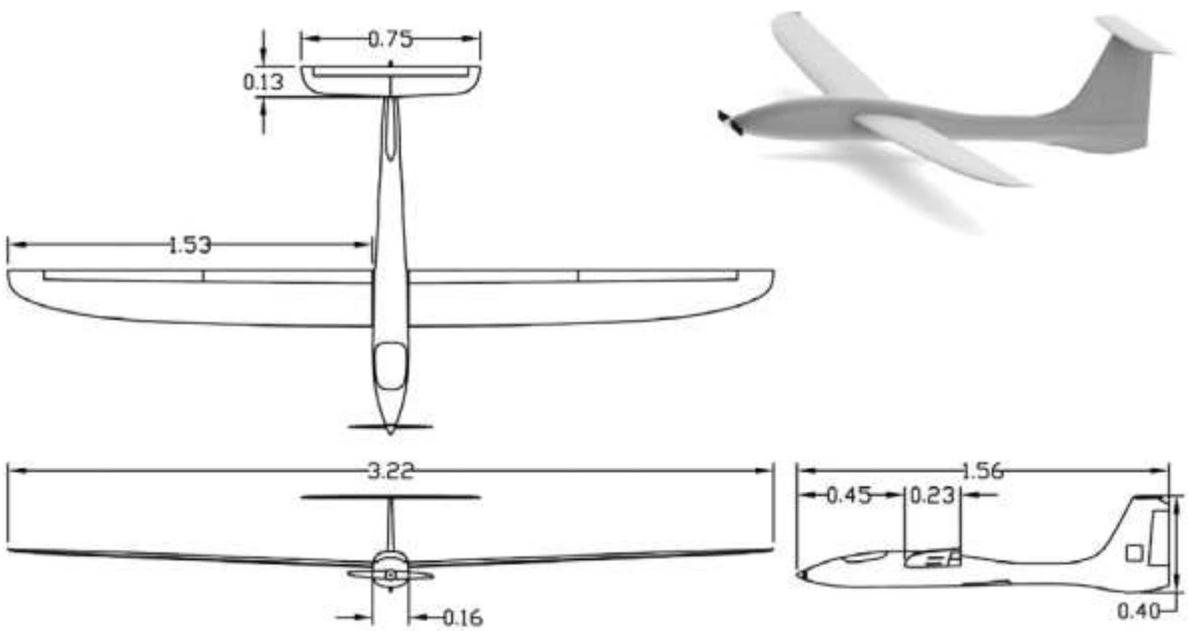


Table 1 Performance characteristics of the Tempest UA

Characteristic	
Wingspan	3.22 m
Wing area	0.63 m^2
Aspect ratio	16.5
Airfoil	MH 32
Length	1.56 m
Empty weight (with propulsion)	4.5 kg
GTOW[a]	6.4 kg
Max load factor	14
Wing loading	33.3 oz/ft^2 (10.2 kg/m^2)
Stall speed	12 m/s
Cruise speed	17–25 m/s
Endurance (with reserve)	45 min
Range	40 km
Typical climb rate	500–2300 ft/ min (2.5–11.5 m/s)

[a]Gross takeoff weight.

Figure 4: Tempest UAS Characteristics

5.2 -- CFD Data

Table 2: Tempest UAS CFD flight data.

α°	***C_L***	***C_D***
-5	-0.32438	0.044251
-4	-0.21503	0.033783
-3	-0.10081	0.028627
-2	0.010503	0.025864
-1	0.12155	0.024643
0	0.24163	0.025099

1	0.34336	0.025635
2	0.45256	0.02766
3	0.56037	0.030677
4	0.66625	0.034855
5	0.76942	0.040403
6	0.86923	0.04759
7	0.96386	0.057108
8	1.0441	0.070132
9	1.0743	0.090921
10	1.0807	0.11193
11	1.0379	0.13254
12	1.034	0.15645

6. Required Discussion & Tasks

1. Lift Curve Comparison (CL vs α): Calculate and plot the estimated C_L vs. α for a 3-D finite wing for -5° < α < 15° and overlay that plot with both the provided 2-D airfoil data for the MH 32 airfoil and the Tempest/TTwistor UAS CFD data provided for the same variables.
2. Drag Polar Comparison (C_D vs. C_L): Calculate and plot on the same figure the estimated 3-D finite wing drag polar, whole aircraft drag polar and the drag polar from the CFD data.
3. L/D Comparison: Calculate and plot the estimated L/D vs α and overlay that plot with the Tempest/TTwistor UAS CFD data provided for the same variables.
4. Performance Flight Condition Comparisons: Calculate the glide/powered flight conditions for maximum range and endurance using both your estimated drag polar and lift curve plot as well as the CFD derived drag polar and lift curve plot (utilize the GTOW of the Tempest and a standard atmosphere cruise altitude of 1.8 km). Summarize the information in a table.
5. Required presentation discussion:
    a. Compare the linear portions of your calculated lift curve plot to that of the 2-D airfoil data and CFD data plot.
        i What are the differences in lift curve slope $\frac{dC_L}{d\propto}$ between the three plots?
        ii Discuss whether the differences between the curves makes sense.
    b. Discuss the differences between your calculated/estimated drag polar and the drag polar developed from the CFD data.
        i Quantify the percent difference between the two drag polar plots with increasing C_L and discuss the results.
        ii Is there more error in the calculated drag polar for the parasite drag term or drag due-to-lift term? Discuss potential reasons for the differences.
        iii How could you improve the calculated drag polar based on what you've found?
    c. Discuss the differences in the performance flight conditions calculated verses those derived from the CFD data and the (L/D) vs α plot.

i What is your estimated maximum L/D and associated velocity and angle of attack? Do this values make sense? Why or why not?
ii How closely does your calculated performance points for max range and endurance (velocity and angle of attack required) match the performance flight conditions determined from the CFD?
iii Which do you trust more (conceptual calculations or CFD) and why?
iv How could you further mitigate the uncertainty and risks in estimating the performance of this UAS?

d. How could you improve the L/D of the Tempest UAS? Vary the design of the Tempest (airfoil, wing planform, etc.) and show how it impacts the lift, drag polar, and L/D calculations.

7. References

Lab derived from:

Argrow, Brian**, ASEN 2004 Experiment 1 Aero 20180124, Spring 2018.**

Additional References:

Anderson, J. D., **Introduction to Flight, 8th Ed.**, McGraw Hill (2012).

Brandt, S, **Introduction to Aeronautics: A Design Perspective, 2nd Ed**, AIAA (2004).

Drela, M., **XFOIL Program**, MIT, https://web.mit.edu/drela/Public/web/xfoil/

Raymer, D. P., **Aircraft Design: A Conceptual Approach, 2nd Ed.**, AIAA Inc., Chap. 12 (2012).

Roskam, J., **Airplane Design Part II: Preliminary Configuration Design and Integration of the Propulsion System**, Chapter 12, DARCorporation (1997).

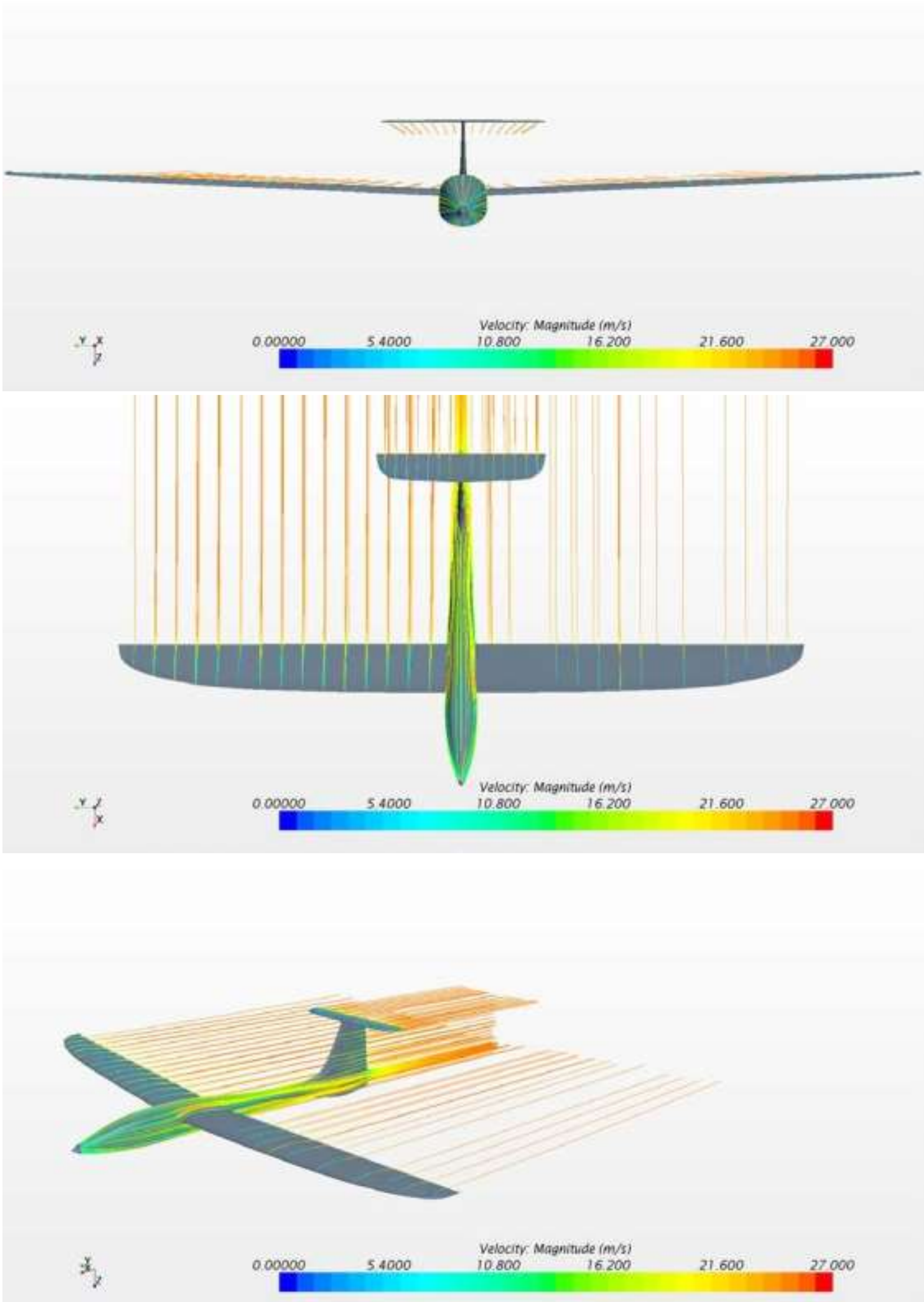
Roskam, J., **Airplane Design Part VI: Preliminary Calculation of Aerodynamic, Thrust, and Power Characteristics**, Chapter 5, DARCorporation (1997).

Roadman, J., Elston, J., Argrow, B., and Frew, E., “Mission Performance of the Tempest Unmanned Aircraft System in Supercell Storms,” *Journal of Aircraft*, Vol. 49, No. 6, pp. 18211830 (2012).

Selig, M., **University of Illinois at Urbana-Champaign Applied Aerodynamics Group**, https://m-selig.ae.illinois.edu/index.html

Appendix: CFD Images

$\alpha = 0^\circ$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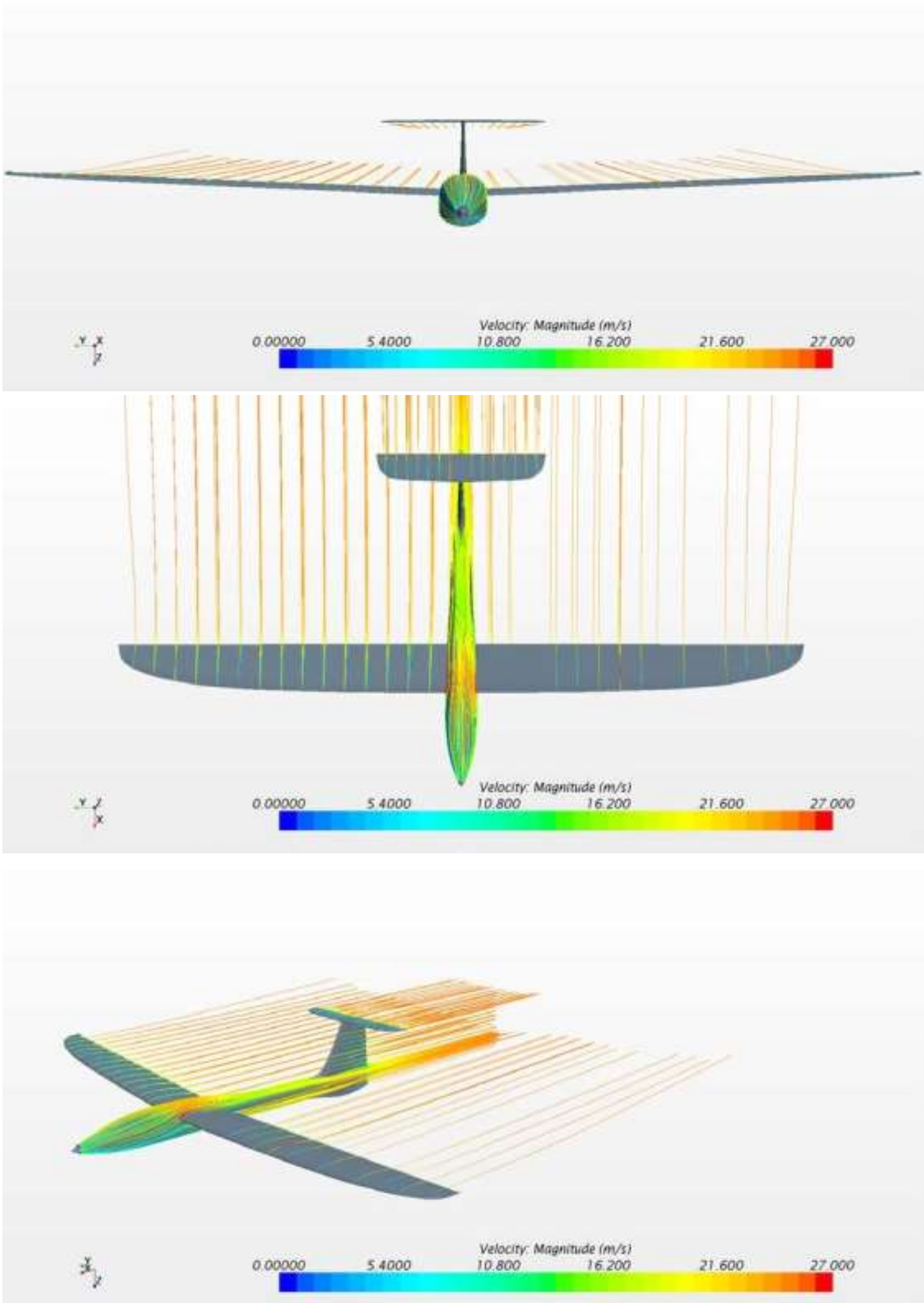
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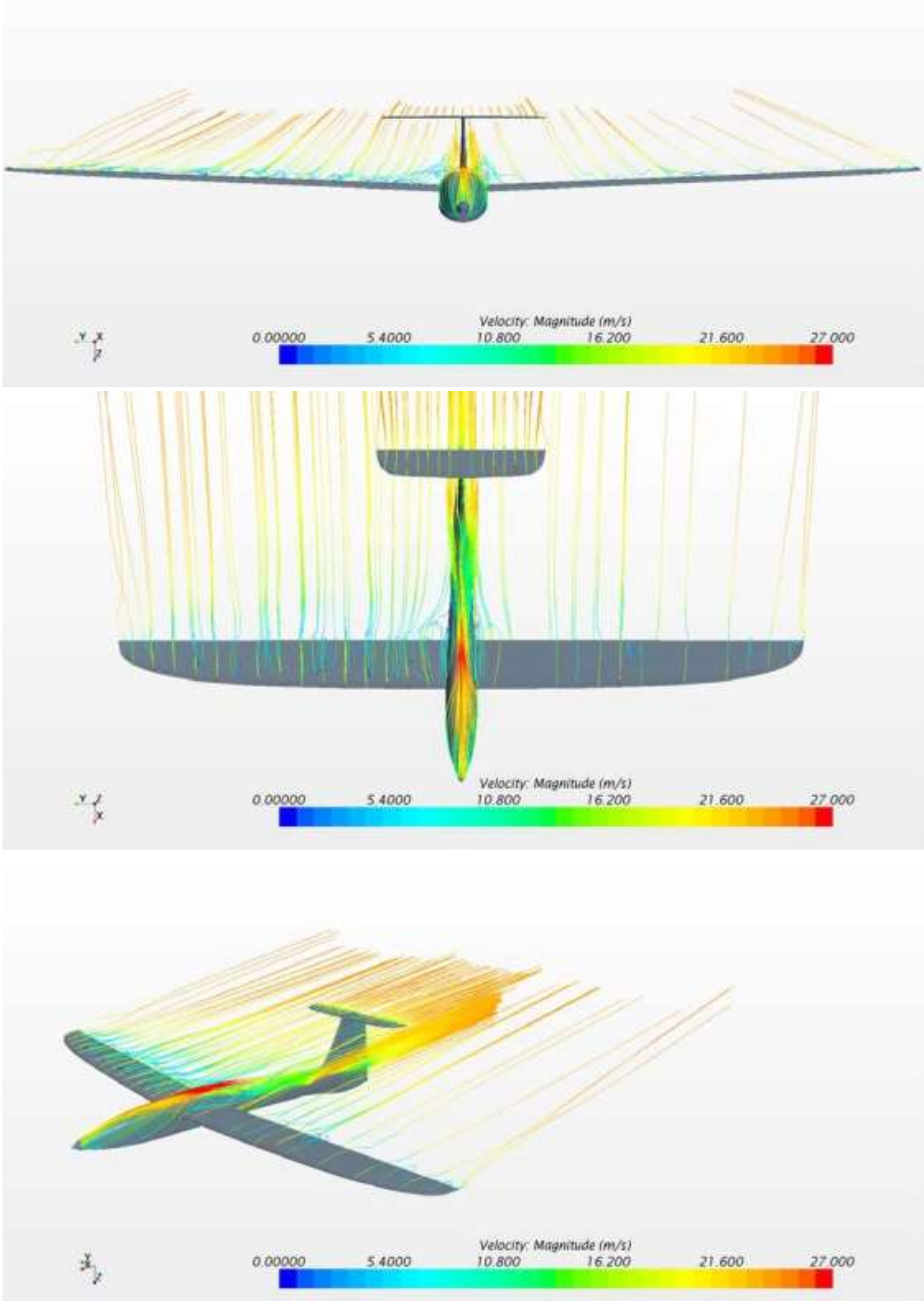
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Velocity: Magnitude (m/s)
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$\alpha = 16^\circ$