V-Tailed Transatlantic Business Jet

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Abstract

This paper summarizes the conceptual configuration design of a V-tail business jet aircraft. The T-Tail configuration compared to a conventional horizontal and vertical tailplane configuration, the T-Tail has a unique stall characteristic and is prone to a deep stall. A deep stall occurs when an aircraft pitches up such that the tail gets caught in the low-energy turbulent air from the wings which strips the rudder and elevator of their control authority. If not designed correctly, a deep stall can be unrecoverable and deadly. Contrary to this, a V-Tail configuration is used on Cirrus Vision Jet. The V-Tail configuration is beneficial and has the potential to reduce cabin noise by allowing for rear-mounted engines and produces less drag due to its 2 control surfaces, named ruddervators, that function both as rudders and elevators. The research presented in this paper indicates that the V-Tail configuration has a smaller overall empennage surface area, producing less overall vehicle drag and reducing mass. Using reference aircraft and statistical class 1 aircraft design models for weight estimations, the initial conceptual size of the aircraft was determined. Historical trends and drag divergence at high speeds were used to select a suitable airfoil and to generate the wing planform. During this wing design process, both clean and landing configurations were also taken into consideration. A class II, by-component, weight estimation was done using numerical methods proposed by Raymer. The design of the empennage was determined by looking at the center-of-gravity travel range under different payloads and examining the stability and maneuverability. The aircraft's overall performance was evaluated, and the resulting design can achieve the desired mission profile. The proposed aircraft "Teletubby3400" can carry a maximum capacity of 16 passengers and 4 crew, with 3 customizable cabin configurations allowing potential customers to suit the aircraft to their specific taste. The Teletubby 3400 has a maximum takeoff weight of 34,455 kg with an empty weight of 18,132 kg, a fuel capacity of 14,268 kg, and a cargo capacity of 2180 kg which is above average for typical midsize business jets. The designed aircraft meets specific performance requirements and can take off within 1,650 m and land within 1,100 m at its maximum landing weight. With an initial selection and resizing of the propulsion system, the Teletubby 3400 aircraft is designed to travel up to 6300 km at a cruising speed of 850 km/h at a cruise altitude of 13,700 m. The appropriate FAR part 25 FAA regulations were taken into consideration and confirmed throughout such that the aircraft would be allowed to operate on international airspaces.

Nomenclature

α = Angle of Attack
α = Angle of Attack
AR = Aspect Ratio
C_D = Coefficient of drag
C_L = Coefficient of lift
CoG = Center of Gravity
FAA = Federal Aviation Administration
MAC = Mean Aerodynamic Chord
MTOW = Maximum Takeoff Weight
RE = Reynolds Number
S = Wing Surface A

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Introduction

The process of designing an aircraft is complicated and time-consuming. A barebones process was explored in AESP 415 Design Part 1, Basics. This process is summarized in the following report as a part of the process, ranging from an analysis of basic requirements to designing the wing to calculating an improved estimate of the drag encountered by the aircraft during the cruise.

Mission Profile

The conceptual aircraft is intended as a private jet that can carry cargo and passengers without assistance from ground infrastructure. It can accommodate up to 3 crew members and 16 passengers. The passengers will board through a cabin door, and the luggage will be transported in a pressurized compartment accessible from the passenger cabin. The aircraft will start and taxi without assistance, take off, climb to a cruising altitude of 50,000 feet, and maintain a speed of 850 km/h throughout the climb, cruise, and descent. The cabin will maintain comfortable internal pressure. If necessary, the aircraft will divert to an alternate airport and then land, taxi to the shutdown location, and be unloaded without ground equipment.

Building T/W – W/S Diagram

To achieve the best performance for an aircraft by designing it with a low cost and high range. The T/W (thrust loading) and W/S (wing load) diagrams are used to determine the optimal thrust-to-weight ratio and wing loading for the aircraft.

Stall Sizing

Stall speed is the minimum speed an aircraft can travel to continue generating lift during landing and cruise operations. The weight of the aircraft is a major factor affecting this speed, and the lift equation can be used to determine the landing and clean stall sizing for an aircraft based on its reference values for Clmax and Vs. The resulting W/S values are be plotted on a T/W-W/S diagram.

	Clean	Land	
CLmax	1.8	2.6	
Vstall	51.44	43.73	m/s
W/S	2917.296	3045.358	N/m^2

Table 1

Take-Off Sizing

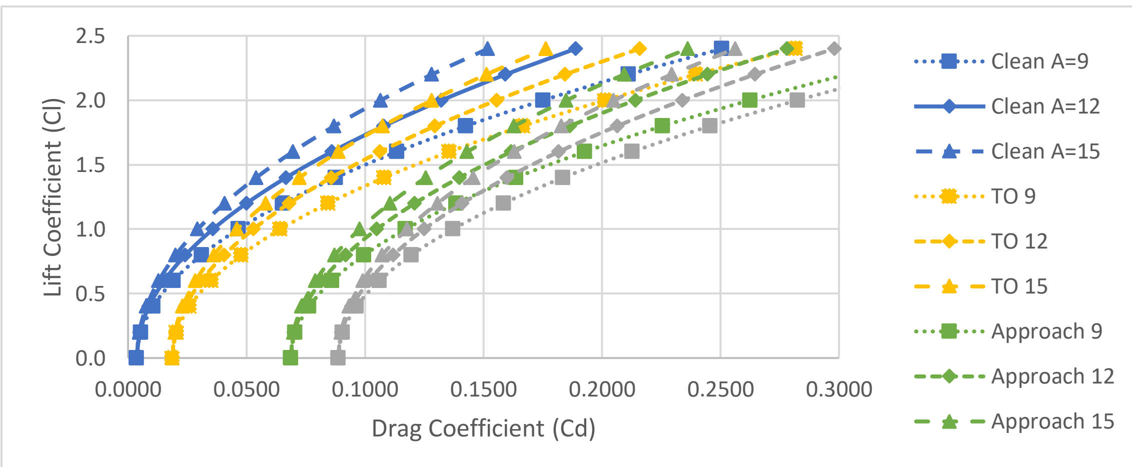
To successfully take-off, an aircraft must meet certain parameters such as take-off weight, take-off speed, and thrust loading. For jet aircraft, T/W and W/S are important variables, and Clmax values range from 1.6 to 2.2 for commercial aircraft. TOP (take-off parameter) combines these elements and can be calculated. By graphing the equation for reference aircraft, TOP values can be found for specific take-off distances. Using a take-off distance of 2100m, the TOP value can be determined.

Landing Sizing

Similar to Take-Off sizing, landing sizing calculations are also driven by airworthiness requirements. Factors such as braking capability, friction, drag, landing weight, wing configuration, and approach speed are considered to ensure safe deceleration of the aircraft to zero speed during landing.

Drag Polar

The drag polar is the relationship between the drag coefficient and lift coefficient of an aircraft. It is used to determine compliance with climb requirements. CD is calculated using equation 16, where CDo = f/Swet, with f being the parasite drag and Swet being the wetted area of the wing. CD is calculated at 4 different configurations (clean, take-off, approach, and landing), with each configuration having more drag due to the extension of high-lift devices and undercarriage. The calculated drag polar values are shown in Figure 2. Climb performance requirements are determined by the aircraft's ability to reach a certain altitude in a given amount of time. Climb gradient is calculated for maximum L/D, which results in the optimum climb gradient value for the aircraft.



T/W – W/S

Diagrams

After getting the necessary data from the MTOW, payload, fuel, and operational empty weight, we can assemble the T/W – W/S diagram. This graph is shown in Appendix A. All important values from the T/W-W/S diagram are compiled in Table 3.

Parameters	Value	Unit
$C_{j\ cruise}$	0.55	lb/lbf/hr
$C_{j\ loiter}$	0.45	lb/lbf/hr
$C_{j\ climb}$	-	lb/lbf/hr
W_{to}	34,455	kg
W_{OE}	18,007	kg
W_F	14,268	kg

W_{Pl}	2,180	kg
S_{wing}	11.33	m^2
A	12	-
e	0.825	-
climb gradient c/V	20	m/s
ρ_{cruise} (13,700 m)	0.2381	kg/m^3
engine failure (c/V)	-	-
cruise velocity	236.11	m/s
$N_{engines}$	2	-
C_{Do}	0.004	-

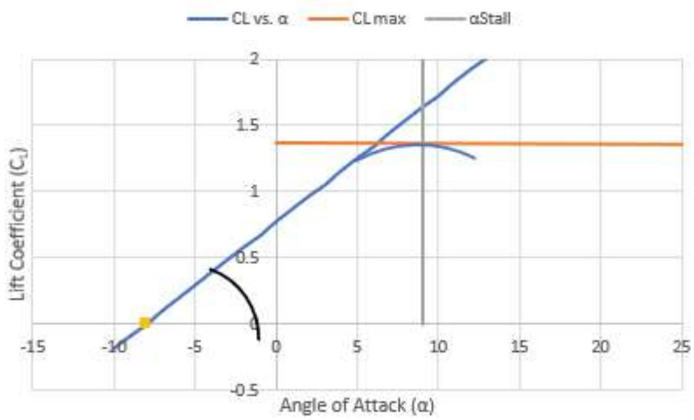
Table 21

The T/W-W/S diagram reveals that the project aircraft's wing loading (W/S) will be 3,046 N/m2, and its thrust loading (T/W) will be 0.3919. Based on a maximum take-off weight of 337,888 N, the aircraft's wing area will be 11.33 m2. Furthermore, this information indicates that the aircraft will necessitate 132,416 N of thrust.

Clean Wing Lift Curve

To determine the clean wing lift curve, we need to find the slope angle, stall angle, and trim angle. The Prandtl-Glauert Compressibility Factor is used to calculate the slope angle for cruise and approach speeds. CLα is then determined using the aspect ratio, wing efficiency factor, and half-chord sweep angle. The trim angle is calculated using the design lift coefficient and curve slope angle. The taper ratio is checked to determine the stall angle calculation method, and the sharpness of the leading edge is also considered.

The C_{Lmax}/C_{lmax} ratio was found to have a value of 0.77 and $\delta\alpha C_{Lmax}$ of roughly 3°. determine C_{Lmax} with δC_{Lmax} equal to 0 as this is at low speeds. This yields a C_{Lmax} value of 1.37. Now, all values in Equation 18 are known which yields an α_s value of 9.57°.



R_E = 6,300,000

α_{0L} = -8°

α_{stall} = 9°

$C_{L\alpha}$ = 0.096 /degree

Figure 2

High Lift Devices
The clean wing's maximum coefficient of lift is 1.37, but during landing and take-off, a maximum lift coefficient of 2.5 and 2.0, respectively, is needed. High-lift devices chosen are Double Slotted Fowler Flaps on the Trailing Edge and Slats on the Leading Edge. The sizing method for the trailing edge devices is detailed in Table 4 and uses Equation 20. The MAC is used to determine the location of the trailing edge devices, with a hinge line location of 70% of the airfoil chord. The value of Swf/S is found to be 0.607, and the ailerons will have a chord equal to 0.25 of the wing. The total CLmax at landing becomes 2.63, and at take-off, CLmax becomes 2.38. For the leading edge devices, the front wing spar is placed at 20% of the wing chord, and the leading edge high-lift device hinge line is placed at 15% of the wing chord. The entire leading edge will be equipped with slats, resulting in an Swf/S value of 0.64, and ΔCLmax values of 0.85 and 0.68 on landing and take-off, respectively.

Evaluating the CL vs. α Curve for the Flapped Wing

Parameters	Clean Value	Take off		Landing Value	
**********	**********	Flaps	Flaps+Slats	Flaps	Flaps+Slats
C_{Lmax}	1.37	2.38	3.06	2.63	3.48
$C_{L\alpha}$	0.079	0.0815		0.0919	
α_{stall}	9.57	23.74		21.16	
α_{0L}	-8	-13.838		-16.757	

Table 3

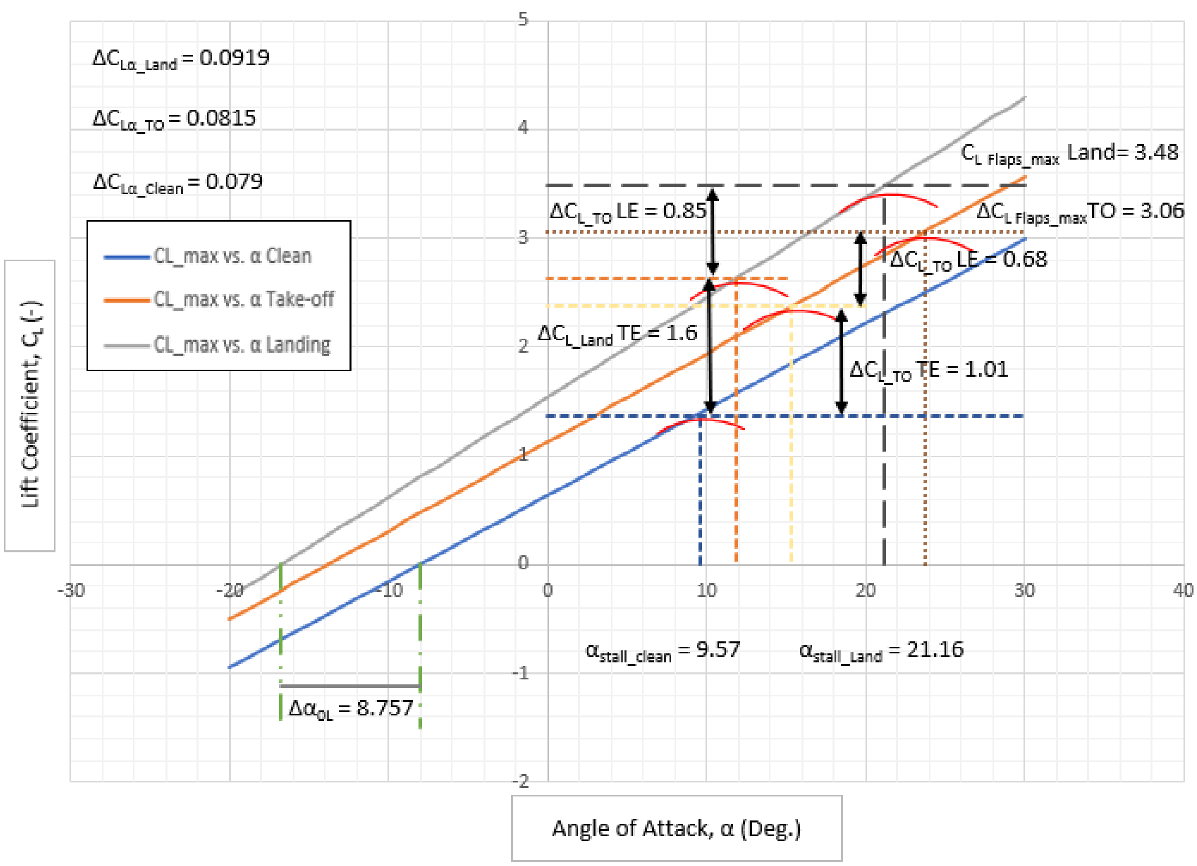


Figure 2

Tail Plain Geometry
Up until this point we have calculated everything needed for our aircraft to fly, and up until this point we always estimated our tail wings, but not anymore. To accurately calculate the stability of the aircraft, we will use the V-bar method in this section to do horizontal and vertical tailplane sizing.

Volume coefficients
The V-bar method is an early preliminary sizing design process for the tails where we can start measuring the weight and aerodynamic groups of the tails.

After studying the reference aircraft data we selected the Bombardier Challenger 601 and the Dassault Falcon 50 as they are the most similar to our aircraft. Since our aircraft is slightly larger and configured differently from standard business jets, we have decided to average both aircraft's data to get a more accurate reference to design the tail volume coefficients. The averaged, individual values can be found below.

Aircraft	Vh Value	Vv Value
Challenger 601	0.67	0.083
Falcon 50	0.68	0.064
Average concerning Our Aircraft	0.7963	0.08728

Table 4

The vertical tail
By using the average of the reference aircraft, we have estimated to find the vertical tail surface. The following surfaces have been found.

Aircraft	Surface Area (S)
Challenger 601	450
Falcon 50	495
Average concerning Our Aircraft	556

Table 5

Since our aircraft is bigger than the reference aircraft, making an estimation reference relative to our aircraft we get a surface of 13.5 Using Equation 1 the two-quarter points of the aerodynamic chord's length can be found.

We are determining the rest of the geometry with more parameters that we have estimated or found from lecture notes. The values for the quarter chord sweep angle, aspect ratio, taper ratio, and thickness ratio are found in previous assignments. The sweep angle is chosen as 40 degrees as we want to make it align with the horizontal sweep angle.

Aircraft	Sv
Challenger 601	8.91
Dassault Falcon 50	9.85
Average concerning Our Aircraft	11.05

Table 6

We have the thickness ratio as 0.13 as it is 1 percent smaller than postponed shock-induced separation. Now we can calculate the length of the root and tip chord from the taper ratio and aspect ratio, we can calculate the span as follows:

Jet T-tail Aircraft (Vertical Tail) parameters	
Aspect Ratio	1.2 to1.5
Taper Ratio	0.6 to 0.8
Stabilizer Sweep Angle	35-45 degree

Table 7

We can find the average height as 9.78 from the reference aircraft. From the following equations, we can find the root and tip chord.

Solving for Cr and Cr

The mean aerodynamic chord is as follows.

The horizontal tail

In the same format, as we did for the vertical, we can size the horizontal tail. Again, since our aircraft is bigger than the reference aircraft, making an estimation reference relative to our aircraft we get a surface of 11.0481. Using Equation 7 the two quarter points of the aerodynamic chord's length can be found.

Aircraft	S_h
Challenger 601	9.7548
Dassault Falcon 50	13.378
Average	13.5490

Table 8

The thickness-to-chord ratio is chosen at 0.13 to match the vertical tail for the same reasons. After using the same equations as before, the span, tip root, chord root, and mean aerodynamic cord can be found as follows.

Jet T-tail Aircraft (Horizontal Tail) parameters	
Aspect Ratio	5
Taper Ratio	0.3 to 0.5
Stabilizer Sweep Angle	40 degrees
The Span	8.23m
Tip root	1.37m
Chord root	3.42m
Mean Aerodynamic Chord	2.54m

Table 9

To size the rudder and elevator, using the reference aircraft, we find the flap area about 25ptc of the horizontal and vertical tail surfaces.

The Load Factor

In this chapter, the loads experienced by the aircraft during different flight envelopes by generating the V-n diagram. From this diagram, the maximum load factor, n_{max}, can be determined. This factor is then used in evaluating the weight of the fuselage and other components. The Characteristic values of the maximum load factors for transport aircraft are shown in Table 8.

Aircraft Type	Load Factor
Commercial Transport	$1.5 \leq n \leq 3.5$

Table 10

Determining the load factor requires the evaluation of two cases: the load factor during maneuvers and the load factor during gusts. The largest load factor of the two multiplied by a safety factor of 1.5 will determine the limit load factor which will in turn be used for the class II weight estimation method detailed in chapter 4.

The maximum Maneuver Loads

The loading experienced during maneuvers is specified in the Airworthiness certification specification. Different requirements are set for different flight conditions. The regulations specify that a large private jet, such as the design aircraft, is subject to CS/FAR-25 regulations. These are specified in Tables 9 and 10 below.

CS-25	Transport	≤ 4100 lb.	3.8
		$4100 < W_{TO} \leq 50{,}000$ lb.	$2.1 + 24{,}000/\ W_{TO} + 10{,}000$
		> 50,000 lb.	2.5

Table 11

As the aircraft has an MTOW of 75,960 lb., the tables above indicate that the maximum maneuver loads are 2.5 and -1 for positive and negative loads respectively.

Maneuver Loading

These load factors may be stated as functions of the aircraft's velocity and used to create a V-n diagram. The 0-A curve represents the maximum normal component load caused by high angles of attack and therefore the maximum lift in the positive direction. The stall speed of 77m/s is marked by the 0-A curve crossing the n=1 line which is exactly what was assumed in assignment 2. The maximum limit is 2.5 as determined from Table 9 for CS/FAR-25 regulations in the previous section. The A-D curve runs parallel to the x-axis along the n=2.5 line from the end of the 0-A curve to the dive velocity. This velocity is determined from the CS/FAR-25 to be $1.05*V_{cruise}$. The 0-H curve represents the maximum load generated by high angles of attack in the negative direction. The H-F curve represents the largest negative load factor this aircraft can experience, specified as -1 by the CS/FAR-25 regulations.

The gust load diagram, the second part of the V-n diagram, will be generated in this section using the loading factor during gusts. The combined maneuver and gust loading diagrams are shown in Figure 2 below.

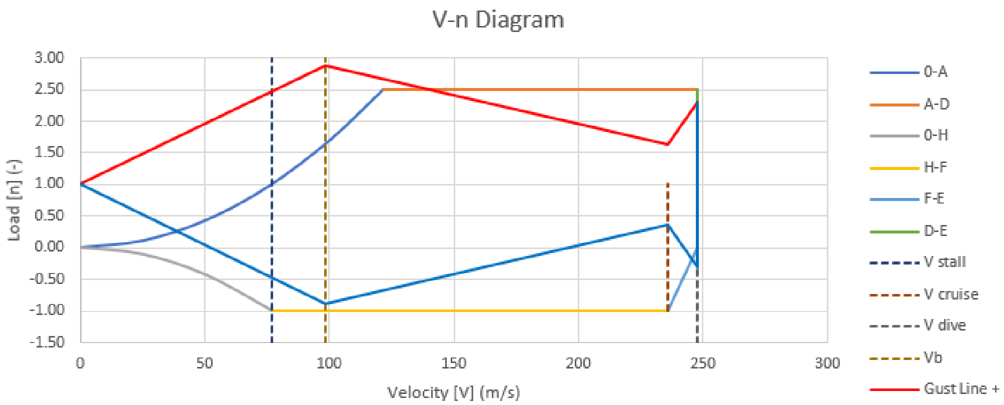


Figure 3: Loading Diagram

The gust loading diagram will be evaluated at three velocities; V_B, the speed at which the highest lift and maximum angle of attack occur, V_C, the cruise speed, and V_D, the dive speed. 1 is the mean load factor. It is assumed that the aircraft will have this load factor before the gust hits and will return to this load factor after the gust has passed.

For the high angle of attack condition, the aircraft is assumed to be flying at sea level yielding a statistical gust velocity value of $\hat{u} = 66\ \mathrm{ft/s}$. For the level flight condition, the aircraft is assumed to be flying at cruising altitude (13,700 m). This gives $\hat{u} = 30\ \mathrm{ft/s}$. Finally, for the dive condition, the aircraft is assumed to be flying at 3440 m which yields $\hat{u} = 25\ \mathrm{ft/s}$.

Drag Estimation
General Method

The general method for calculating the effects of drag is done by calculating the zero-lift drag coefficient, C_{D_0}, and the lift-induced drag coefficient C_{D_L} as shown in Equation 1.

$$C_D = C_{D_0} + C_{D_L}$$

Equation 1

The reason for this is simple. C_{D_0} remains constant, while C_{D_L} depends on the lift coefficient C_L or the angle of attack α. The improved drag coefficient, C_D, will be calculated for the most significant parts of the aircraft. From the CAD drawings, these are; The wings, fuselage, horizontal tail, vertical tail, engine nacelle, engine pylons, landing gear, stores, windscreen, and trim settings.

Since the cruise Mach number is 0.8, which is above 0.6, the aircraft will be fully evaluated as transonic. This indicates that the cruise phase of the mission profile is quite significant. Therefore, the improved drag prediction will be calculated for cruise conditions.

Reynolds Number

The Reynolds number, Re, is calculated as $2.58 * 10^6$ using the values in Table 2 below.

Symbol	Description	Value	Unit

M	Cruise Mach Number	0.8	-
V	Cruise Speed	236.11	m/s
μ	Kinematic Viscosity of Air	5.83×10^{-5}	m^2/sec
ρ	Density at Cruise Altitude	0.2381	kg/m3
T	Cruise Air Temperature	217	k
MAC	Mean Aerodynamic Chord	2.71	m

Table 12: Basic flight conditions

Wing Contribution

The wing drag is calculated by separating it into its different drag coefficients. C_{DWwave}, the wing wave drag coefficient has a value of 0.01314. C_{D0W}, the wing zero-lift drag coefficient, has a value of 0.02101.

Fuselage Contribution

$C_{Dw,fus}$,the fuselage wave drag coefficient, has a value of 0.02, while the fuselage zero-lift drag coefficient, fuselage zero lift coefficient, has a value of 0.01212.

Empennage Contribution

The empennage drag contribution is split between the contribution of the horizontal tail and the contribution of the vertical tail. The process for calculating the drag contribution of the horizontal tail is nearly identical to the process used in determining the drag contribution. The process for calculating the contribution of the vertical tail is similar, however, it is assumed there is no lift contribution (no sideslip).

The Horizontal Tail

$C_{D\ h\ tail\ wave}$, the horizontal tail wave drag coefficient, has a value of 0.002111. The horizontal tail zero-lift drag coefficient,$C_{D0\ h\ tail}$, has a value of 0.011806.

The Vertical Tail

$C_{D\ v\ tail\ wave}$, the horizontal tail wave drag coefficient, has a value of 0.002. The horizontal tail zero-lift drag coefficient,$C_{D0\ v\ tail}$, has a value of 0.07206.

The C_l – C_d Graph

Summing all the calculations from chapter 2 and using the Cl and angles of attack, α, the C_l-C_d graph can be constructed. It is shown in Figure 1. The curves are added from left to right. For example, the fuselage line is the drag contribution of the wing plus the drag contribution of the fuselage. The right-most line, in this case, the trim setting line, is a total drag with the contributions of all major components accounted for.

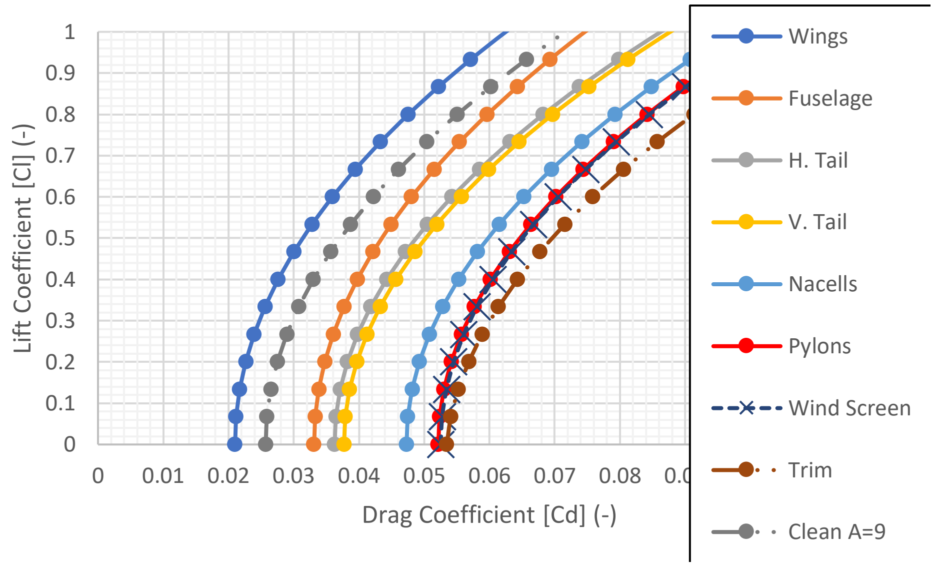


Figure 4: Drag Polar Graph

As expected, the biggest zero-lift drag coefficients belong to the wings, fuselage, and empennage. The biggest drag contribution is caused by the induced drag from the wings. The data here is most accurate when cruising at Mach 0.8 where C_L is 0.2.

Compared to the old drag estimation, the new drag estimation is larger than the previous one, however, the general shape of the C_l-C_d curve is quite similar. The new total drag coefficient is roughly 2 times larger than the old one. There are many potential reasons for this, however, the most concerning is that the method used previously severely underestimated the drag coefficient. This could lead to fewer iterations when sizing the engines and weight.

Conclusion

To decrease the drag contributions in aircraft design, which is crucial for reducing the environmental impact of airplanes during flight. Suggestions include shortening engine pylons, adding winglets, smoothing the transition between empennage surfaces and the tail cone, and changing the empennage configuration to a V-tail or adding vertical fins. These changes would decrease drag, reduce energy consumption, and improve the aircraft's maneuverability. However, some changes, such as attaching engines directly to the fuselage, could also increase noise and vibration, which would need to be mitigated with proper sound insulation.

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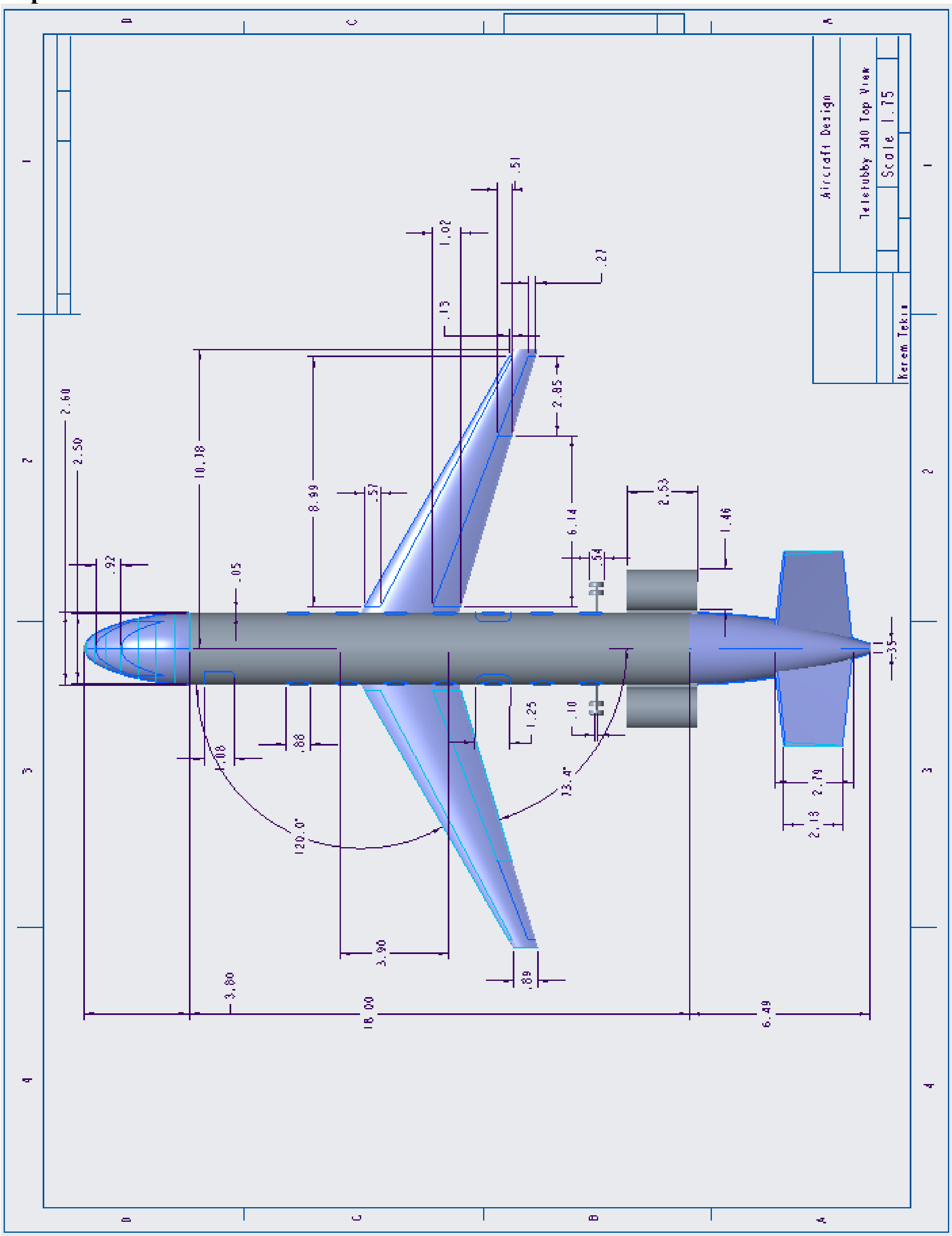
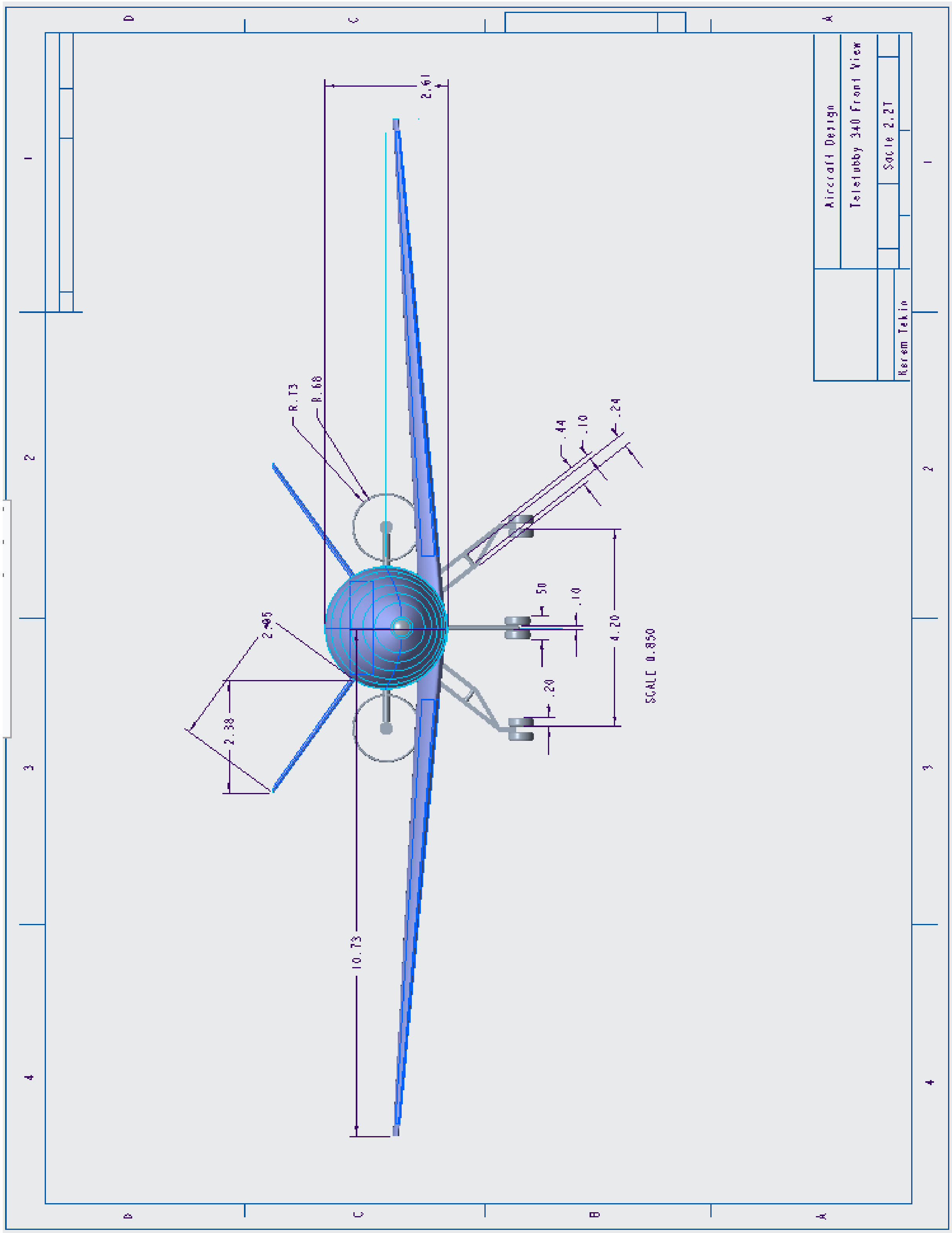
Appendix A Technical Drawings
Top View

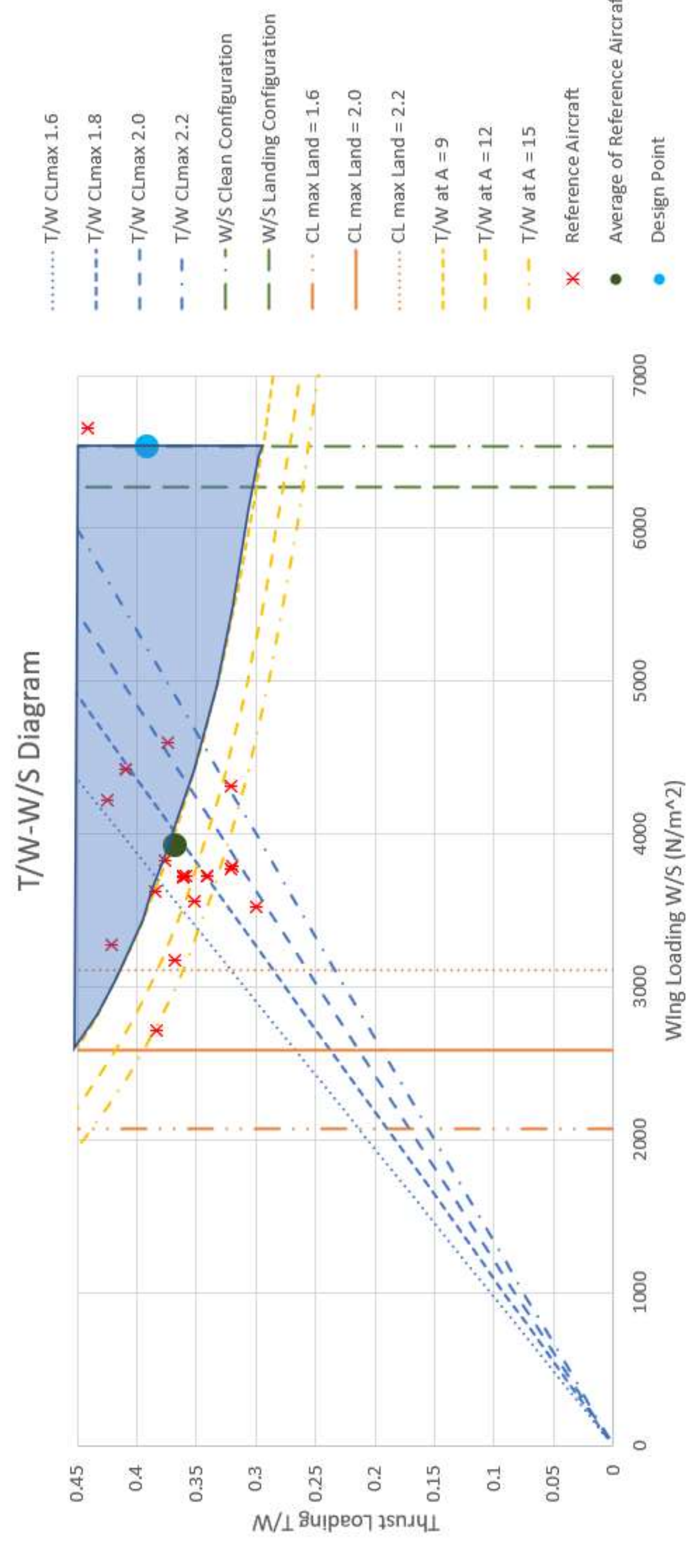


Front View



Aircraft Design
Teletubby 340 Front View
Socle 2.27
Kerem Tekin
SCALE 0.050

Appendix B: T/W-W/S Diagram



Appendix D: Aircraft Parameters Table

Parameter	Value	Unit
Wings		
Wing Span	28	m
Wing Area	11.33	m^2
Aspect Ratio	12	-
Sweep Angle	30	degrees
Cabin		
Cabin Length	19	m
Flight Deck Length	2.5	m
Tail Cone Length	6.49	m
Nose Cone Length	3.75	m
Cabin Height	1.5	m
Cabin Width	2.5	m
Chair Width	25	in
Aisle Width	20	in
Fuselage		
Wall Thickness	5	cm
Slenderness Ratio	10.54	-
Nose Slenderness Ratio	1.44	-
Tail Slenderness Ratio	2.5	-
Visibility		
Over-nose Angle	13.1	degrees
Overside Angle	25	degrees
Grazing Angle	29.9	degrees
Upward Angle	53	degrees
Weights and Loads		
Empty Weight	18,132	kg
Maximum Take-off Weight	34,455	kg
Fuel Weight	14,268	kg
Maximum Wing Loading	3,046	N/m^2
Performance		
Cruise Altitude	13,700	m
Cruise Speed	850	km/h
Cruise Lift Coefficient	1.2	-
Landing Lift Coefficient	3.48	-
Maximum Lift Coefficient (take-off)	3.06	-
Take-off distance	1650	m
Landing Distance	1,650	m
Range	6,300	km
Engine		
take off thrust	132,419	N
climb thrust	83,621	N

cruise thrust	2,181	N
climb thrust one engine out	14,761	N
number of engines	2	-