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Chad Atlantic Presents: The A-55 "Vulture"

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Airfoil Selection and Wing Geometry Design

Airfoil Selection

Important in determining the aerodynamic characteristics of the aircraft.

- Assumptions:
  - Subsonic aircraft between 0.2 and 0.6 Mach
  - Reynolds number range between 1 to 10 million
  - Analysis performed about the aerodynamic center of the airfoil
  - 2-dimensional analysis for drag and moment
- Airfoil Camber
  - Help determine the airfoil takeoff and landing range.
  - Greater camber will give more lift at a given angle of attack.

Airfoil Selection Continued

- Airfoil Thickness
  - Determined by the historical data based on the design Mach number and thickness ratio.
  - Impacts the airfoil's drag, lift, stall characteristics, and weight.
  - The thickness ratio was found to be 0.15.
  - Max coefficient of lift based on the thickness ratio for NACA 63A-415 was found to be 1.5.
- Airfoil Consideration Elimination Process
  - NACA 4-Series was not considered because it was used primarily throughout the mid to late 20th century.

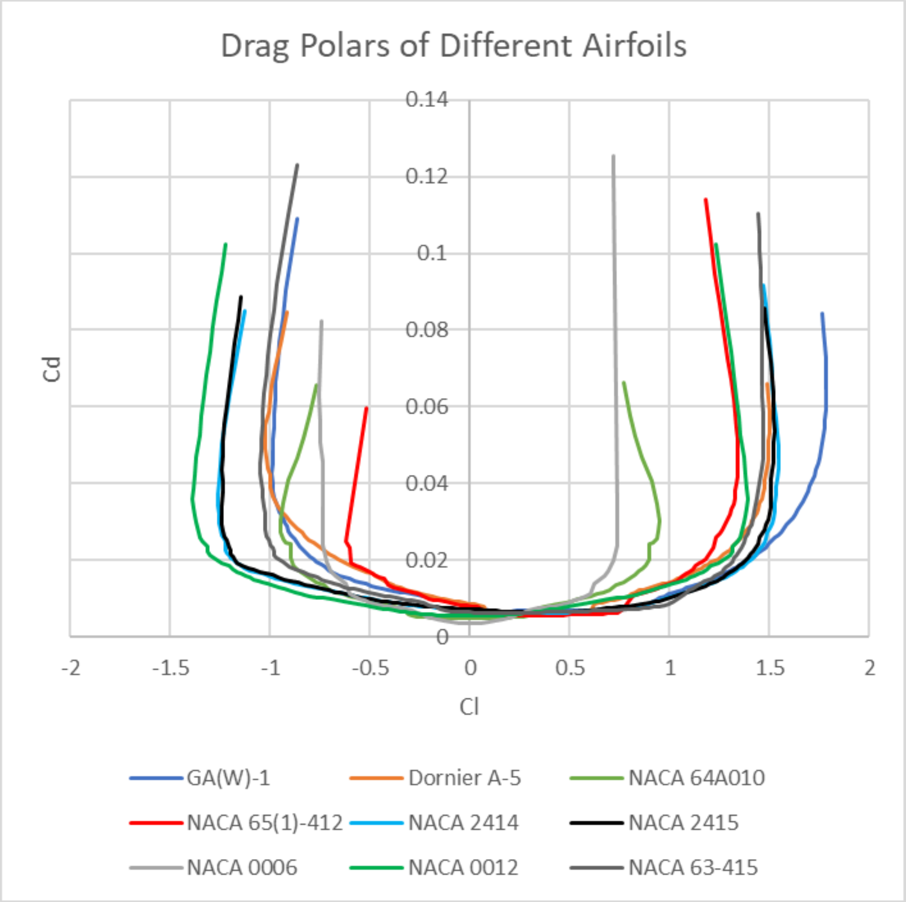
Airfoil Selection Continued

- Airfoil Consideration Elimination Process continued
  - The 7-series and 8-series was not considered for this design because of their experimental nature and lack of similar historical usage.
  - The 16-series was not considered because the primarily usage was propeller rather than wings.
  - The 5-series was not considered because they had poor stall behavior and relatively high drag and weight.
  - Only the 4-series and 6-series were viable options for our mission. The airfoils that was considered were plotted as drag polar curves that showed the coefficient of lift plotted against the coefficient of drag.

Airfoil Selection Continued

Drag polar charts for the considered airfoils

- Curve results from airflow separation effects rather than a drag-due-to-lift calculation.
- Using wing loading determined from the required takeoff and landing distances to determine the target velocity of 200mph and resulted the target coefficient of lift at 0.748.
- NACA63A-415 was determined to be best airfoil for this design.



Wing Geometry

Wing geometry include Aspect Ratio, Wing Sweep, Taper Ratio, Geometric Twist and Incidence Angle, etc.

- Aspect Ratio
  - Low aspect ratio offers structural advantage for storage purposes.
  - During subsonic flight, the lift to drag is proportional to aspect ratio.
  - The aspect ratio determine through historical trend was 2.77.
- Wing Sweep
  - Offers lateral stability at high speeds and improve response for low speeds.
  - This design wing sweep angle was 5 degrees.
- Taper Ratio
  - The taper ratio determined from the quarter chord sweep was 0.2.

Wing Geometry Continued

- Geometric Twist and Incidence Angle
  - To mitigate manufacturing expenses, material constraints, and error propagation associated with optimizing a potentially-nonlinear twist distribution throughout the wing and noting compensatory stability contributions associated with wing sweep and dihedral, the geometric twist was determined to be 0.
  - The wing incidence angle was determined 0 because the contribution of wing incidence for drag optimization in the context of attack aircraft is commonly assumed to be negligible.
- Vertical Displacement
  - For maneuverability and lightweight vehicle, the lower surface of the wing root lies approximately flush with the bottom of the fuselage.

Wing Geometry Continued

- Wing Dihedral
  - The wing dihedral was found to be 4 degrees for this design's subsonic swept wing.
- High-Lift Devices
  - Because the max coefficient of lift estimated was so close to the max coefficient of lift of the airfoil selected, the high-lift devices were considered as unnecessary.
- Aerodynamic Characteristics of the Wing
  - The aerodynamic characteristics were found by using XFOIL's 'visc'n program and iteratively testing the airfoil at different angles of attack under different Reynolds and Mach Numbers

Aerodynamic Characteristics Plots

Coefficient of Drag

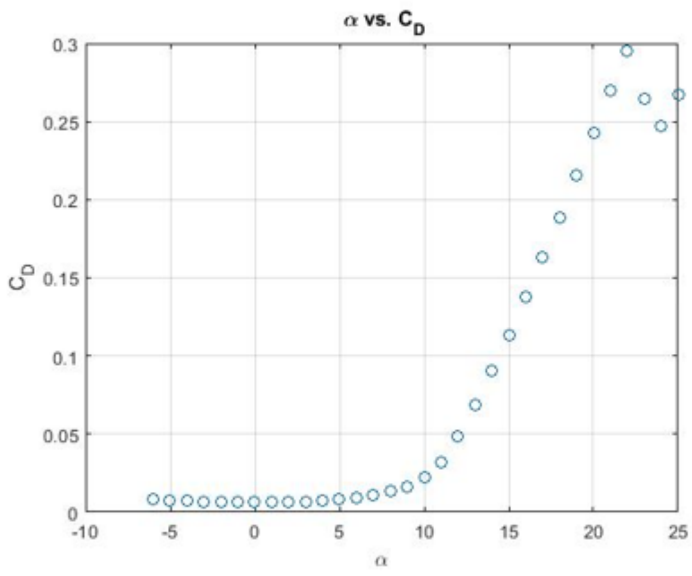


Fig. 11. Angle of Attack vs Coefficient of Drag for a NACA2415 Airfoil ranging from -6° to 26° for Reynolds Number of 5322000.

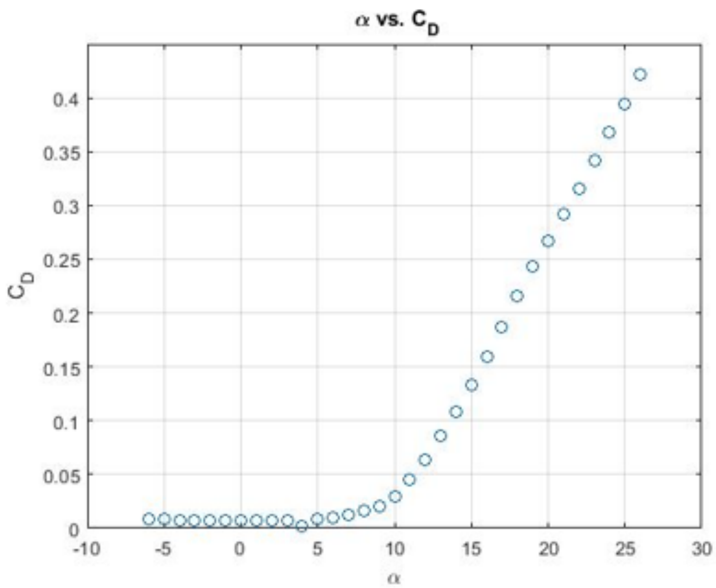


Fig. 10. Angle of Attack vs Coefficient of Drag for a NACA2415 Airfoil ranging from -6° to 26° for Reynolds Number of 3498000.

Aerodynamic Characteristics Plots

Coefficient of Lift

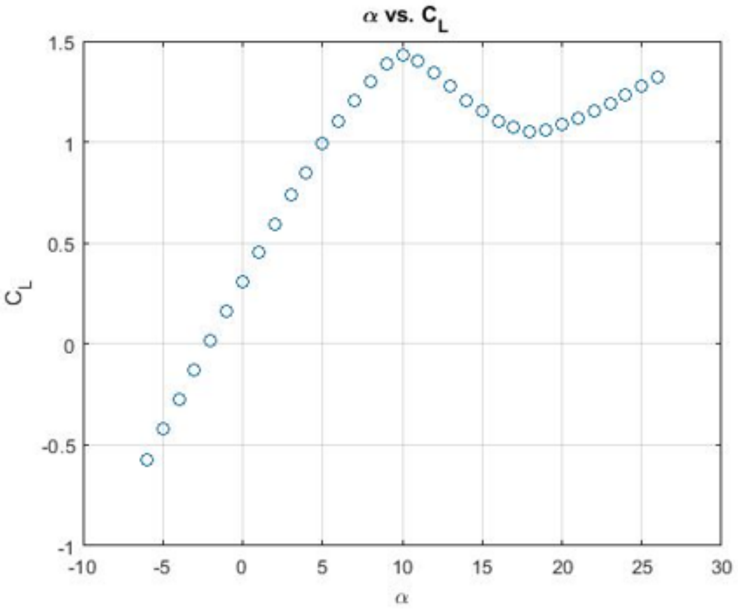


Fig. 13. Angle of Attack vs Coefficient of Lift for a NACA2415 Airfoil ranging from -6° to 26° for Reynolds Number of 5322000.

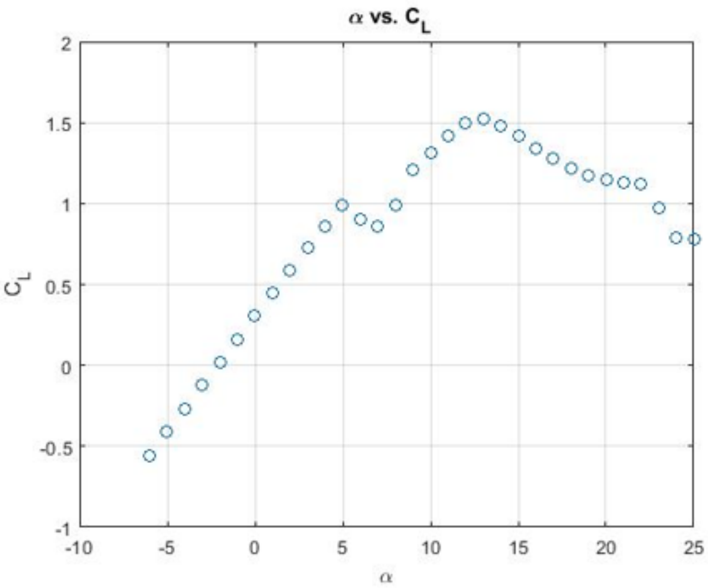


Fig. 12. Angle of Attack vs Coefficient of Lift for a NACA2415 Airfoil ranging from -6° to 26° for Reynolds Number of 3498000.

Aerodynamic Characteristics Plots

Coefficient of Moment

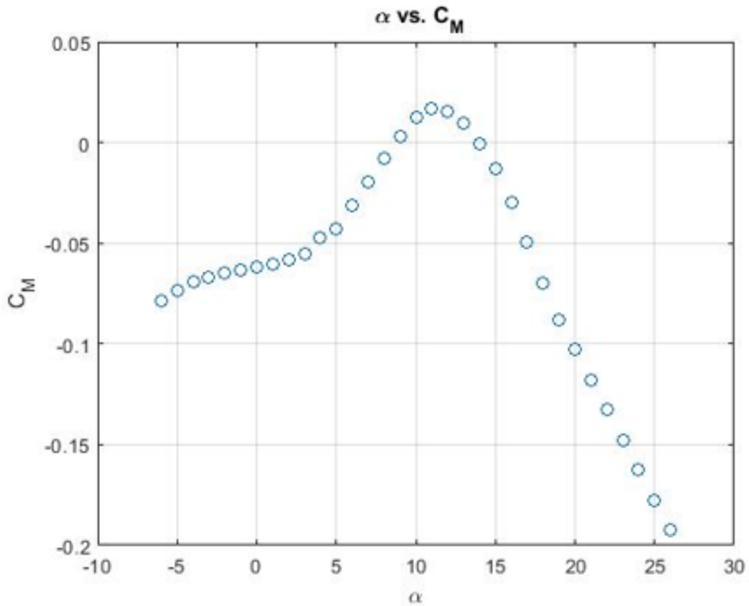


Fig. 15. Fig. 16. Angle of Attack vs Quarter-Chord Moment Coefficient for a NACA2415 Airfoil ranging from -6° to 26° for Reynolds Number of 5322000.

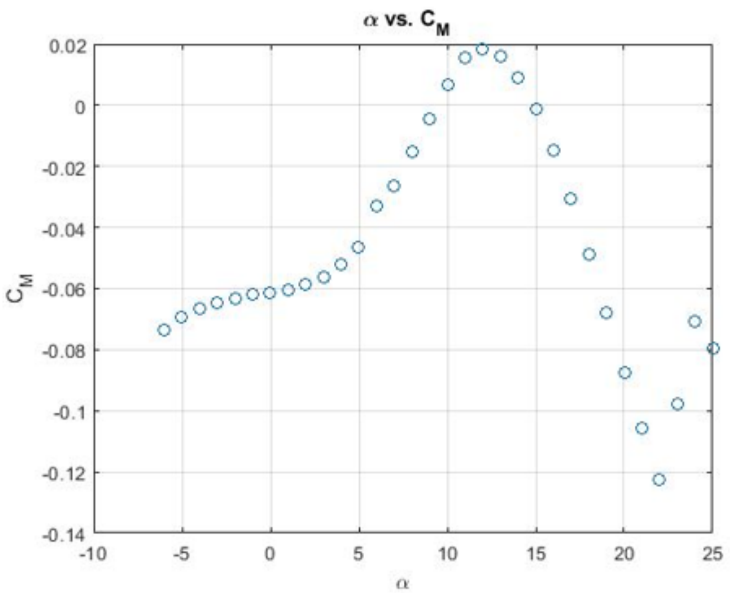


Fig. 14. Angle of Attack vs Quarter-Chord Moment Coefficient for a NACA2415 Airfoil ranging from -6° to 26° for Reynolds Number of 3498000.

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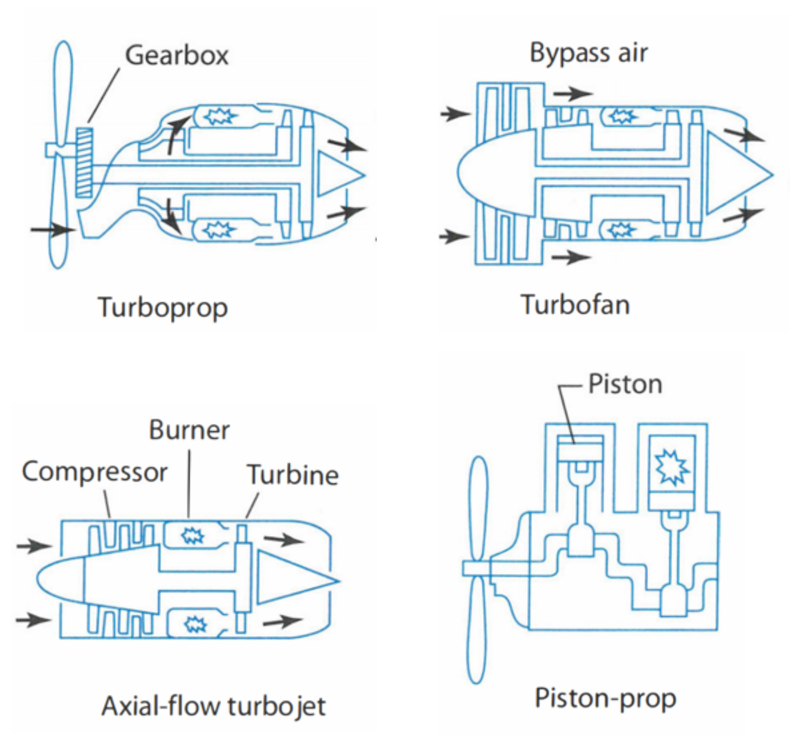
Propulsion

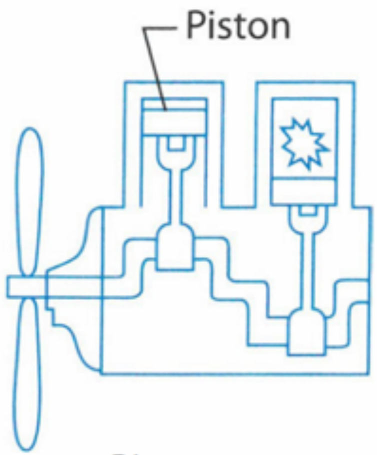
Engine types

- Important in determining capabilities, limitations, and design
  - Fuel consumption, weight, and thrust had to be considered
- Main types of engines Considered
  - Turboprop
  - Turbofan
  - Turbojet
  - Piston-prop

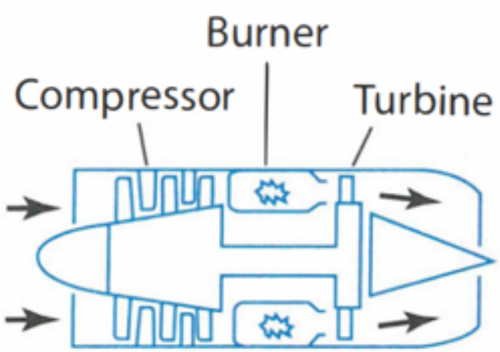


Early Elimination

- Two types eliminated early
  - Piston Prop
    - Too low performance
    - Very low max speeds
    - Unable to reach high altitudes
    - Not enough power generation for payload
  - Turbojet
    - Too high performance
    - Do not need such high speeds or power generation
    - "Overkill" for mission at hand
    - Less fuel efficient
    - Greater runway distance needed



Piston-prop



Axial-flow turbojet

Remaining Considerations

- Turbofan and Turboprop
  - Variations of of the turbojet
  - Aim to be more efficient at lower speeds
    - Applies the power of the engine to a cross section of air
      - Turboprop: uses a propeller
      - Turbofan: uses multiple fan stages

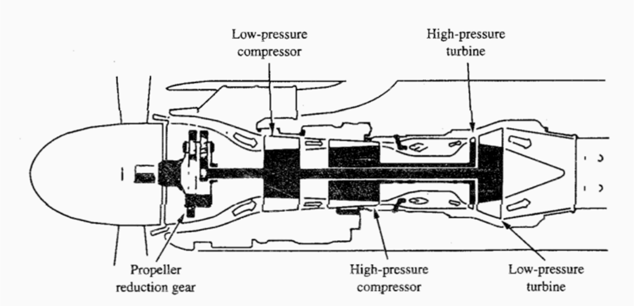


Figure 2: The general design of a turboprop engine. [10]

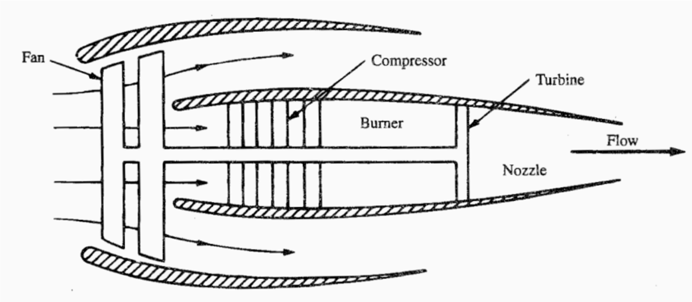


Figure 3: The general design of a turbofan engine. [10]

Bypass Ratio

- Ratio of mass flow rate of air coming in to air going out
- Turboprop has much higher ratio: 20 – 100
  - High ratio reduces max flight speed
    - Due to shockwave formation
- Turbofan is much lower: 0.2 - 5

Fuel Consumption Comparison

- Turboprops perform better at lower altitudes
  - Reduced air density is a hinderance at high altitudes
  - Efficiency is a function of propeller diameter
- Turbofans perform better at higher altitudes
  - Increased airframe drag at lower altitudes
  - Efficiency s function of internal compressor limits
    - Ex: Temperature

Takeoff/Landing Distance

- Turboprop
  - Can land on shorter runways: usually need ~ 3,000 ft.
  - Can land on rougher terrain, like grass or dirt
- Turbofan
  - Need longer runways: usually ~5,000 ft.
  - Need concrete runways and more solid ground

Final Engine Selection

- Trade studies were conducted on multiple engine models

Engine	Dry Weight [lb]	Length [in]	Diameter [in]	Bypass Ratio	SFC/h
Pratt & Whitney Canada PT6A-68B turboprop	575	72.2	19	50-60:1	0.54 lb/hph
Pratt & Whitney Canada PT6A-68 turboprop[5]	572	72.2	19	50-60:1	0.54 lb/hph
Honeywell TPE331 turboprop[7]	336	46	21	10.55:1	0.534 lb(hp-hr)
Walter M601D-1 turboprop[8]	434.3	65.94	23.23	6.55:1	0.62 lb/(hp-h)
Honeywell/ITEC F124-GA-100 turbofan[5]	1050	102.1	36	0.49:1	0.78 lb/lbf-hr
Honeywell TFE731 turbofan[5]	743-899	49.7	39.4	2.8:1	0.469-0.571 lb/(lbf-hr)
Ivchenko-Progress AI-222-25 turbofan[6]	970-1234	77.17	25.2	1.19:1	0.66 kg/(kgf-hr)

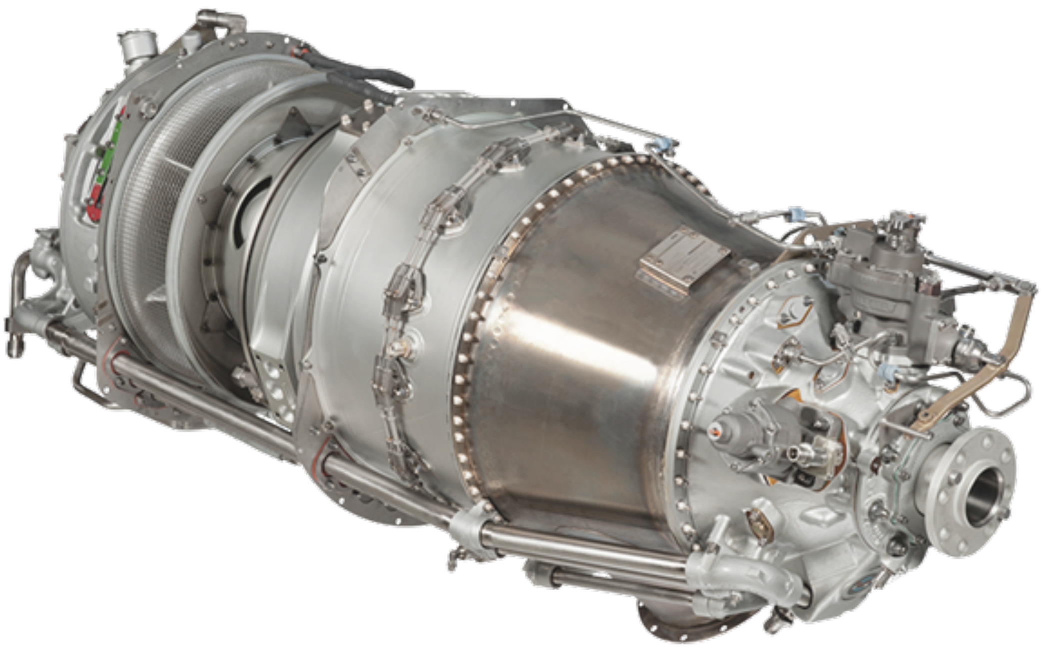
Turboprop shows advantages in dry weight and Bypass ratio

Suitable power found for turboprops

Engine	Thrust	Engine Shaft Power	Equivalent SHP
Pratt & Whitney Canada PT6A-68B turboprop	*DNF*	1600 hp	1694 hp
Pratt & Whitney Canada PT6A-68 turboprop	1178 kw	1250 hp	1324 hp
Honeywell TPE331 turboprop	*DNF*	940 hp	1000 hp
Walter M601D-1 turboprop	*DNF*	724 hp	*DNF*
Honeywell/ITEC F124-GA-100 turbofan	6280 lbf (max)	*DNF*	*DNF*
Honeywell TFE731 turbofan	3500-4750 lbf (max)	*DNF*	*DNF*
Ivchenko-Progress AI-222-25 turbofan	5,552.78 lbf	*DNF*	*DNF*

Final Engine Selection

- Turboprop showed many advantages over the Turbofan
- Final Selection: Pratt & Whitney Canada PT6A-68B Turboprop



Final Engine Selection

- Parameters that led to Turboprop selection
  - Flight Ceiling
  - Payload
  - Flight endurance
  - Range
  - Takeoff/Landing Conditions
- The selection of THIS model over the others was due to...
  - Dry weight
  - Bypass Ratio

Propeller Design - Diameter

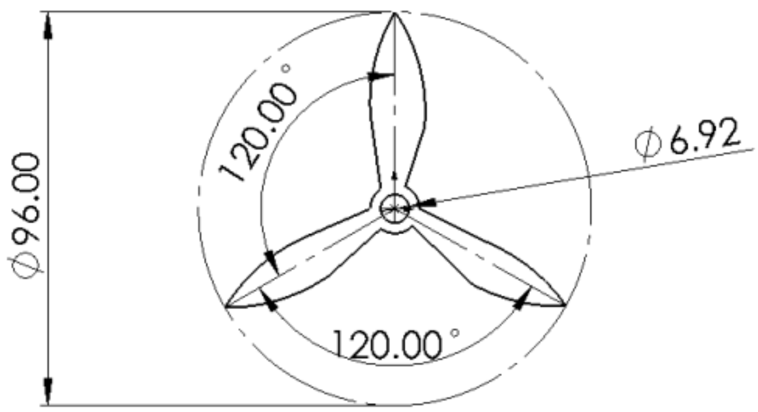
- Must be designed to achieve desired helical and static tip speed
  - 290 m/s and 252 m/s, respectively
- Gives a diameter of 8 ft.

$$(V_{tip})_{helical} = \sqrt{(V_{tip})^2_{static} + V^2} \ ,$$

$$\text{where } (V_{tip})_{static} = \pi n D$$

Number of Blades

- Number of blades can not be too high or too low
  - Too few: large pressure pulses, increased vibration and cabin noise
  - Too man: compromises efficiency
- Our choice: 3 blades



Thrust

- Forward flight thrust: ~ 1,500 lbf

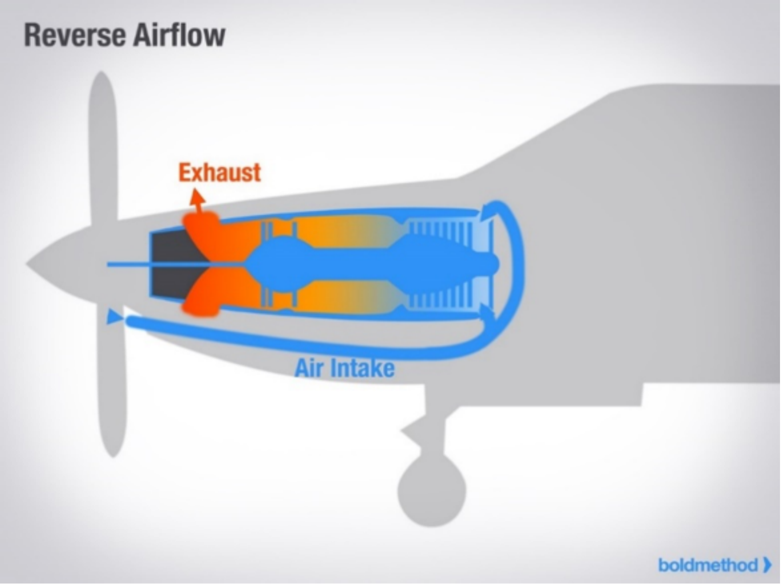
$$T_{ff} = \frac{550\ bhp*\eta_p*P}{V}$$

- Static Flight Thrust: ~2325 lbf

$$T_s = \frac{C_T}{C_p} * \frac{550\ bhp*P}{n*D}$$

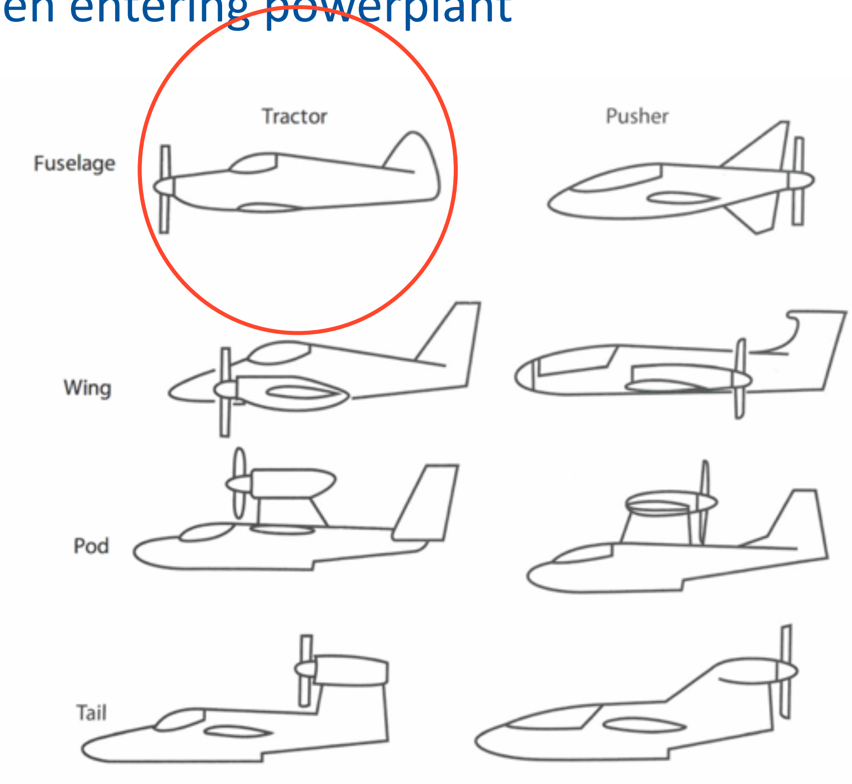
Exhaust Design

- Placed directly behind the propellers
- Advantages of location:
  - Protection from foreign objects
  - Increased survivability
  - Takeoff/land on a range of terrain



Powerplant Location

- Tractor design: fuselage
- Allows for air to be cleaner when entering powerplant
- Provides greater stability



Fuel System

- Bladder fuel tanks will be used
  - Provide greater survivability through self-sealing technology
  - Allows for storage in wing and fuselage
    - 3300 lbs of fuel
      - 61.4 ft^3 in fuselage
      - 9.2 ft^3 in each wing
- Fuel Type: JP-8
  - Commonly used in military aircraft
  - Density allows us to be lighter
    - Allows us to complete ferry and payload missions

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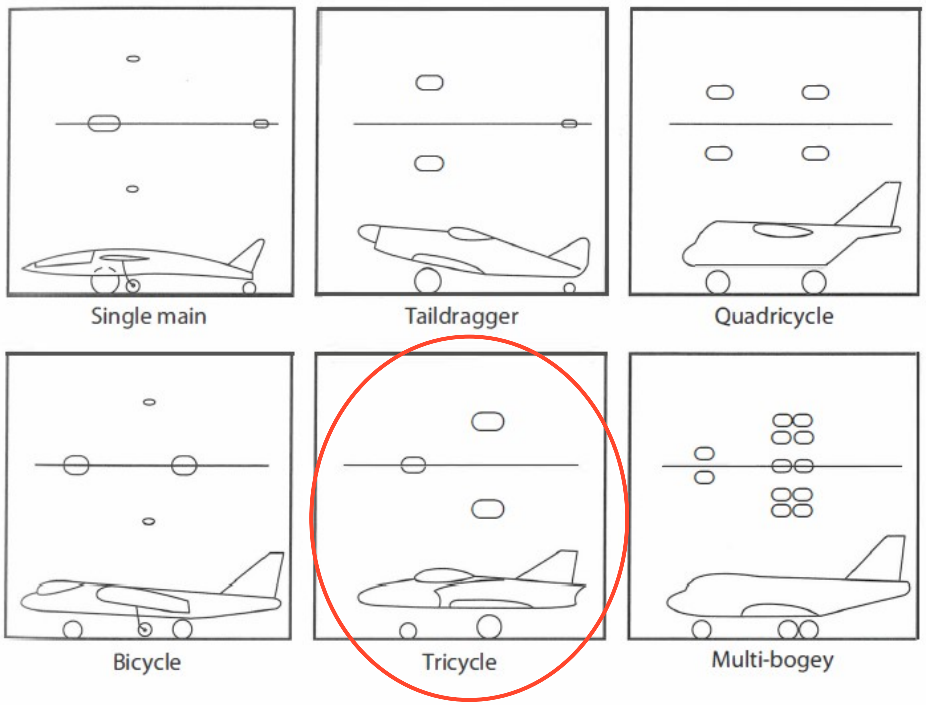
Landing Gear

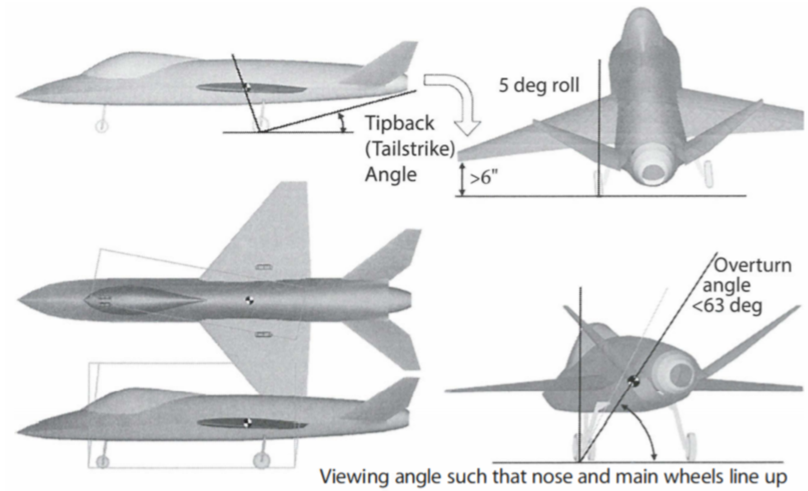
Configuration

- Design Options
  - Tail Dragger – too unstable
  - Tricycle – stable and versatile
- Nose does not need to be perfectly aligned with runway
- Above average forward visibility of the ground



Location

- Nose wheel must carry between 5% and 20% of total weight
- Calculated angles
  - Overturn: 45°
  - Tipback: 15°
  - Tailback: 10°



Wheel	X-Location (aft of nose) [ft]	Y-Location [ft]	Z-Location (off of ground) [ft]
Nose	2.99269	0.00000	2.771698
Left	11.00223	-1.93957	2.771698
Right	11.00223	1.93957	2.771698

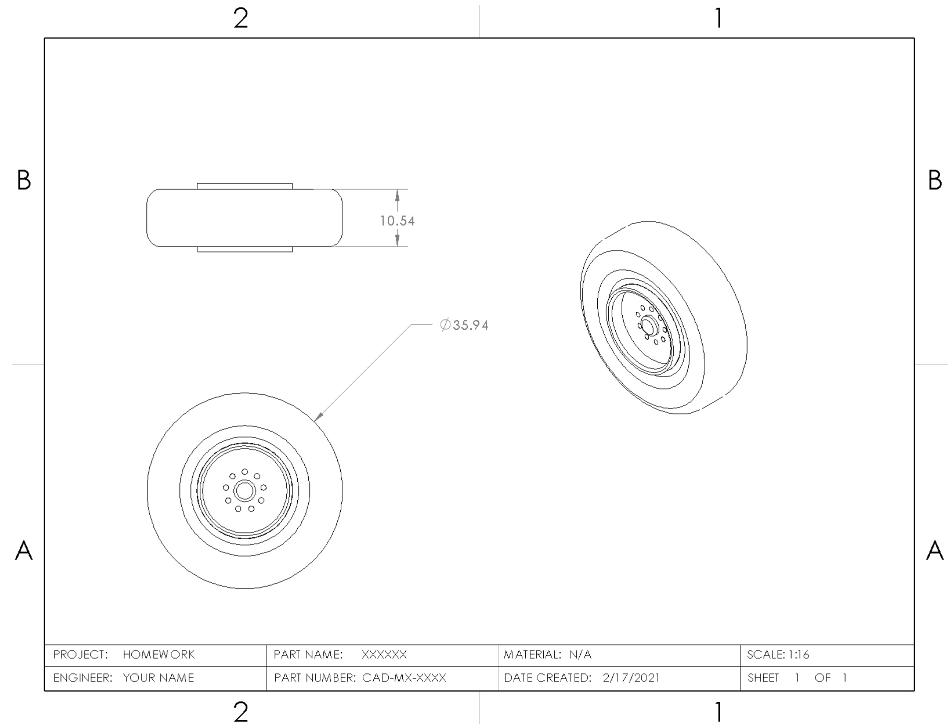
Tire Sizing/Selection

- Tire Selection
  - Nose: Type VII 30x7.7
  - Main: Type VII 36x11

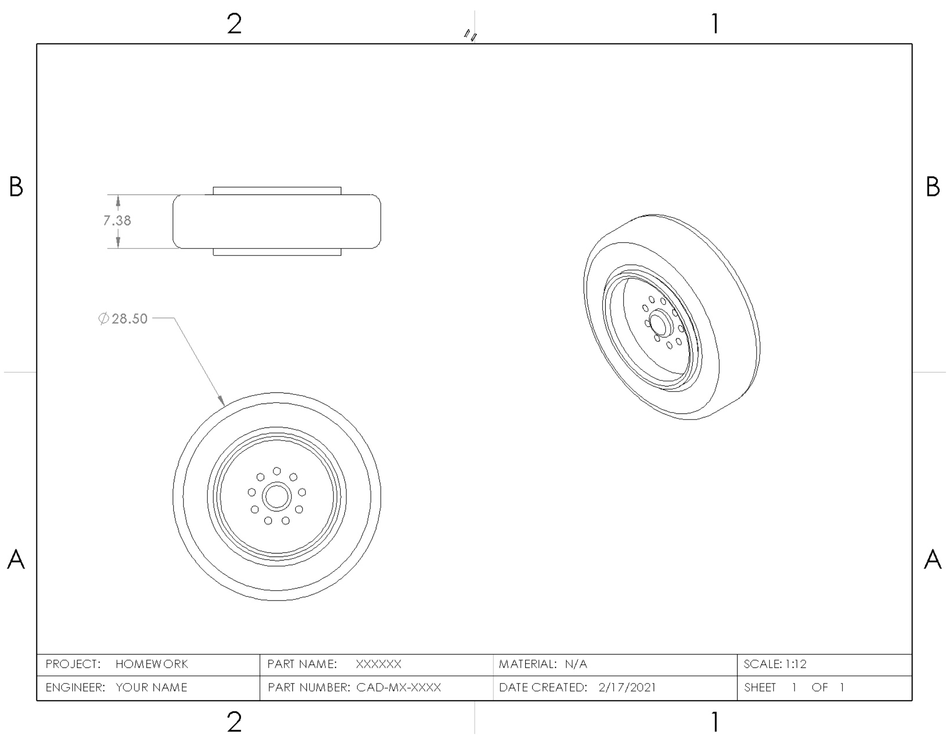
Size	Speed, mph	Max Load, lb	Infl, psi	Max Width, in.	Max Diameter, in.	Rolling Radius	Wheel Diameter	Number of Plies
Type III								
5.00–4	120	1200	55	5.05	13.25	5.2	4.0	6
5.00–4	120	2200	95	5.05	13.25	5.2	4.0	12
7.00–8	120	2400	46	7.30	20.85	8.3	8.0	6
8.50-10	120	3250	41	9.05	26.30	10.4	10.0	6
8.50–10	120	4400	55	8.70	25.65	10.2	10.0	8
9.50–16	160	9250	90	9.70	33.35	13.9	16.0	10
12.50–16	160	12,800	75	12.75	38.45	15.6	16.0	12
20.00–20	174 kt	46,500	125	20.10	56.00	22.1	20.0	26
Type VII								
16 × 4.4	210	1100	55	4.45	16.00	6.9	8.0	4
18 × 4.4	174 kt	2100	100	4.45	17.90	7.9	10.0	6
18 × 4.4	217 kt	4350	225	4.45	17.90	7.9	10.0	12
24 × 5.5	174 kt	11,500	355	5.75	24.15	10.6	14.0	16
30 × 7.7	230	16,500	270	7.85	29.40	12.7	16.0	18
36 × 11	217 kt	26,000	235	11.50	35.10	14.7	16.0	24
40 × 14	174 kt	33,500	200	14.00	39.80	16.5	16.0	28
46 × 16	225	48,000	245	16.00	45.25	19.0	20.0	32
50 × 18	225	41,770	155	17.50	49.50	20.4	20.0	26
Three-part Name								
18 × 4.25–10	210	2300	100	4.70	18.25	7.9	10.0	6
21 × 7.25–10	210	5150	135	7.20	21.25	9.0	10.0	10
28 × 9.00–12	156 kt	16,650	235	8.85	27.60	11.6	12.0	22
37 × 14.0–14	225	25,000	160	14.0	37.0	15.1	14.0	24
47 × 18–18	195 kt	43,700	175	17.9	46.9	19.2	18.0	30
52 × 20.5–23	235	63,700	195	20.5	52.0	21.3	23.0	30

Wheel	Diameter [in]	Width [in]
Nose	28.50	7.37
Main	35.94	10.54

CAD Drawings



Main Wheel Assembly



Nose Wheel Assembly

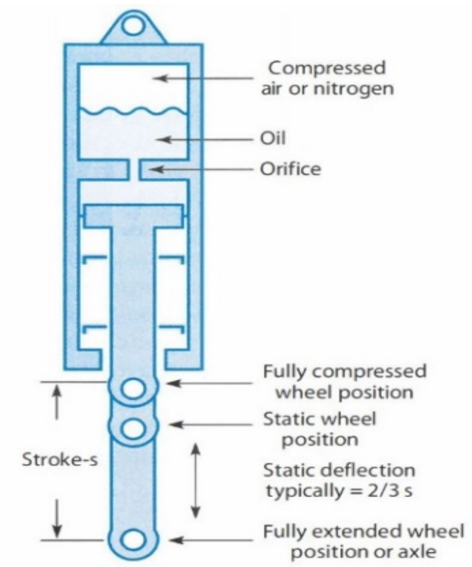
Tire Pressure

- Surface Condition Limitations
  - Nose Tire: hard packed sand to aircraft carrier
  - Main Tires: dry grass on hard soil to aircraft carrier

Surface	Maximum Pressure psi	Maximum Pressure {kPa}
Aircraft carrier	200+	1380+
Major military airfield	200	1380
Major civil airfield	120	828
Tarmac runway, good foundation	70–90	480–620
Tarmac runway, poor foundation	50–70	345–480
Temporary metal runway	50–70	345–480
Dry grass on hard soil	45–60	310–415
Wet grass on soft soil	30–45	210–310
Hard packed sand	40–60	275–415
Soft sand	25–35	170–240

Tire	Pressure [psi]
Nose	37.47
Main	47.67

Shock Absorbers

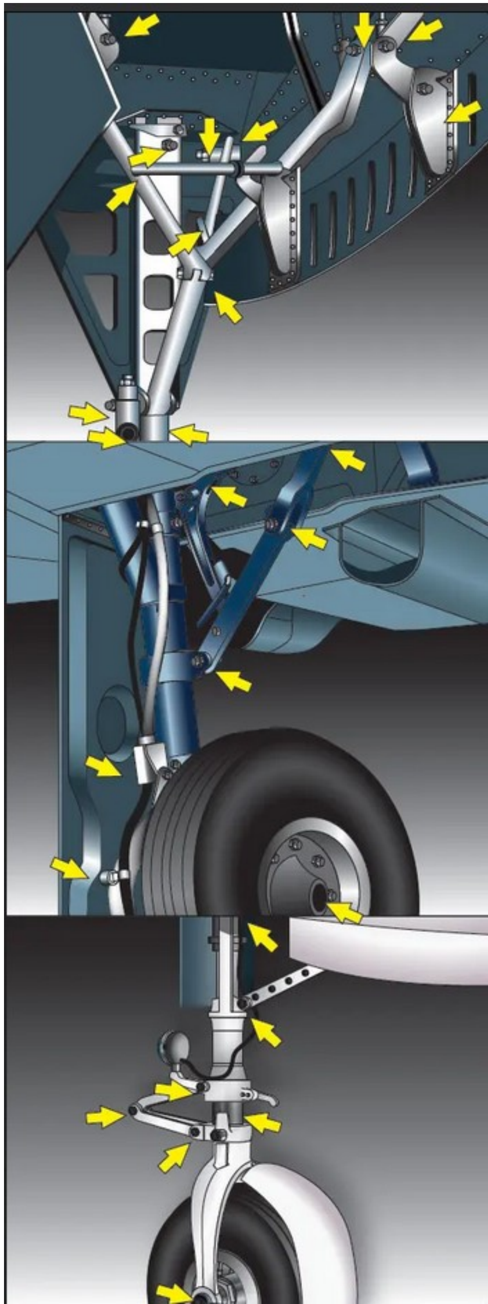


- Oleopneumatic shock-absorber configuration
  - Advantages:
    - Lightweight as compared to a spring system
    - Absorbs the kinetic energy experienced by the landing gear during impact
- This configuration leaves an excess of 29 in. to accommodate any foreseeable displacement of the propeller below the fuselage due to mounting configurations

Stroke [in]	Oleo Total Length [in]	Diameter, main [in]	Diameter, nose [in]
17.43	52.29	3.90	2.66

Retraction System

- Push-pull rod system
  - Provides protection for the landing gear
  - Improves aerodynamic properties
- Hydraulic system
  - Driven by selector and sequencing valves linked to actuating cylinders
  - Design transmission systems for electric motors cannot withstand repeated impulses associated with landings



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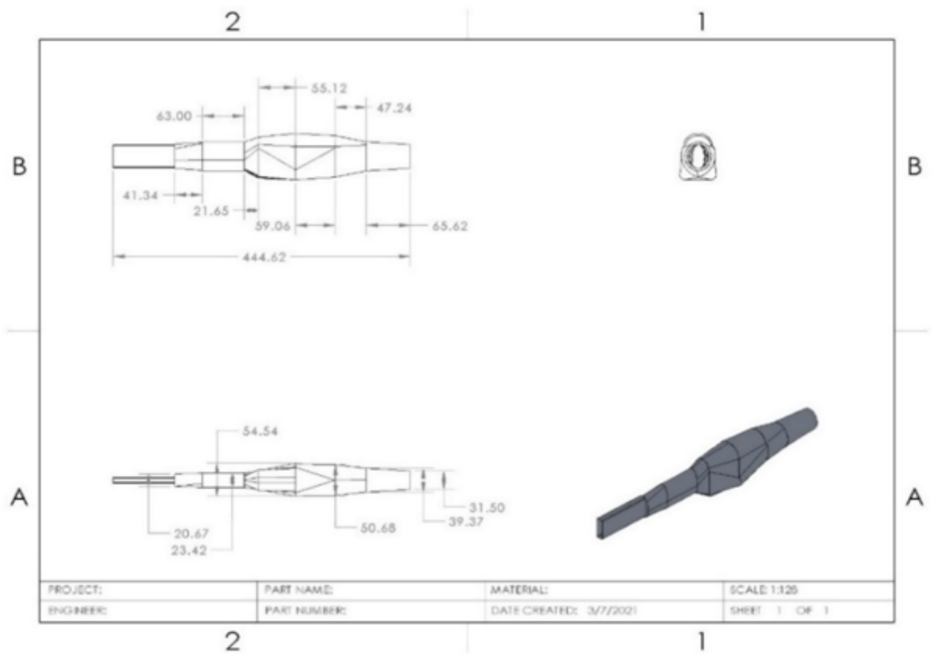
Fuselage Design and Crew Station

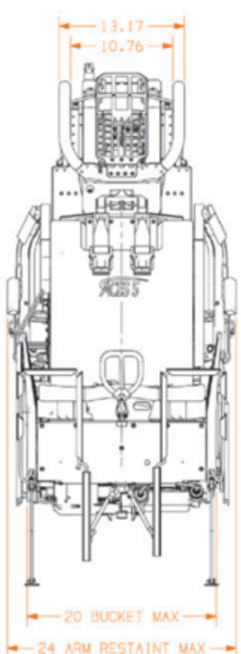
Fuselage Design

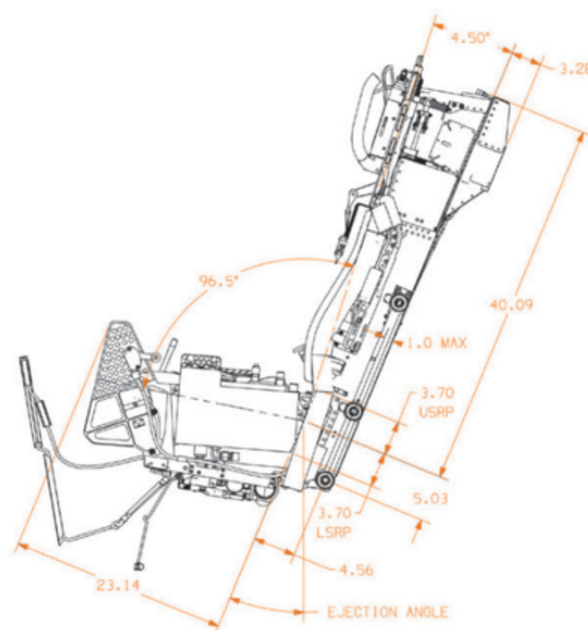
- Designed to hold up to 2 crew members
- Designed to minimize wasted space
- 37.05 ft. In length
- Streamlined profiles minimizes radar detectability



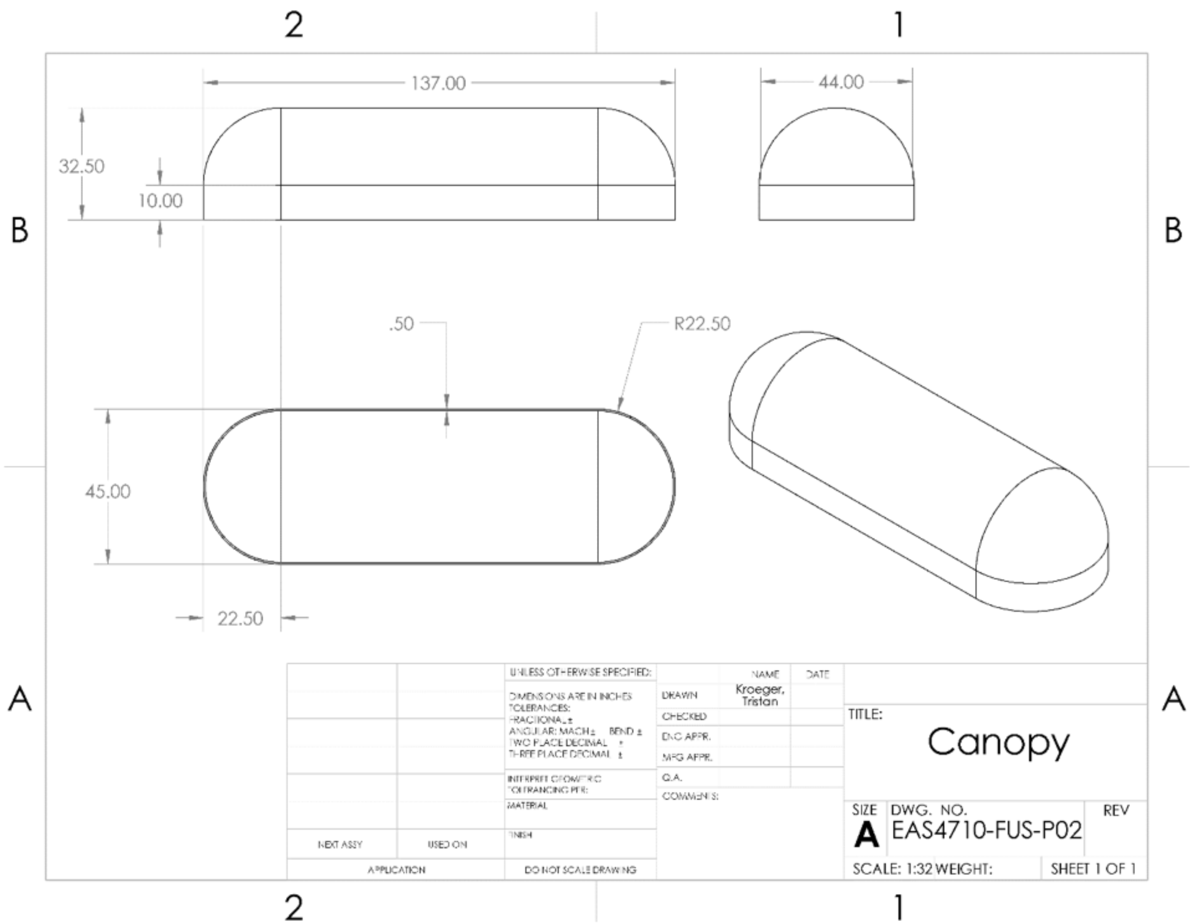
Crew Station Design

- ACES 5 zero-zero ejection system from Collins Aerospace
  - 46 inches high, 24 inches wide, 32 inches deep
  - 18 inches added for overhead, 30 inches for leg room

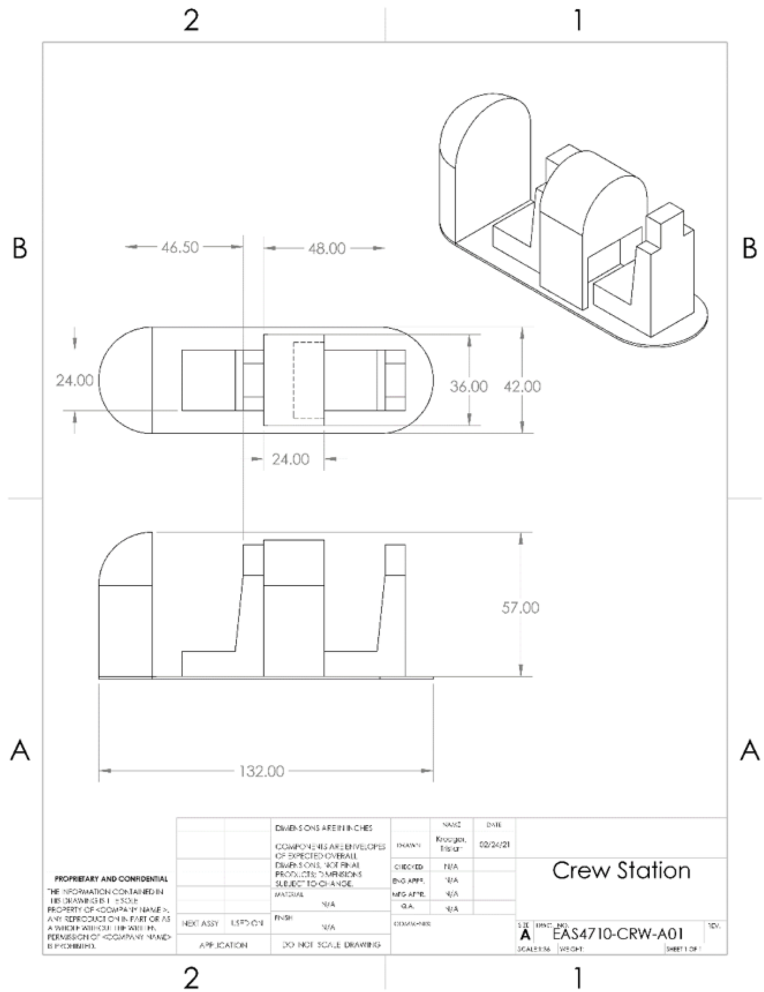




Crew Station Design (cont.)



Crew Station Design (cont.)



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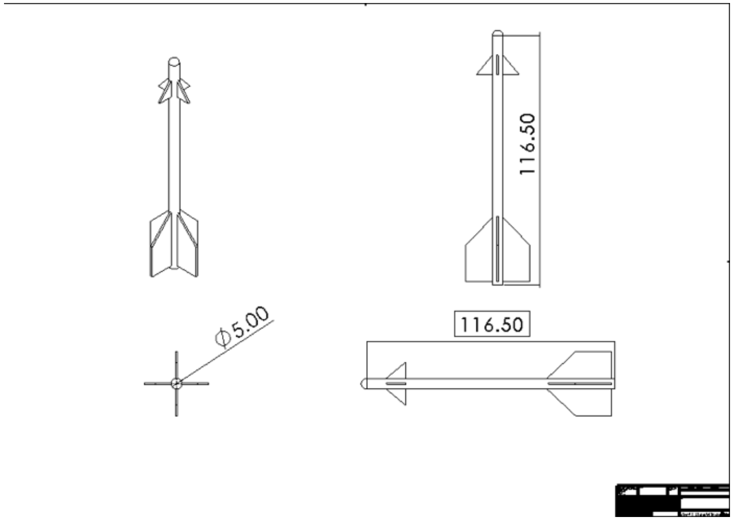
Weapons and Survivability

Weapons Carriage

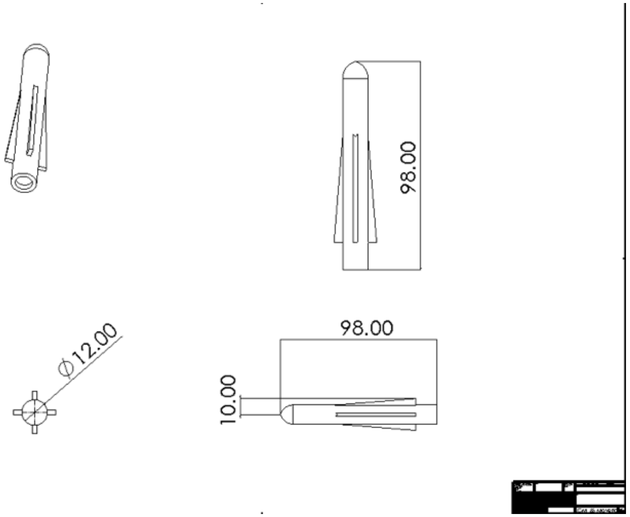
- Parameters to consider include:
  - Flexibility, safety, drag, and stealth
- Must be able to carry a 3,300 lb. Payload
  - Range from rocket, missiles, and guns
  - Dispersed over 5 hardpoints:
    - 2 under each wing, 1 under the fuselage

Weapons Selection

- Missiles
  - 2 AIM-9L Sidewinder
    - Air-to-Air Missile

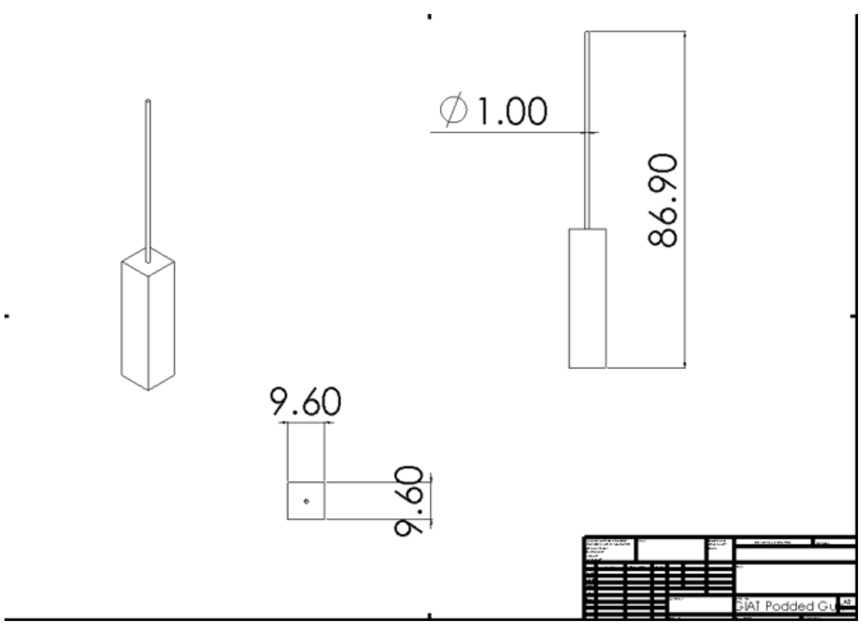


  - 2 AGM-65 Maverick
    - Guided Air-to-Ground Missile



Weapons Selection (cont.)

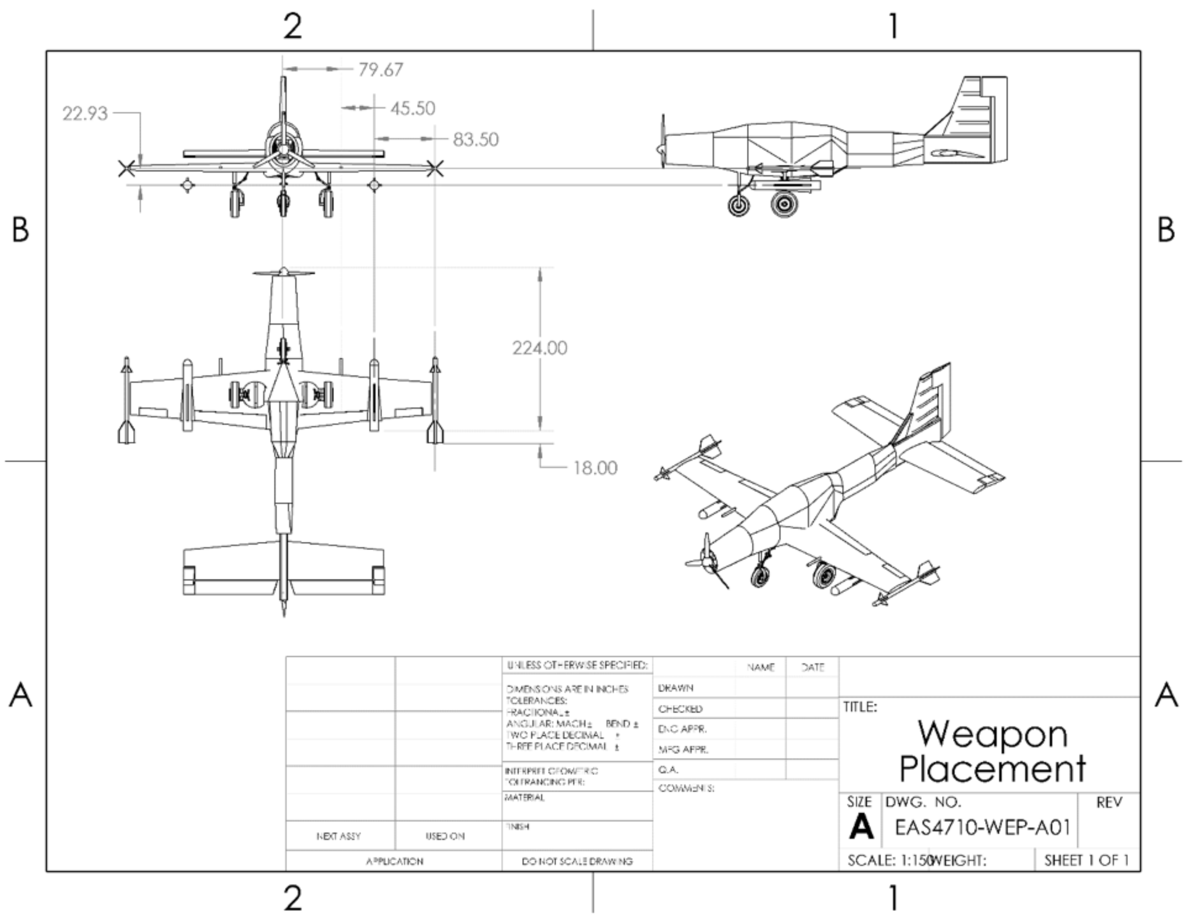
- Guns
  - 1 FN Herstal M3P Machine Gun
    - Mounted under the fuselage
  - 2 GIAT M20A1 podded guns
    - Integrated into the wings



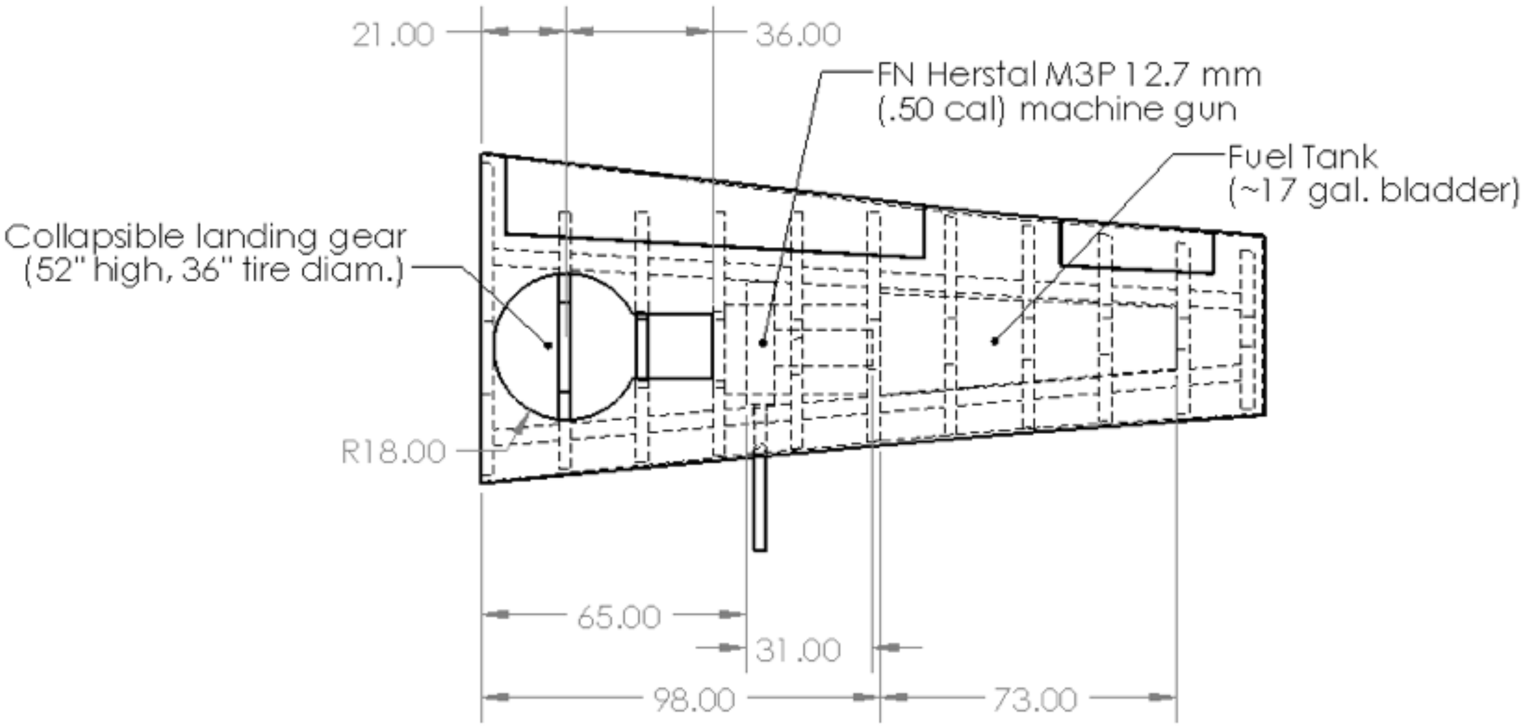
Weapon Carriage Selection

- AIM-9L Side winder
  - Rail launch method
- AGM-65 Maverick
  - Ejection method
    - Due to large weight

Placement of Weapons, Landing gear and Fuel tanks



Placement of Weapons, Landing gear and Fuel tanks



Stealth Features

- Slender Fuselage to reduce radar detectability
- Turboprop exhaust reduces IR detectability
- Light grey/blue paint on underside to blend into sky
- Turboprop creates less noise than other engines

Vulnerability Considerations

$$Vulnerable\ Area = (Projected\ Area) \times P_k$$

Component	Projected Area [ft^2]	Probability Given Hit, P_k	Vulnerable Area [ft^2]
Pilot	5	1	5
Computer Systems	4	0.5	2
Engine Blade	1.96	0.4	0.784
Wing	532.7	0.5	266.4
Fuselage	557.6	0.3	167.3
Cockpit	52.25	0.4	20.9

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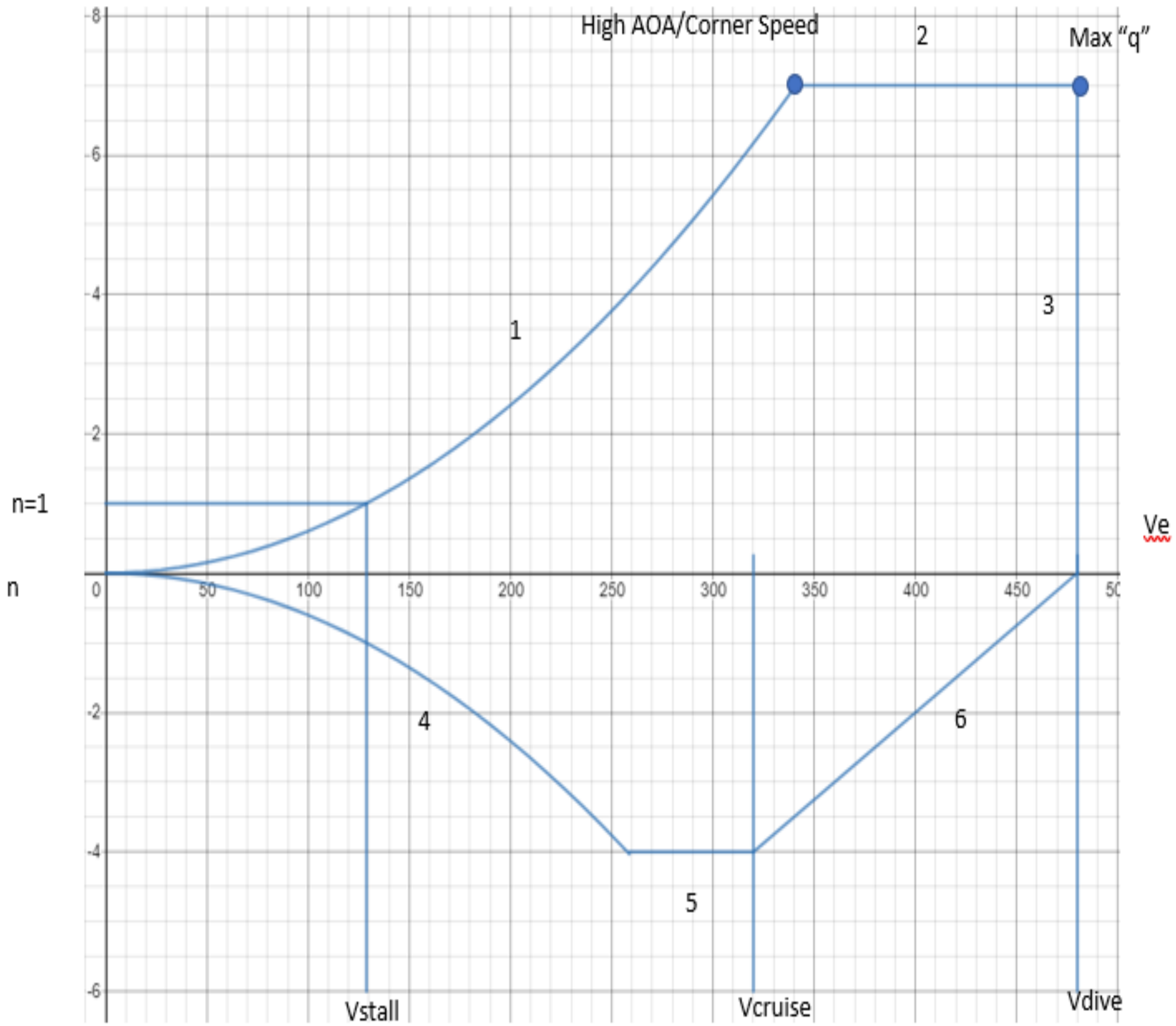
Structure

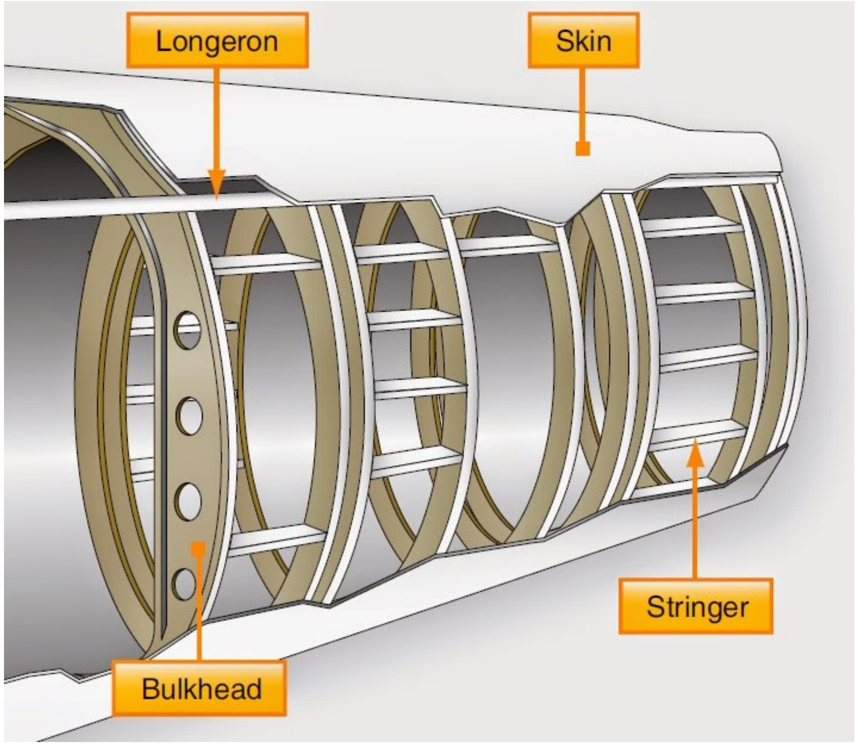
V-n Diagram

1. $n_{max} = \frac{1}{2}\rho V^2 \frac{(C_L)_{max}}{W/S}$
2. Peak load factor, high α
3. High-speed limit (dive)
4. $n_{max} = \frac{1}{2}\rho V^2 \frac{(C_L)_{max}}{W/S}$
5. Cruise speed

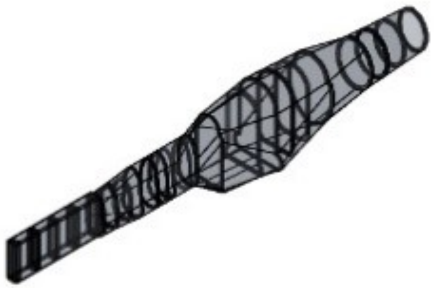


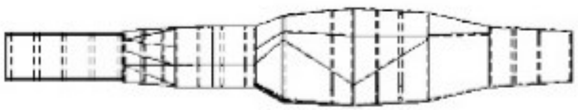
Fuselage Structure

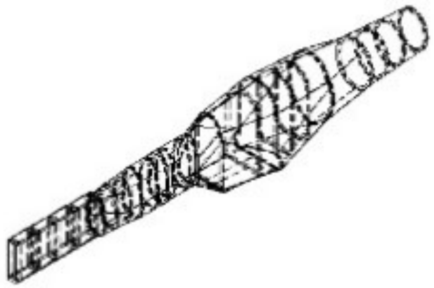
- Semi-monocoque
  - Load-bearing cross-sectional frames (bulkheads)
  - Lengthwise/axial linkages (longerons)
  - Aluminum shell
  - Members along shell interior (stringers)
- Cost-effectiveness
- Weight reduction
- Fuel efficiency



Fuselage Frame

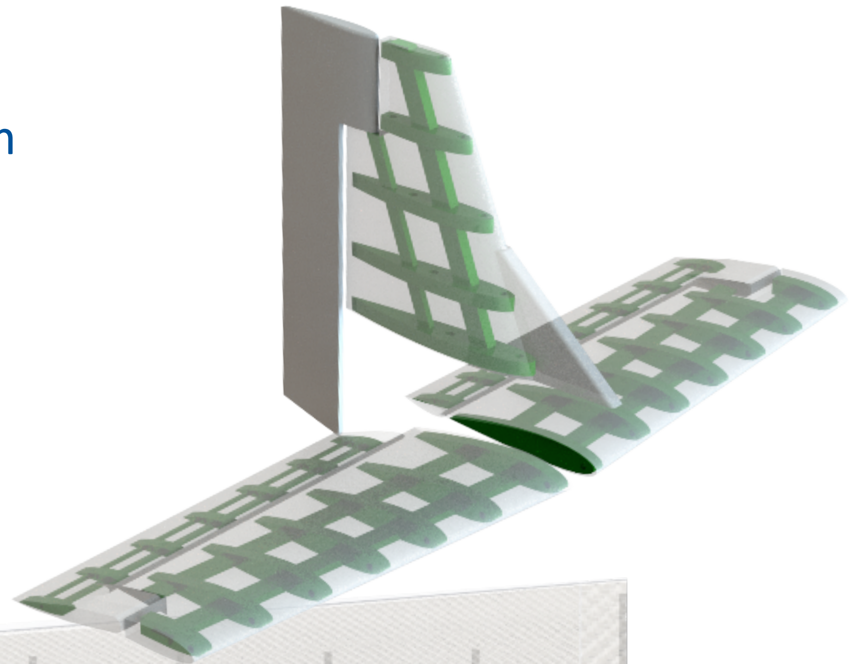


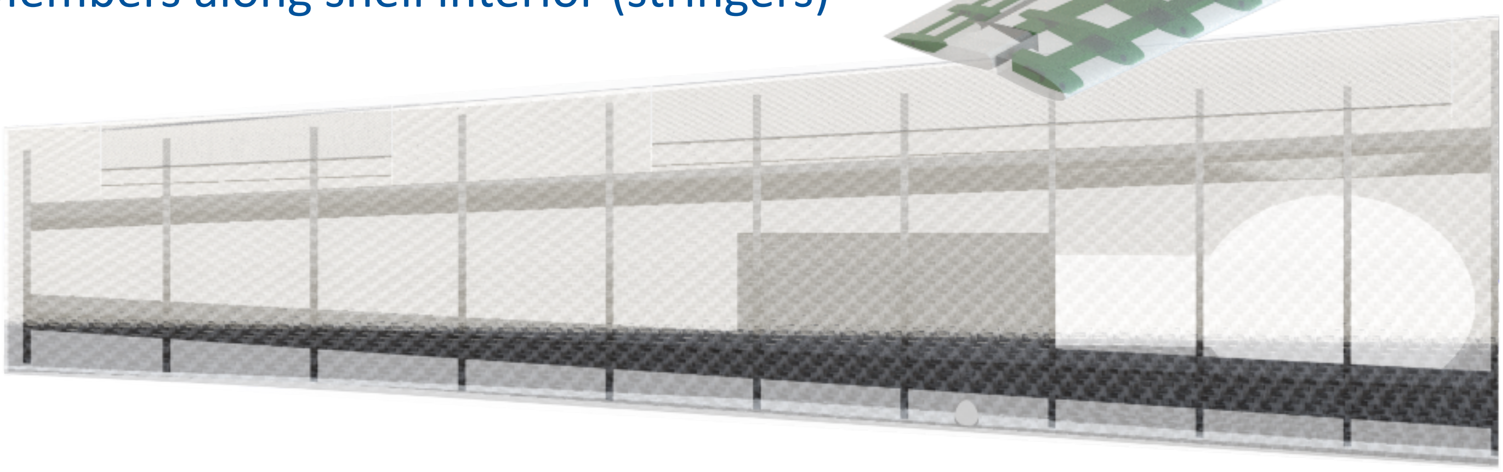




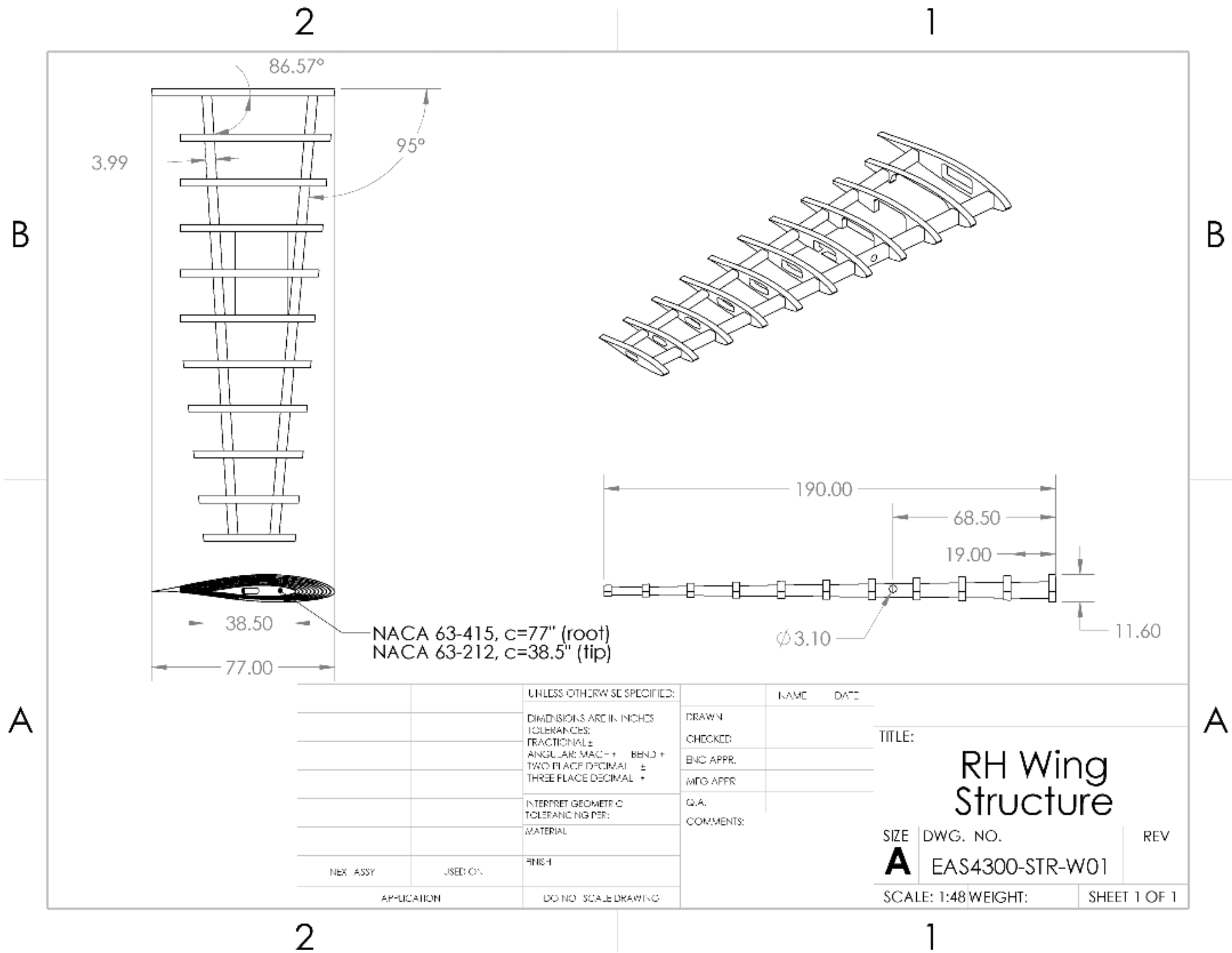
Wing Structure

- Cross-sectional members (ribs); aluminum
- Spanwise members/linkages (spars)
- Composite laminate/sandwich shell; carbon-fiber
- Members along shell interior (stringers)

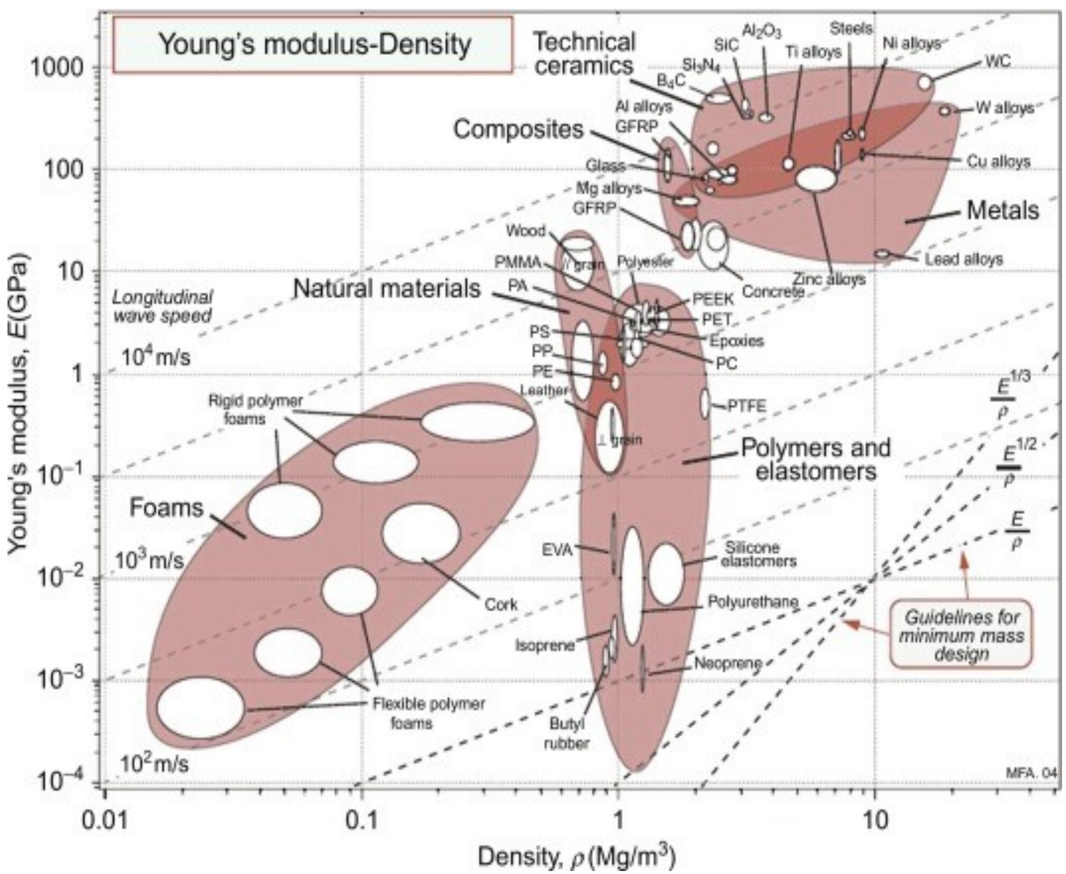




Wing Structure



Material Selection

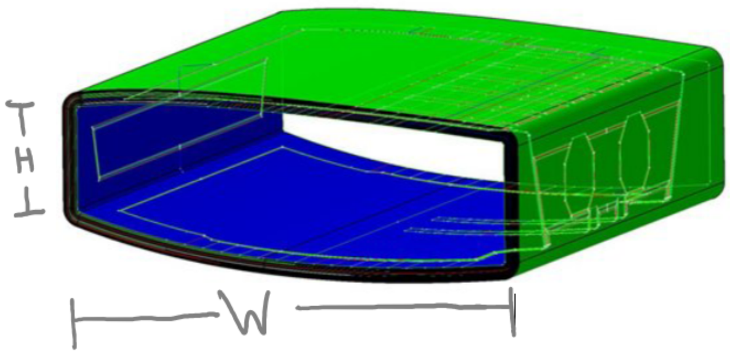


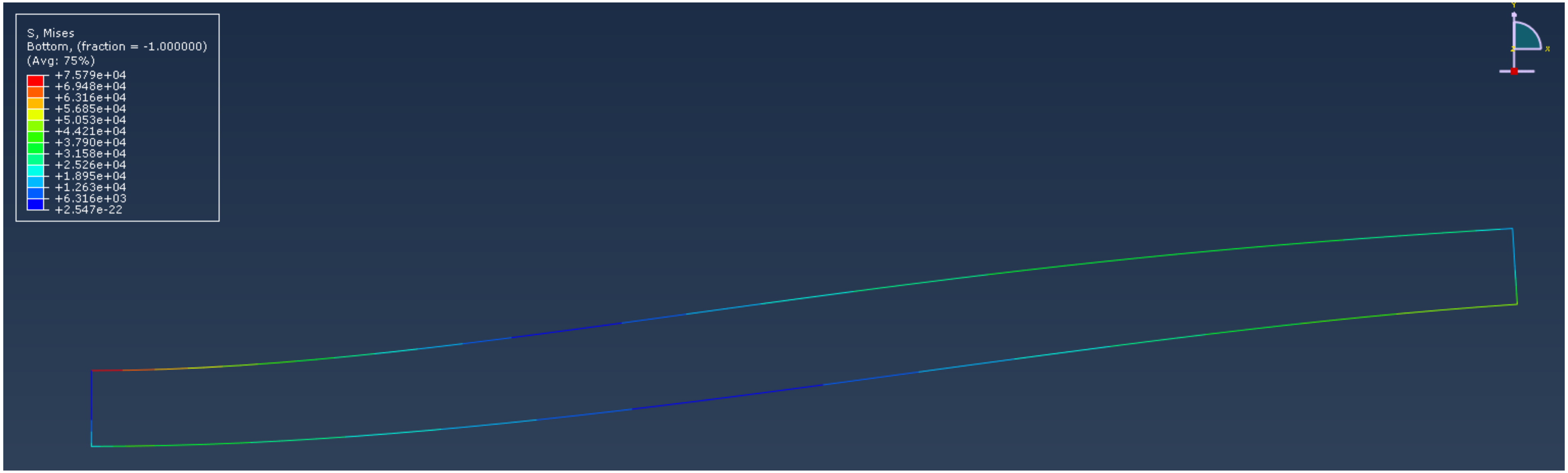
Property	Aluminum (7050-T7451)[a]	Titanium (6Al-4V)[a]	Steel (PH13-8Mo)[a]	Carbon/ Epoxy (IM7/ 977-3)[a]	Carbon/ BMI (IM7/ 5250-4)[a]	Carbon/ Thermo-plastic (IM7/PEEK)[a]
Density (lb./sq in.)	0.102	0.160	0.279	0.057	0.056	0.058
Strength (KSI)	70	134	201	332	349	323
Stiffness (MSI)	10.3	16.0	28.3	22.2	22.2	22.7
Specific strength (K in.)	685	840	720	5825	6230	5570
Specific stiffness (M in.)	100	100	100	390	395	390
Service tempera-ture (degrees F)	250	450	1000	275	325	275

[1]

Stress Analysis

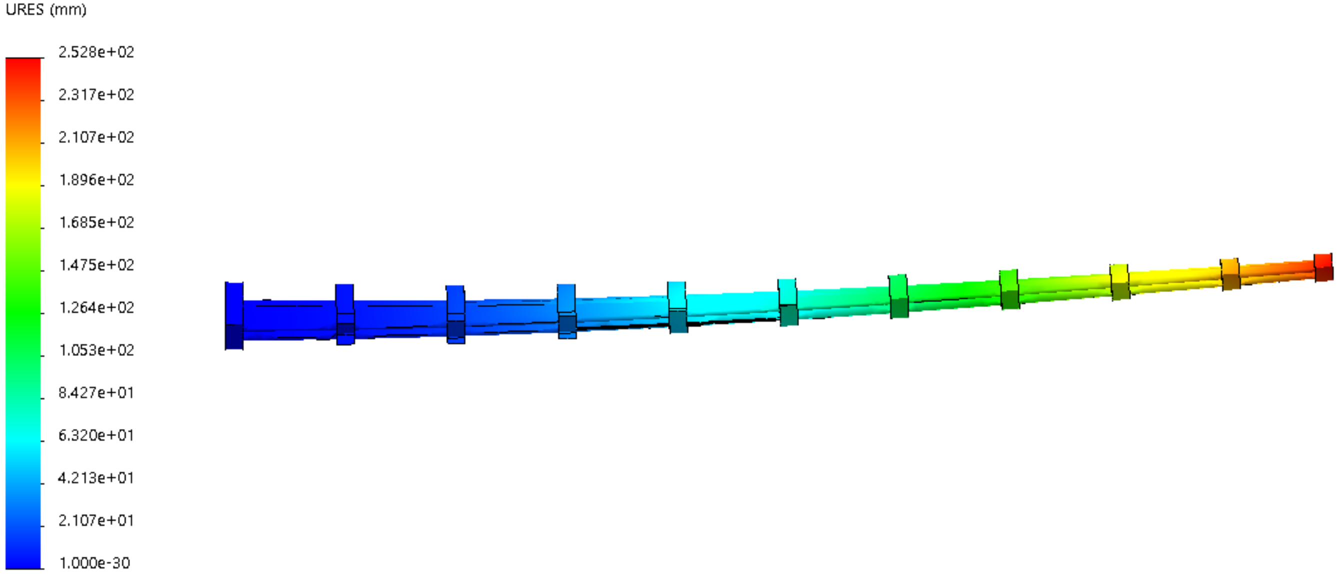
- Wing box simplification
- Cantilever model constraints
- Von-Mises stress varies between 0 and 44.2 ksi; does not exceed yield





Displacement Analysis

- Interior structure subjected to max lift conditions
- Omission of shell/skin components
- Cantilever simplification
- Tip displacement does not exceed 8 inches



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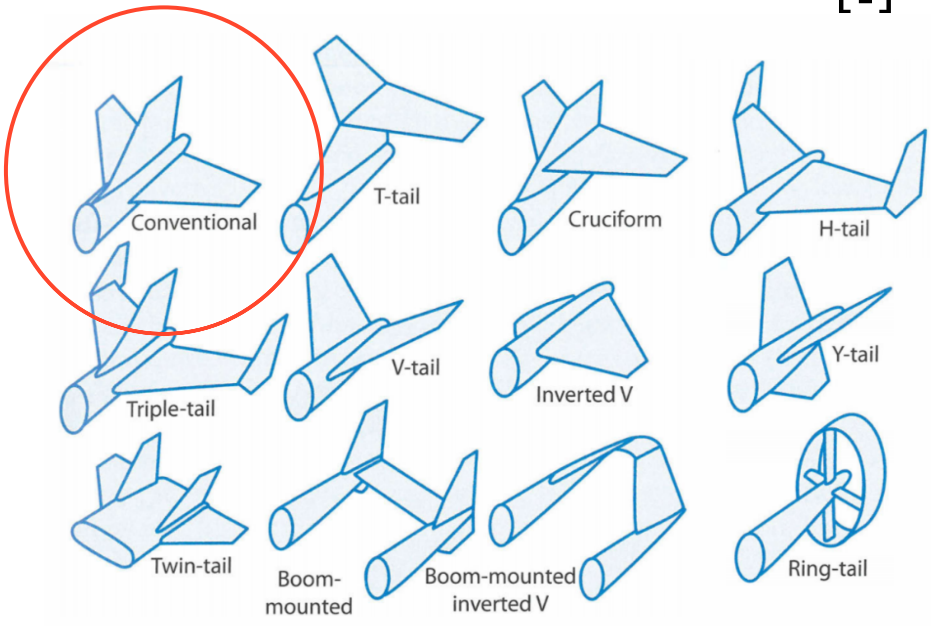
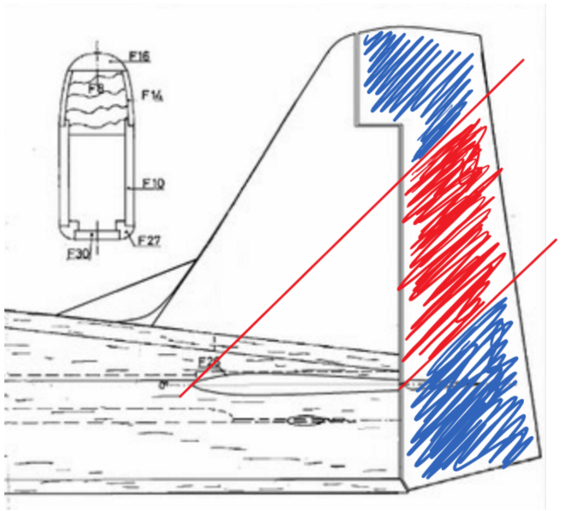
Tail Design, Systems, Weight

Tail Arrangement

- No unique stability considerations
- Weight optimization
- Spin recovery consideration
  - At least 1/3 of rudder area unblanketed by turbulent wake from horizontal tail

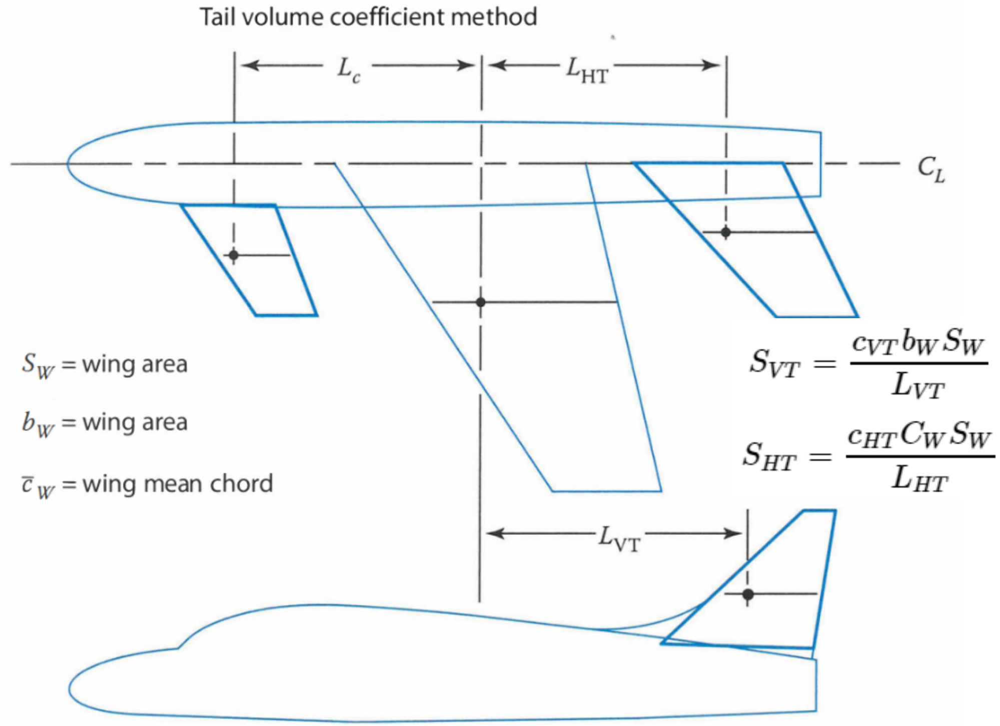
[1]





Tail Geometry

- H/V volume coefficients of 0.7 and 0.06
- Tail arm: 60% of fuselage length
- Sweep angle of 20
- AR = 1.2
- Taper ratio λ = 0.4



[1]

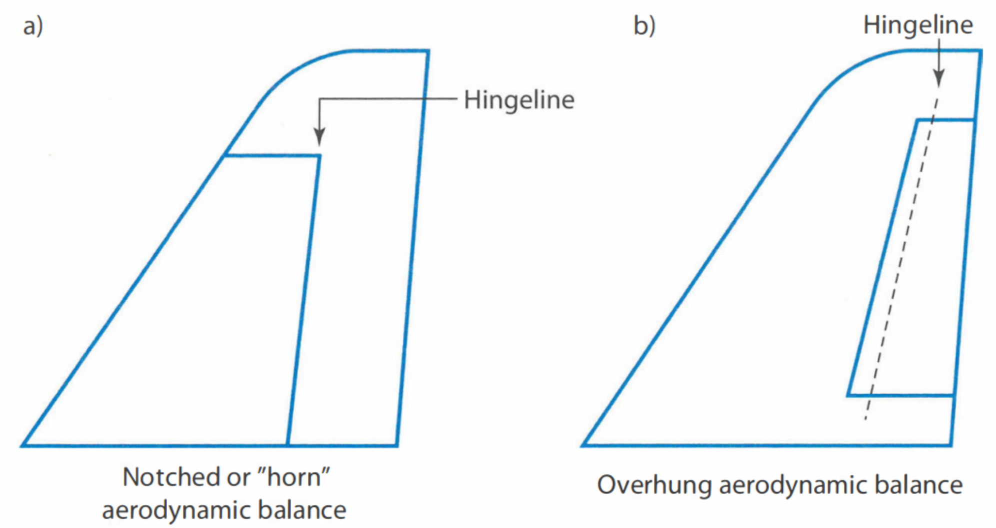
	Horizontal Tail		Vertical Tail	
	A	λ	A	λ
Fighter	3–4	0.2–0.4	0.6–1.4	0.2–0.4
Sailplane	6–10	0.3–0.5	1.5–2.0	0.4–0.6
Others	3–5	0.3–0.6	1.3–2.0	0.3–0.6
T-tail	–	–	0.7–1.2	0.6–1.0

[1]

Control Surface Sizing

[1]

- Notched balance
- Elevator area: 35% of horizontal tail
- Rudder area: 35% of vertical tail

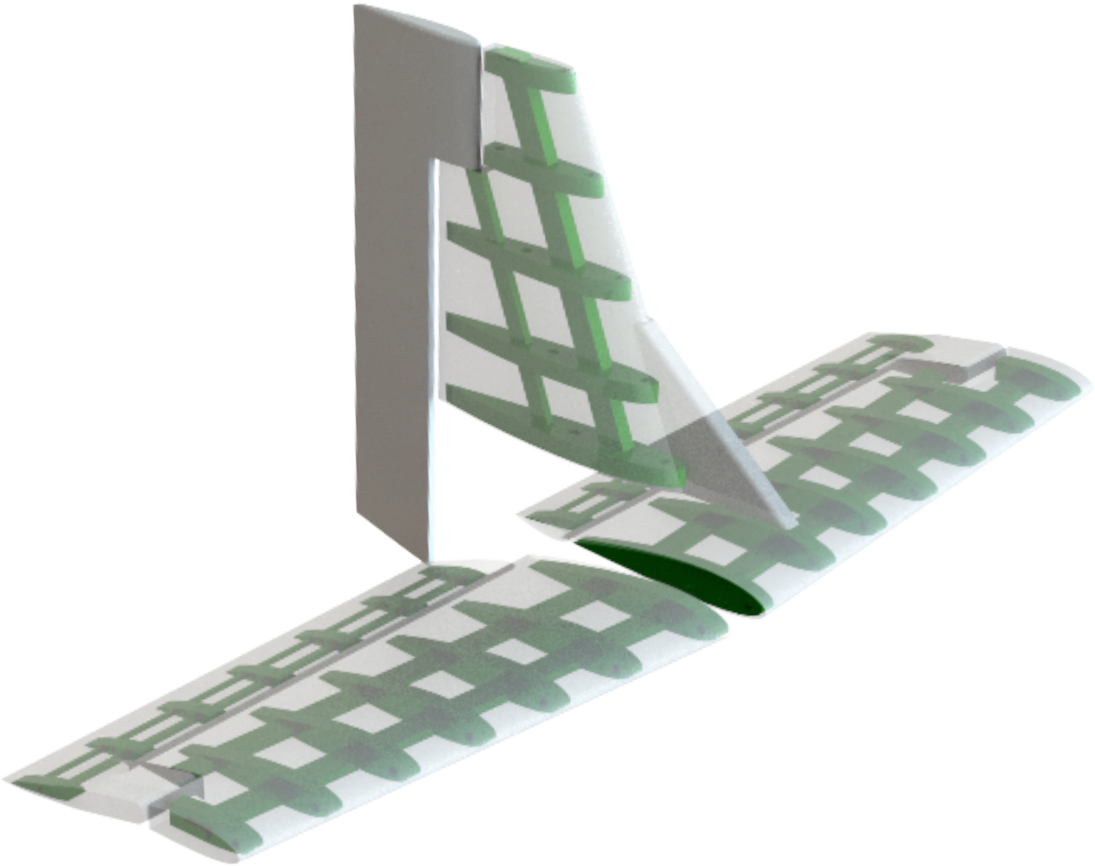


Aircraft	Elevator C_e/C	Rudder C_r/C
Fighter/attack	0.30*	0.30
Jet transport	0.25†	0.32
Jet trainer	0.35	0.35
Biz jet	0.32†	0.30
GA single	0.45	0.40
GA twin	0.36	0.46
Sailplane	0.43	0.40

*Supersonic usually all-moving tail without separate elevator.
†Often all-moving plus elevator.

[1]

Tail Structure



System Weight Estimation

[1]

- Controls: 530 lbs
- Instruments: 128 lbs
- Avionics: 632 lbs
- Electrical: 447 lbs
- Cooling: 38 lbs
- Furnishing: 435 lbs
- A/C and de-icing: 230 lbs

(Consolidated in crew-station model)

$$W_{flight\ controls} = 36.28 M^{0.003} S_{cs}^{0.489} N_s^{0.484} N_c^{0.127}$$

$$W_{instruments} = 8.0 + 36.37 N_{en}^{0.676} N_t^{0.237} + 26.4(1 + N_{ci})^{1.356}$$

$$W_{avionics} = 2.117 W_{uav}^{0.933}$$

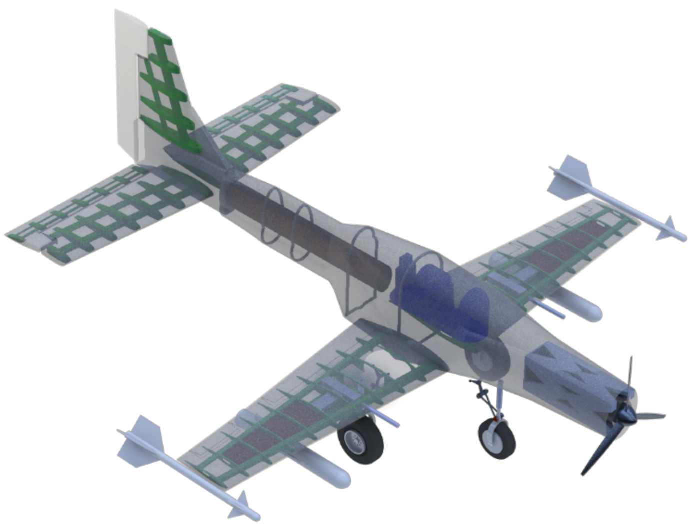
$$W_{electical} = 172.2 K_{mc} R_{kva}^{0.152} N_c^{0.1} L_a^{0.1} N_{gen}^{0.091}$$

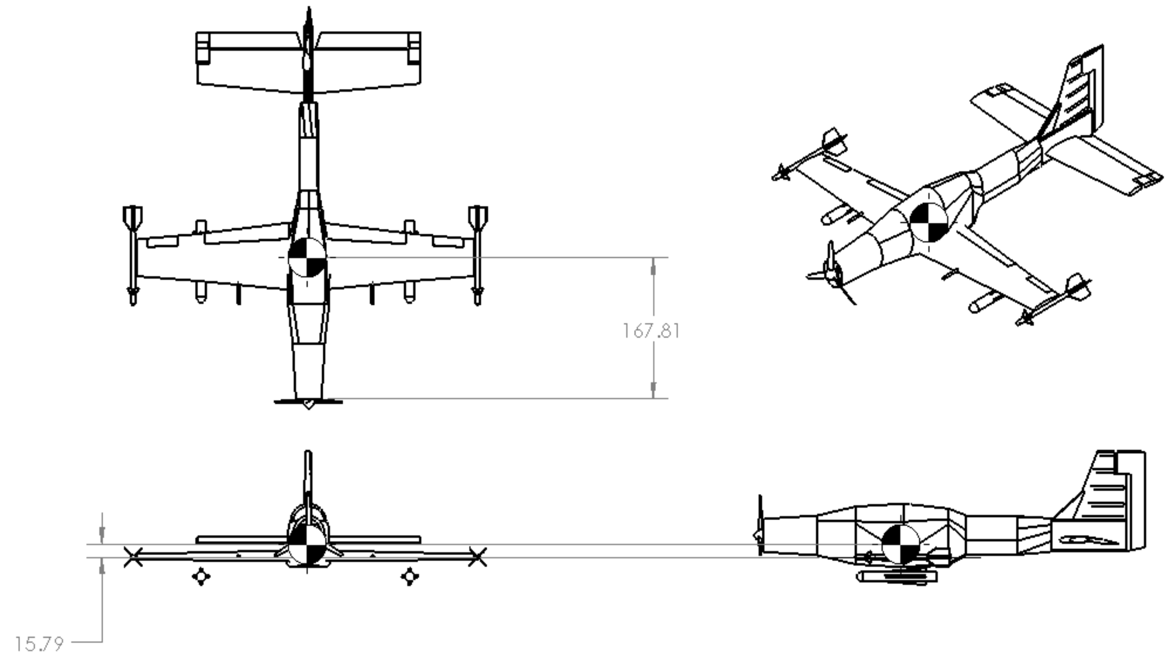
$$W_{oil\ cooling} = 37.82 N_{en}^{1.023}$$

$$W_{furnishing} = 217.6 N_c$$

$$W_{air\ conditioning\ and\ anti-ice} = 201.6 \left[\frac{W_{uav} + 200 N_c}{1000} \right]^{0.735}$$

Overall Weight/CG Estimation





Empty Weight: 8,754 lbs

Design Mission: 12,742 lbs

x_{CG}	Axial location of CG (from nose)	167.8 in
y_{CG}	Lateral location of CG (from fuselage axis)	0.0 in
z_{CG}	Vertical location of CG (from wing chord, positive through bottom of fuselage by convention)	-15.8 in

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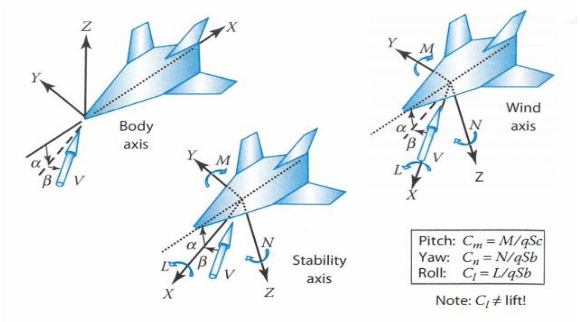
Stability

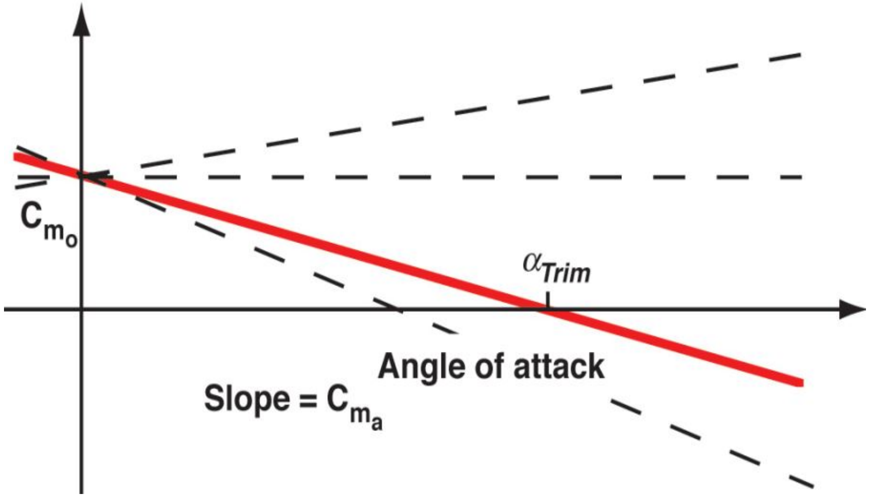
Stability Analysis

Longitudinal Stability

- Concerned with aircrafts pitching stability under steady-level flight
- Aircraft must generate moments that oppose any change in angle of attack such as wind perturbations
- The change in pitching moment with respect to angle of attack must be negative



Stability Axis and Direction



Pitching moment vs alpha

Pitching Moment Contributions

- Wing
- Fuselage
- Horizontal Tail
- Engine

Pitching Moment Arms

- Center of gravity is measured from referenced datum, nose
- Aerodynamic center is measured using thin airfoil theory
- $x_{ac} = x_{c/4} + \Delta x_{ac}\sqrt{S_{wing}}$
- $\Delta x_{ac} = 0.26(M - 0.4)^{2.5}$

Chord Length and Aerodynamic Center for wings

Component	Chord [m]	$x_{c/4}$ [m]	S_{ref} [m²]	x_{ac} [m]
Wing	2.032	0.508	7.439	0.521
Horizontal Tail	1.833	0.458	5.725	0.469

Moment Arms Measured from Datum

$\overline{X}_{cg}$ [m]	$\overline{X}_{acw}$ [m]	$\overline{X}_{ach}$ [m]	$\overline{X}_{cg} - \overline{X}_{acw}$ [m]	$\overline{X}_{ach} - \overline{X}_{cg}$ [m]	$\overline{X}_{cg} - \overline{X}_{p}$ [m]
4.262	3.817	9.677	0.446	5.415	4.262

Derivative of pitching moment with respect to angle of attack

- $C_{m_\alpha} = C_{L_\alpha}(\bar{X}_{cg} - \bar{X}_{acw}) + C_{m_{\alpha\,fus}} - \eta_h \frac{S_h}{S_w} C_{L_{\alpha_h}} \frac{\partial \alpha_h}{\partial \alpha}(\bar{X}_{ach} - \bar{X}_{cg}) + \frac{F_{p\alpha}}{qS_w}\frac{\partial \alpha_p}{\partial \alpha}(\bar{X}_{cg} - \bar{X}_p)$
- $C_{m_\alpha} = -0.376$; Ensures longitudinal static stability

Variable	Value	Units
C_{m_α}	-0.3757	
C_{L_α}	1.488	
$\bar{X}_{cg}$	4.262	[m]
$\bar{X}_{acw}$	3.818	[m]
$C_{m_{\alpha\,fus}}$	5.192E-3	
η_h	0.9	
S_h	5.725	[m^2]
S_w	19.4	[m^2]
$C_{L_{\alpha_h}}$	1.239	
$\frac{\partial \alpha_h}{\partial \alpha}$	0.6	
$\bar{X}_{ach}$	9.677	[m]
$F_{p\alpha}$	430.210	[N]
q	4230.965	[pa]
S_w	19.4	[m^2]
$\frac{\partial \alpha_p}{\partial \alpha}$	1.2	
$\bar{X}_p$	0	[m]

Neutral Point and Static Margin

Neutral Point

- Occurs when derivative of pitching moment is zero
- $\bar{X}_{NP} = \dfrac{C_{L\alpha}\bar{X}_{acw} - C_{m_{\alpha\,fus}} - \eta_h \frac{S_h}{S_w} C_{L\alpha_h} \frac{\partial \alpha_h}{\partial \alpha}(\bar{X}_{ach}) + \frac{F_{p\alpha}}{qS_W}\frac{\partial \alpha_p}{\partial \alpha}(\bar{X}_p)}{C_{L\alpha} + \eta_h \frac{S_h}{S_w} C_{L\alpha_h} \frac{\partial \alpha_h}{\partial \alpha} + \frac{F_{p\alpha}}{qS_W}\frac{\partial \alpha_p}{\partial \alpha}}$

Variable	Value	Units
$C_{L\alpha}$	1.488	
$\bar{X}_{acw}$	3.817	[m]
$C_{m_{\alpha\,fus}}$	5.192E-3	
η_h	0.9	
S_h	5.725	[m²]
S_w	19.4	[m²]
$C_{L\alpha_h}$	1.239	
$\frac{\partial \alpha_h}{\partial \alpha}$	0.6	
$\bar{X}_{ach}$	9.677	[m]
$F_{p\alpha}$	430.210	[N]
q	4230.965	[pa]
$\frac{\partial \alpha_p}{\partial \alpha}$	1.2	
$\bar{X}_p$	0	[m]

Static Margin

- Distance between center of gravity and neutral point
- Positive static margin yields a stable aircraft since the center of gravity is ahead of the neutral point
- $Static\ Margin = \bar{X}_{NP} - \bar{X}_{cg}$

Parameter	Value
$\bar{X}_{NP}$	4.484 [m]
$Static\ Margin$	0.222 %

Lateral-Directional Stability

Lateral Stability

- Affected by roll and yaw angle
- Negative rolling-moment derivative with respect to side slip angle is stabilizing, dihedral effect
- $C_{n_\beta} = C_{n_{\beta_w}} + C_{n_{\beta_{fus}}} + C_{n_{\beta_v}} - \frac{F_{p_\beta}}{q S_w} \frac{\partial \beta_p}{\partial \beta} (\bar{X}_{cg} - \bar{X}_p)$
- $C_{l_\beta} = C_{l_{\beta_w}} + C_{l_{\beta_v}}$

Yaw Moment Coefficient Values

Parameter	Value
$C_{n_{\beta_w}}$	-0.02072
$C_{n_{\beta_{fus}}}$	-0.005607451
$C_{n_{\beta_v}}$	0.129069562
C_{n_β}	0.10028

Roll Moment Coefficient Values

Parameter	Value
$C_{l_{\beta_w}}$	-0.08976
$C_{l_{\beta_v}}$	-0.03175133
C_{l_β}	-0.121511328

Trim Analysis

Static Trim

- Total pitching moment must be zero in steady-level flight
- $C_{m_{cg}} = C_L\left(\bar{X}_{cg} - \bar{X}_{acw}\right) + C_{m_w} + C_{m_{w\delta f}}\delta_f + C_{m_{fus}} - \eta_h \frac{S_h}{S_w} C_{L_h}\left(\bar{X}_{ach} - \bar{X}_{cg}\right) - \frac{T}{qS_w}\bar{Z}_t + \frac{F_p}{qS_w}((\bar{X}_{cg} - \bar{X}_p)$

	$\delta_E = -2$ **[deg]**	**0 [deg]**	**2 [deg]**
α = 0 [deg]	$C_{m_{cg}} = 0.0236$	0.0032	-0.0172
	$C_{L_{total}} = -0.0376$	0.0011	0.0376
α = 5 [deg]	$C_{m_{cg}} = 0.0021$	-0.0182	-0.0386
	$C_{L_{total}} = 0.4611$	0.4987	0.5363
α = 10 [deg]	$C_{m_{cg}} = -0.0193$	-0.0397	-0.0601
	$C_{L_{total}} = 0.9598$	0.9974	1.0350

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Performance

Range Analysis

Calculated with the Breguet Range Equation

- Maximum Range (Internal Fuel Supply): *748 [nmi]*
- Maximum Range (With External Fuel Tanks): *1447 [nmi]*

- Target Range (Design Mission): *200 [nmi]*
- Target Range (Ferry Mission): *900 [nmi]*

Endurance

Estimated using the Total Endurance Equation

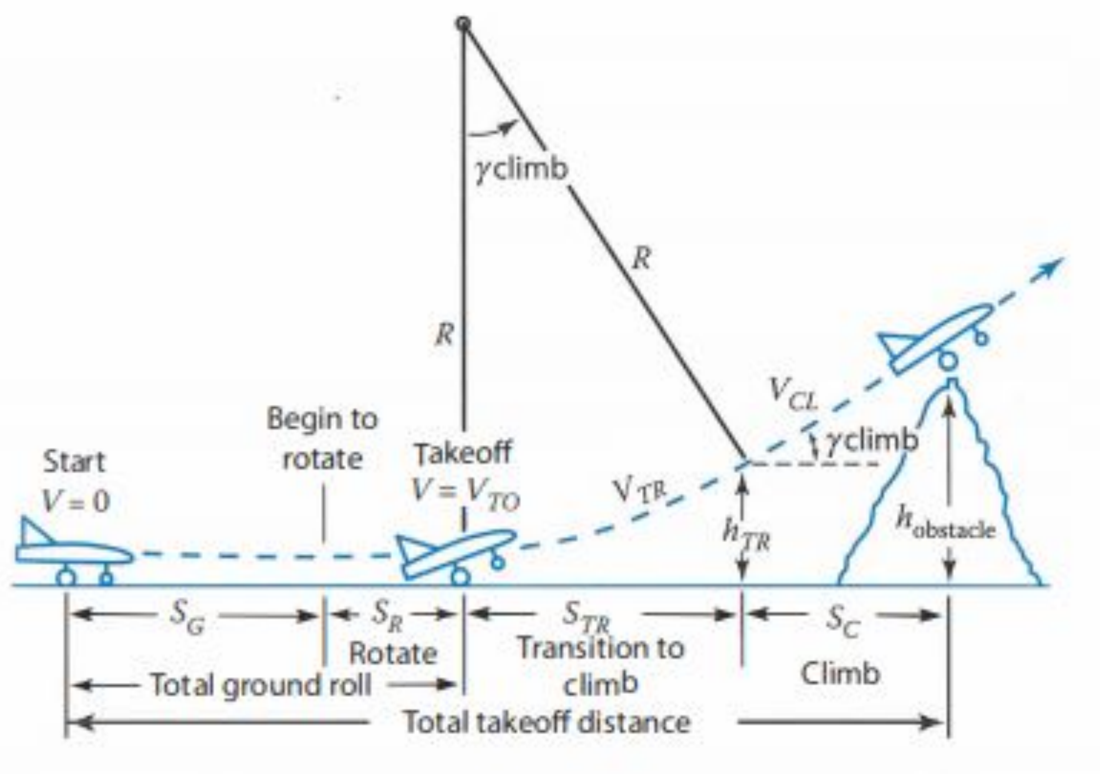
- Maximum Endurance (Internal Fuel Supply): *2.69 [hr]*
- Maximum Endurance (With External Fuel Tanks): *5.21 [hr]*

- Target Endurance: *4.00 [hr]*

Takeoff Distance

The takeoff distance is the sum of three intermediary quantities

- Ground Roll Distance
- Transition to Climb Distance
- Climb DIstance



Takeoff Distance

The landing distance was calculated using the approach angle, stall velocity and the height of the obstacle.

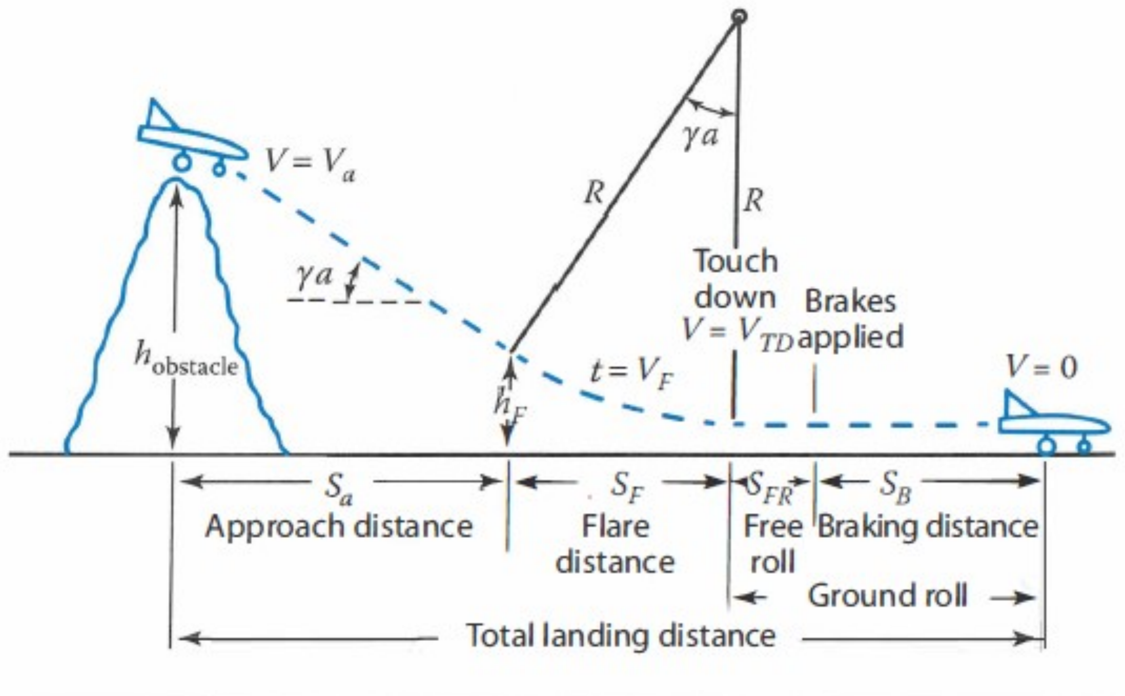
Distance Interval	Value
Ground Roll	670 [ft]
Transition to Climb	827 [ft]
Climb	244 [ft]

Total Estimated Takeoff Distance: *1741 [ft]*

Landing Distance

Like the Takeoff Distance, the Landing Distance is calculated as the sum of several intermediary quantities

- Approach Distance
- Flare Distance
- Ground Roll Distance



Landing Distance

The landing distance was calculated using the approach angle, stall velocity and the height of the obstacle.

Distance Interval	Value
Approach	216 [ft]
Flare	294 [ft]
Ground Roll	1722 [ft]

Total Estimated Landing Distance: *2232 [ft]*

Summary of Performance Analysis

- Maximum Range (Internal Fuel Supply): *748 [nmi]*
- Maximum Range (With External Fuel Tanks): *1447 [nmi]*
- Maximum Loiter Time: *5.21 [hr]*
- Runway Takeoff Distance: *1,740 [ft]*
- Runway Landing Distance: *3,424 [ft]*

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Cost Analysis

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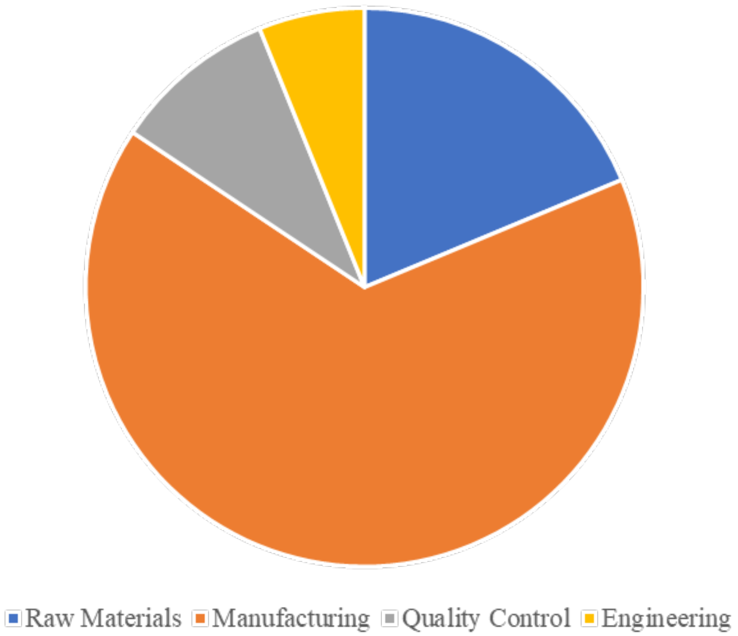
Quick Cost Overview

Per Unit Cost: ~$21.7M

- Manufacturing: *~$14.2M*
- Raw Materials: *~$4M*
- Quality Control: *~$2M*
- Engineering: *~$1.5M*



Non-Recurring Costs

- Separated into four parts
  - Engineering
  - Tooling
  - Quality Control
  - Manufacturing

Hourly Rates	2012 $	Adj $	Hours	Total Cost
Engineering	115	132.25	72598.29607	9601124.656
Tooling	118	135.7	68428.47558	9285744.136
Quality Control	108	124.2	6254735.744	776838179.4

Acquisition Cost

- Procurement Cost of 50 A-55 Vulture aircrafts
- The cost per unit is the sum of all the values divided by 50

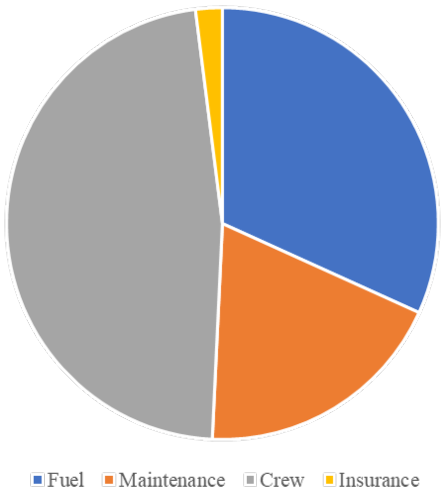
Costs	Dollar Amount
Development- Support Cost, C_D	4,073,616
Flight Test Cost, C_F	20,063,991
Manufacturing Materials Cost, C_M	203,528,974
Engine Production Cost, C_{eng}	33,062,417
Avionics Cost, C_A	50,000
RDTE+flyaway	1,087,894,064
Cost Per Unit	21,757,881.28

Operating Cost

- Separated into four main parts
  - Fuel & Oil Cost
  - Maintenance:
    - Material Cost/flight and per flight hour
    - Tire Replacement
    - Brake Replacement
  - Crew (2-man)
  - Insurance
    - 2% of total operating cost
- $844.908/hr

Per Year Operational Costs: ~$1M

- Fuel: *~$318K*
- Maintenance: *~$190K*
- Crew: *~$473K*
- Insurance: *~$20K*



References

[1] Raymer, Daniel. AIRCRAFT DESIGN : A Conceptual Approach. S.L., Amer Institute Of Aerona, 2019.

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