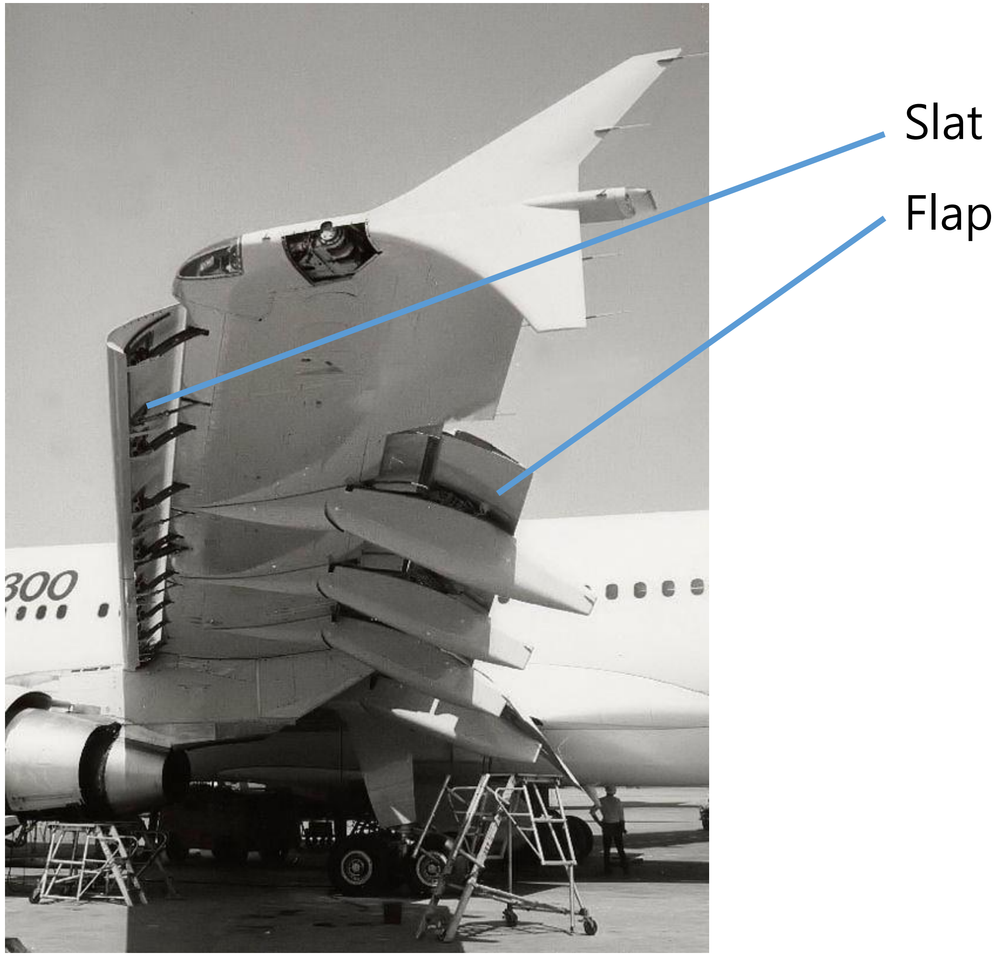
Slat & Flap 작동에 따른

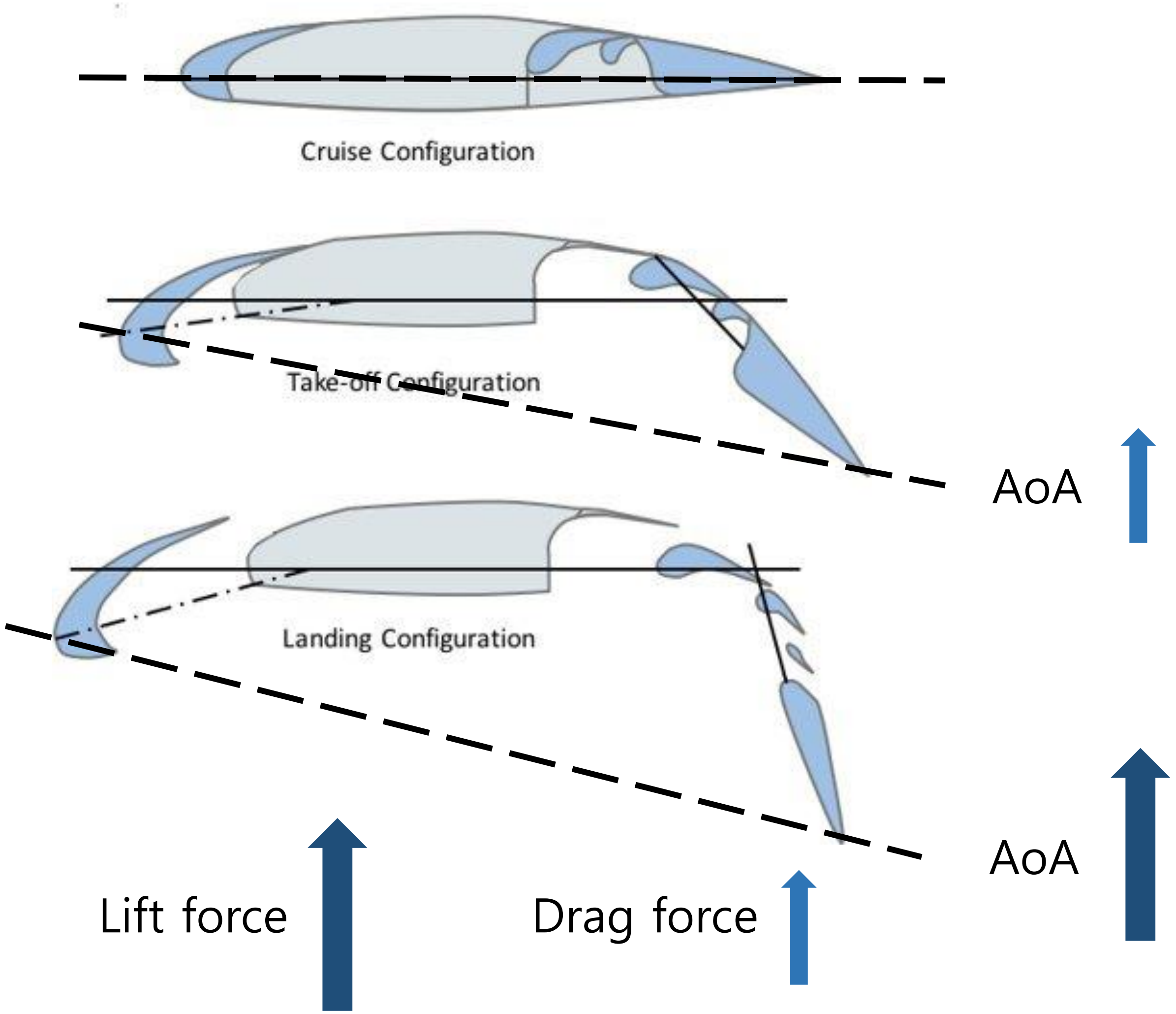
Boeing 737-500 이륙 상황 분석

2018016044 여준성
2021081121 송기원

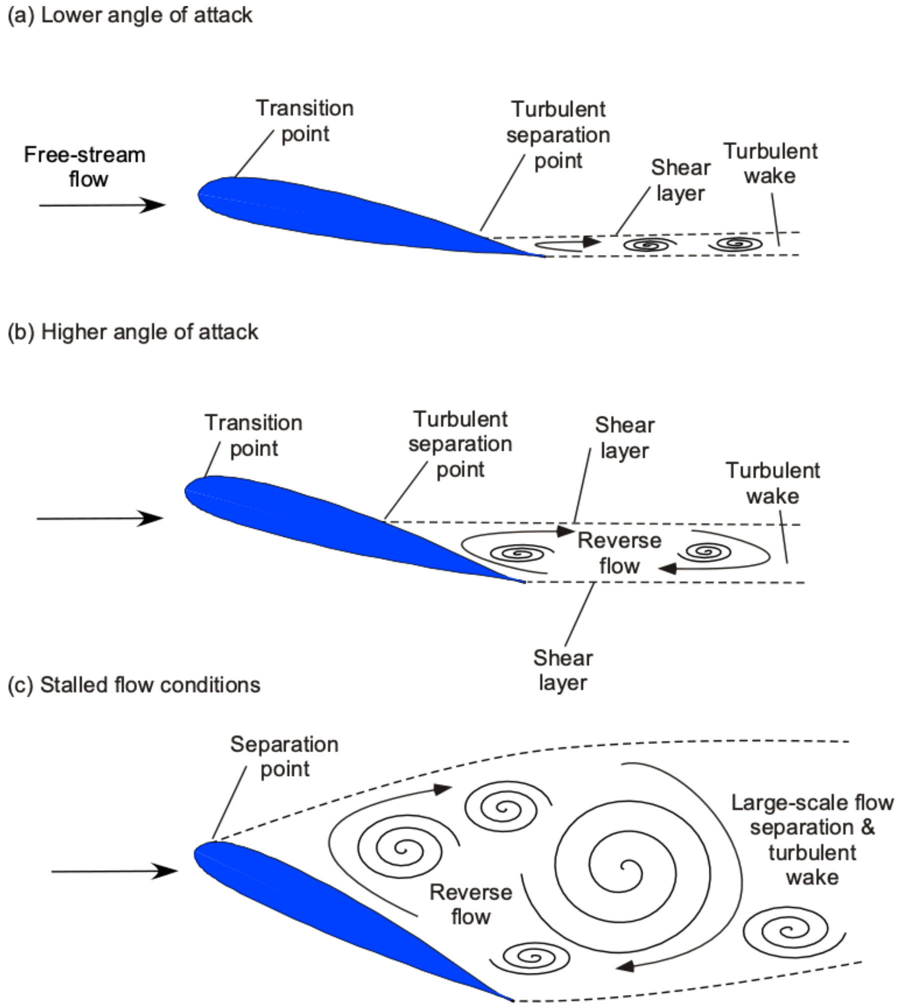
주제 소개

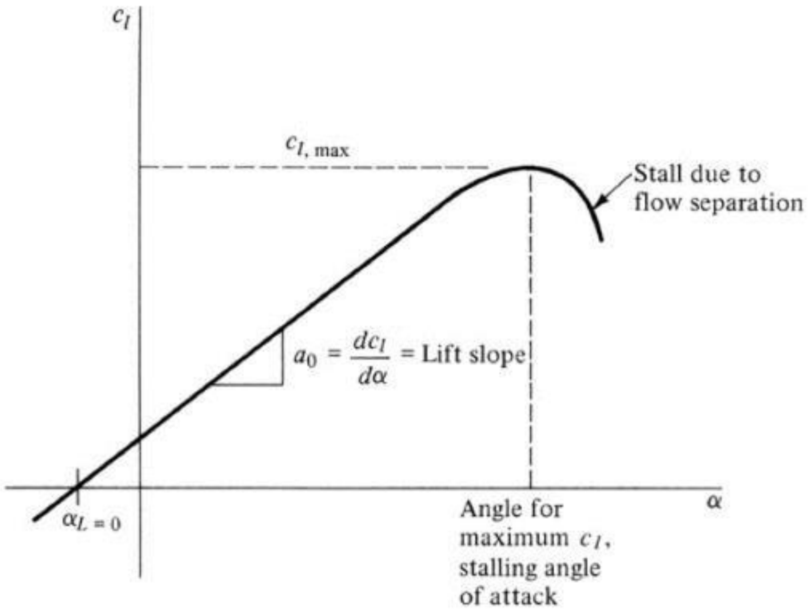
Slat & Flap


300
Slat
Flap


Cruise Configuration
Take-off Configuration
AoA
Landing Configuration
AoA
Lift force
Drag force

Stall

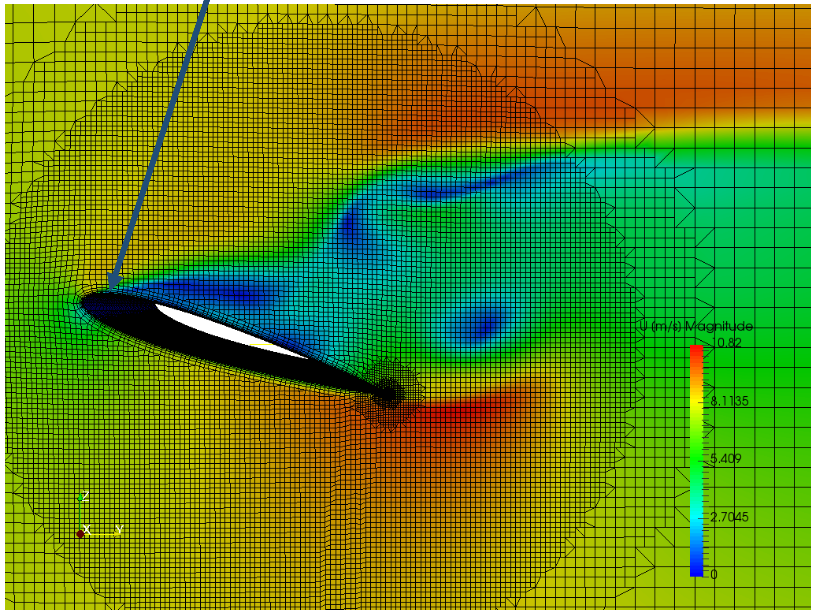

(a) Lower angle of attack
Free-stream flow
Transition point
Turbulent separation point
Shear layer
Turbulent wake
(b) Higher angle of attack
Transition point
Turbulent separation point
Shear layer
Turbulent wake
Reverse flow
Shear layer
(c) Stalled flow conditions
Separation point
Large-scale flow separation & turbulent wake
Reverse flow


c_l
$c_{l,\,max}$
Stall due to flow separation
$a_0 = \frac{dc_l}{d\alpha}$ = Lift slope
$\alpha_{L=0}$
α
Angle for maximum c_l, stalling angle of attack

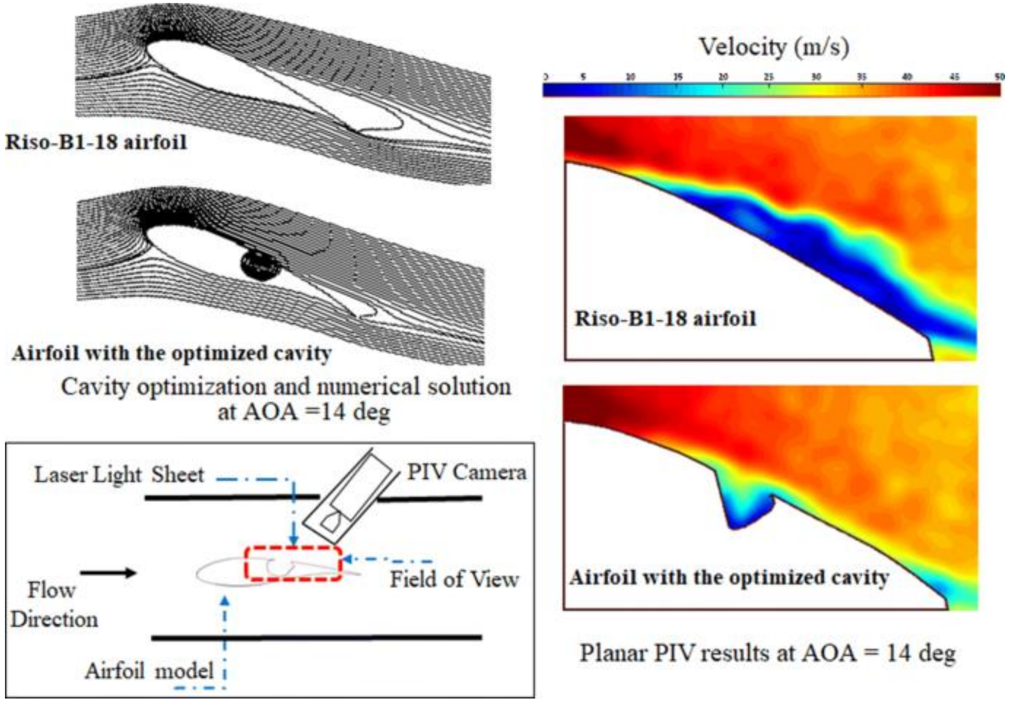
CAE Project

Stall

Solution : Separation point를 최대한 뒤로 이동시킨다.

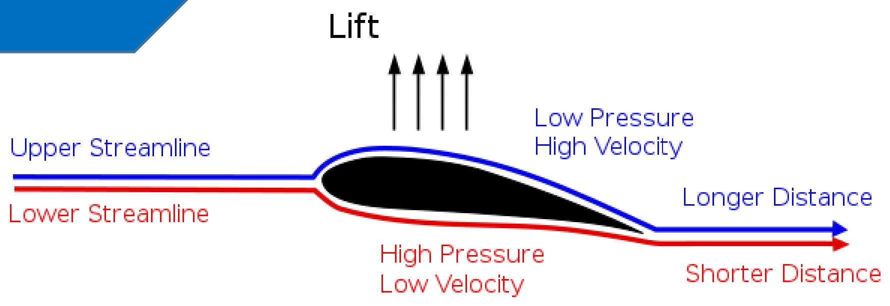


1.



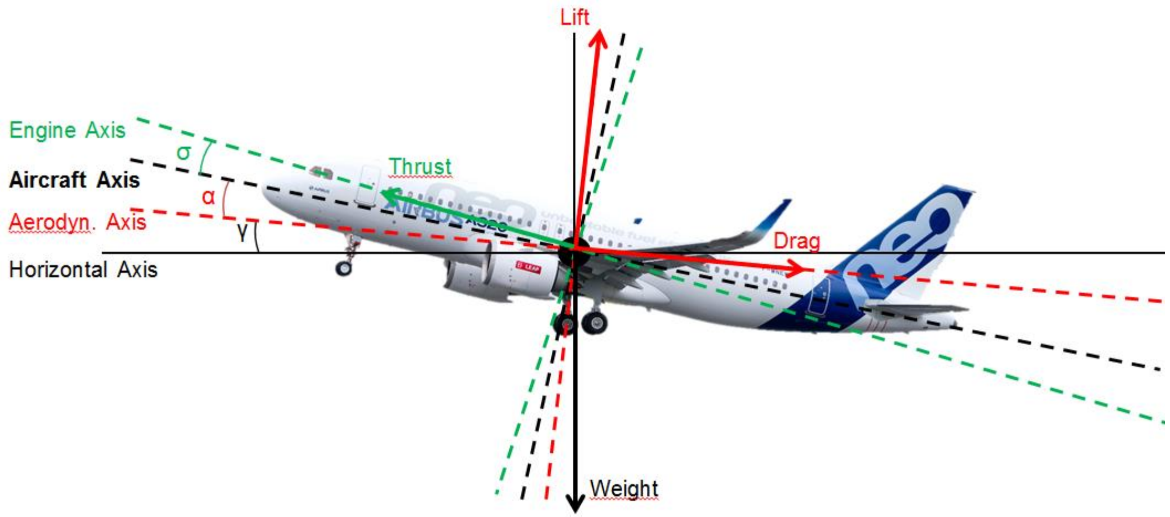
2.

Lift & Drag



$$F_{Lift} = \frac{1}{2} \cdot C_L \cdot \rho \cdot A \cdot v^2$$

$$F_{Drag} = \frac{1}{2} \cdot C_D \cdot \rho \cdot A \cdot v^2$$

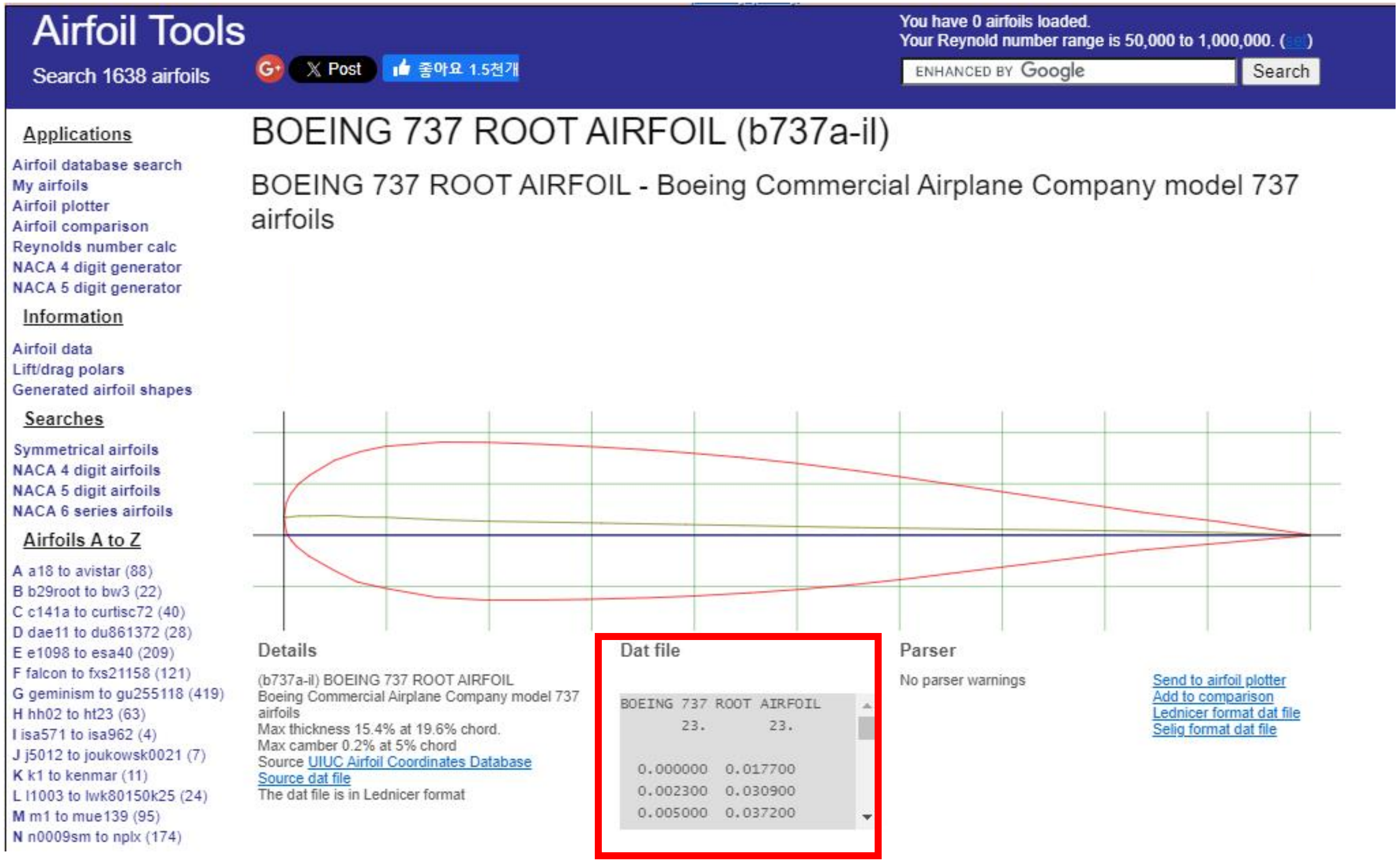


For convenience, we will use $\frac{1}{2} \cdot C_L \cdot \rho \cdot A$ instead of C_L to calculate force versus velocity

Also, we will neglect Aerodynamic Axis >> No nose up or down situation

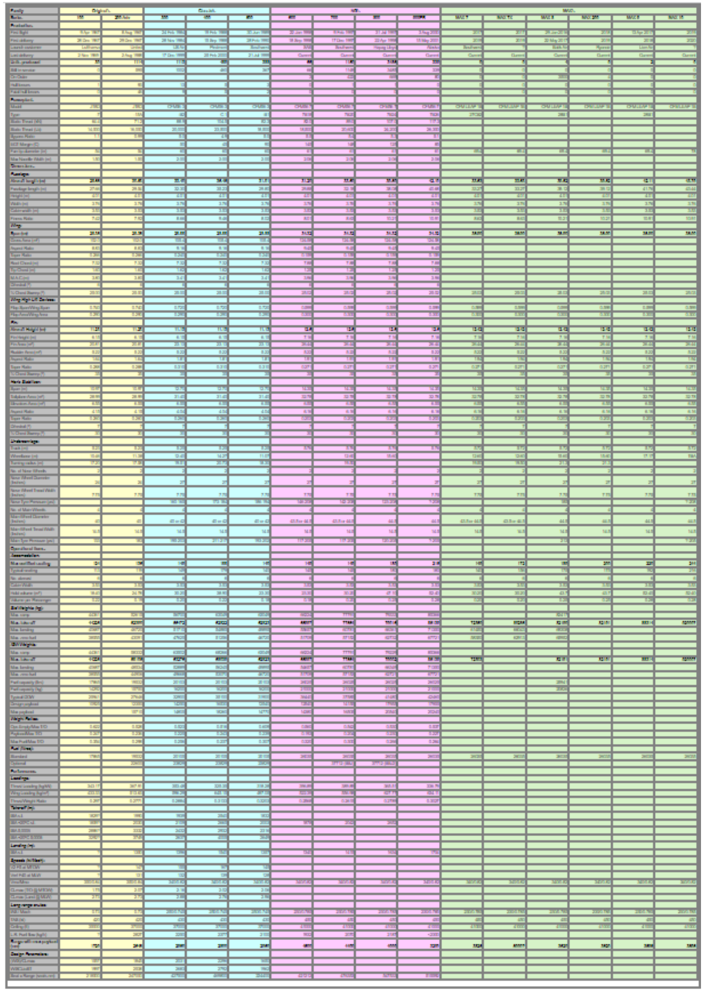
CAE Project

Airfoil – Boeing 737



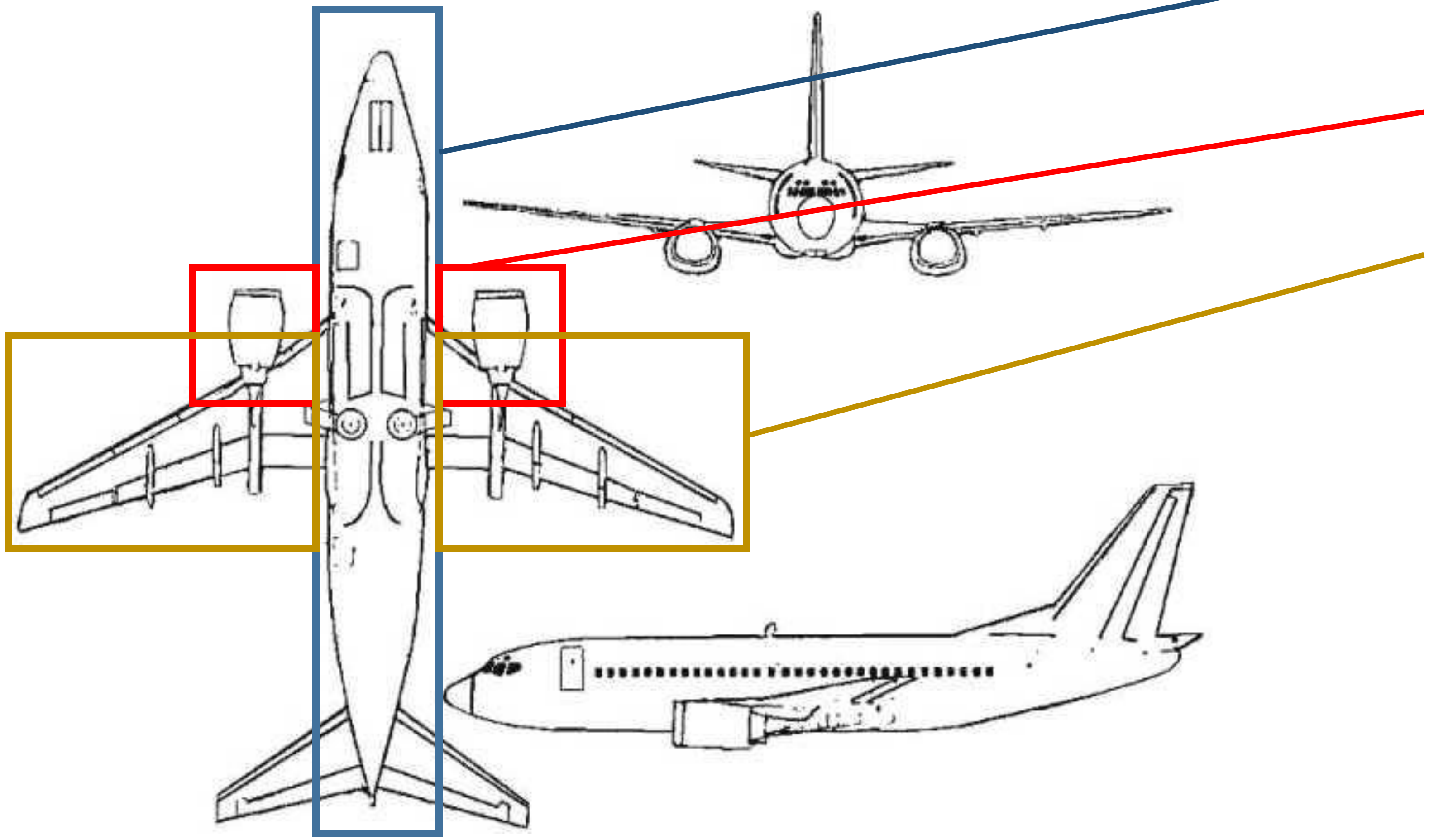
CAE Project

Data – Boeing 737



Powerplant:					
Model	JT8D	JT8D	CFM56-3	CFM56-3	CFM56-3
Type	-7	-15A	-B2	C-1	-B1
Static Thrust (kN)	64.4	71.2	88.9	104.5	82.3
Static Thrust (Lb)	14,000	16,000	20,000	23,500	18,500
Bypass Ratio	1.1	0.99	5.0	4.9	5.0
EGT Margin (C)			50	45	90
Fan tip diameter (in)	54	54	60	60	60
Max Nacelle Width (m)	1.50	1.50	2.00	2.00	2.00
Wing:					
Span (m)	**28.35**	**28.35**	**28.88**	**28.88**	**28.88**
Gross Area (m²)	102.0	102.0	105.4	105.4	105.4
Aspect Ratio	8.83	8.83	9.16	9.16	9.16
Taper Ratio	0.266	0.266	0.240	0.240	0.240
Root Chord (m)	7.32	7.32	7.32	7.32	7.32
Tip Chord (m)	1.60	1.60	1.62	1.62	1.62
M.A.C.(m)	3.80	3.80	3.41	3.41	3.41
Dihedral (°)	6	6	6	6	6
¼ Chord Sweep (°)	25.00	25.00	25.00	25.00	25.00
Wing High Lift Devices:					
Flap Span/Wing Span	0.740	0.740	0.720	0.720	0.720
Flap Area/Wing Area	0.290	0.290	0.290	0.290	0.290

Full model



3D Analysis

+

Neglect

+

2D Analysis

= Full model

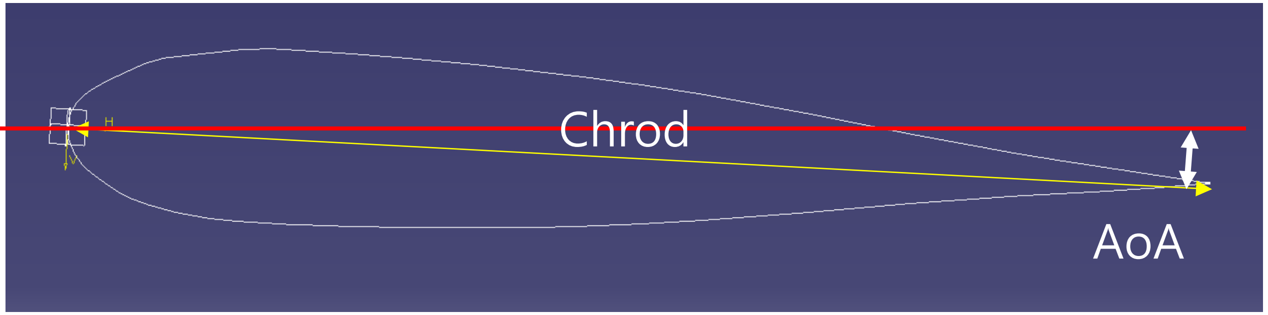
Wing Analysis - Comsol

AoA 6deg 2D CFD

CAE Project

AoA 6deg 2D Model

	A	B	C	D
1	StartLoft			
2	StartCurve			
3	0	0.0177	0	
4	0.0023	0.0309	0	
5	0.005	0.0372	0	
6	0.0076	0.0415	0	
7	0.0143	0.0499	0	
8	0.0249	0.0582	0	
9	0.0495	0.073	0	
10	0.074	0.0814	0	
11	0.099	0.0866	0	
12	0.153	0.0907	0	
13	0.1961	0.0905	0	
14	0.2504	0.0887	0	
15	0.3094	0.0858	0	
16	0.352	0.0833	0	
17	0.3919	0.0804	0	
18	0.4477	0.0756	0	
19	0.5034	0.0696	0	
20	0.5593	0.0626	0	
21	0.5965	0.0575	0	
22	0.6488	0.0498	0	
23	0.8351	0.0224	0	
24	0.9109	0.0132	0	
25	1	0.0003	0	
26	EndCurve		0	
27	EndLoft			
28	0	0.0177	0	
29	0.0022	0.0038	0	
30	0.0049	-0.0018	0	
31	0.0072	-0.0053	0	
32	0.0119	-0.0106	0	
33	0.0243	-0.0204	0	
34	0.0486	-0.0342	0	
35	0.0716	-0.0457	0	
36	0.0979	-0.0516	0	
37	0.1488	-0.0607	0	
38	0.1953	-0.0632	0	
39	0.2501	-0.0632	0	
40	0.2945	-0.0626	0	
41	0.3579	-0.061	0	
42	0.3965	-0.0595	0	
43	0.4543	-0.0563	0	
44	0.505	-0.0527	0	
45	0.5556	-0.0482	0	
46	0.6063	-0.0427	0	
47	0.6485	-0.0375	0	
48	0.8317	-0.0149	0	
49	0.941	-0.0053	0	
50	1	-0.0003	0	
51	EndCurve			
52	EndLoft			
53	End			
54				
55				
56				



Root Chord (m)	7.32	7.32	7.32	7.32	7.32
Tip Chord (m)	1.60	1.60	1.62	1.62	1.62

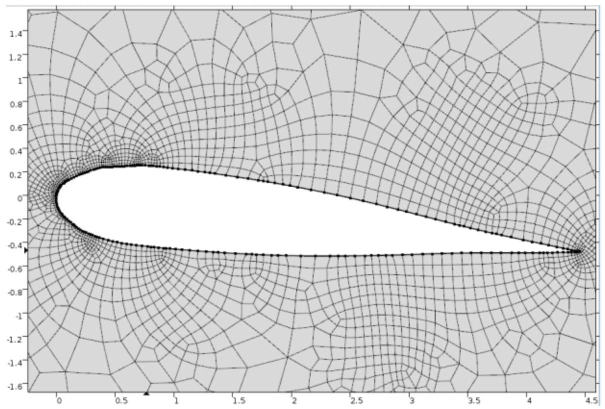
$$\frac{(7.32 + 1.62)}{2} = 4.47\ m : Chord\ Length$$

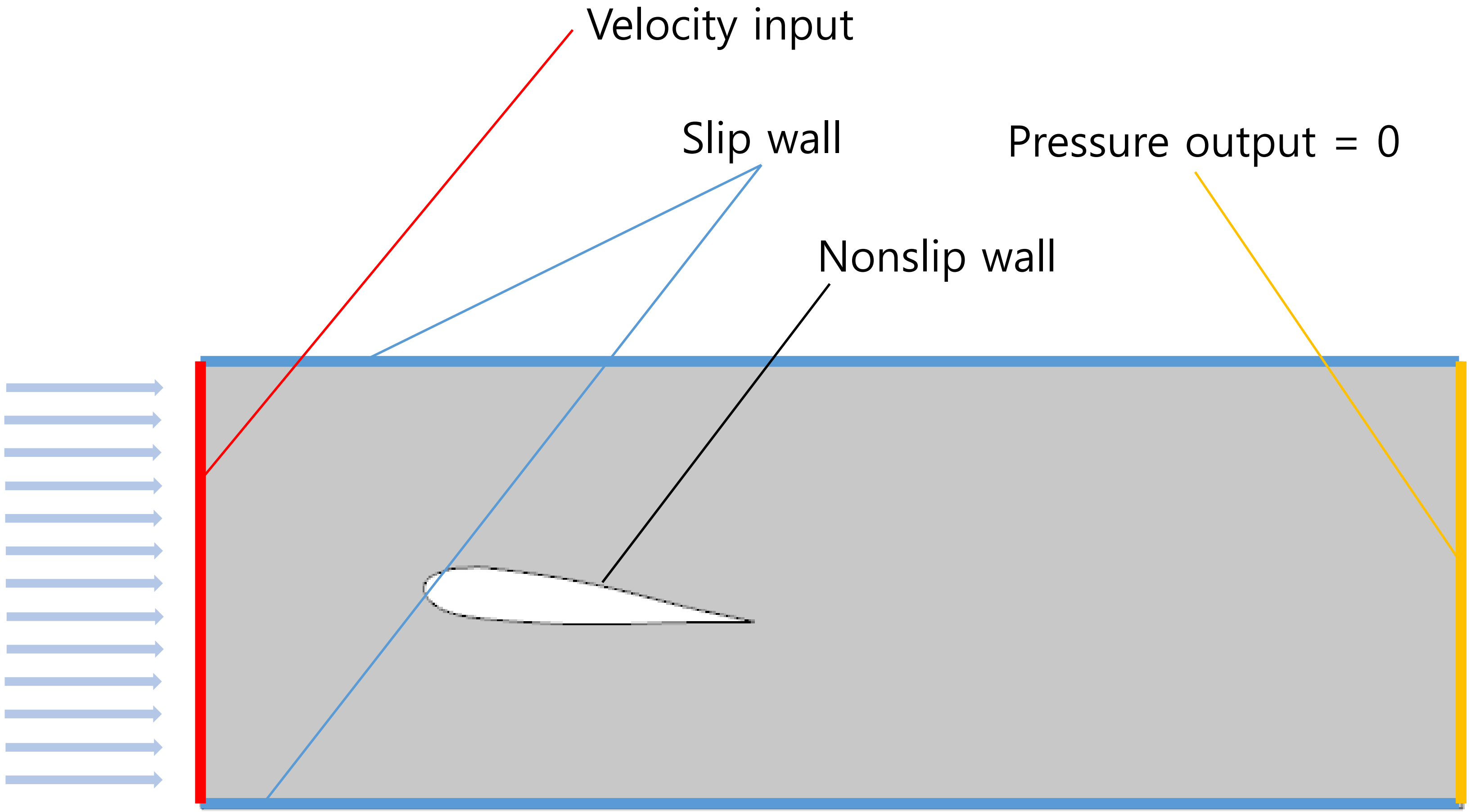
Dihedral (º)	6	6	6	6	6

$$6\ ° : Attack\ of\ Angle$$

CAE Project

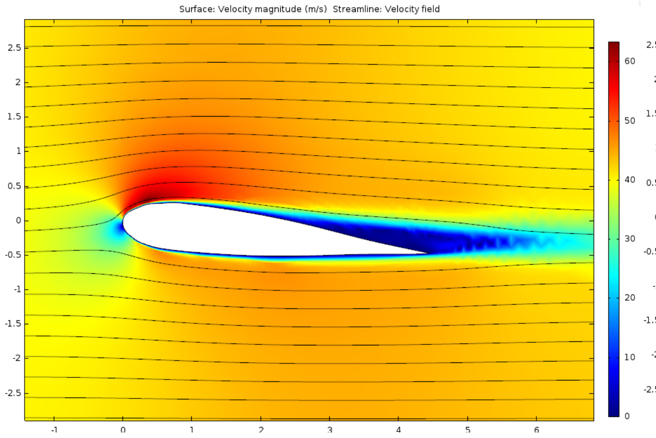
AoA 6deg 2D Model



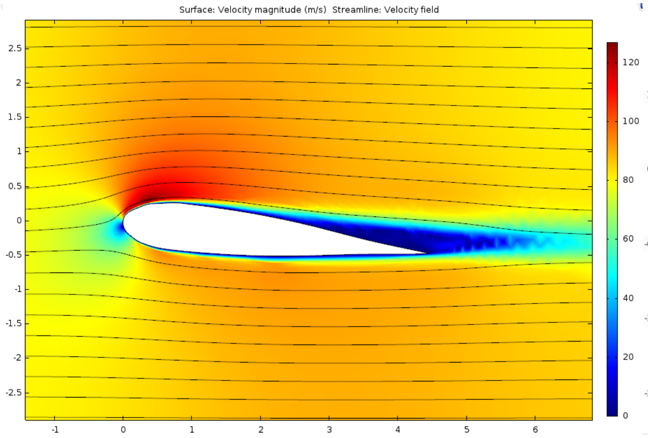


Altitude = Sea level
Temperature = 293.15 K
Air density(ρ) = 1.22 kg/m^3

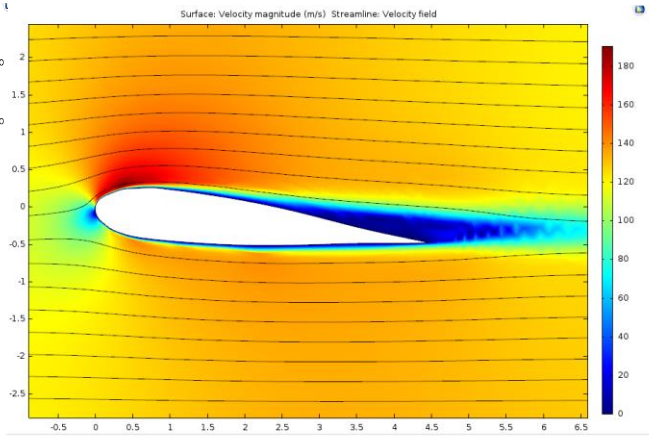
AoA 6deg 2D CFD



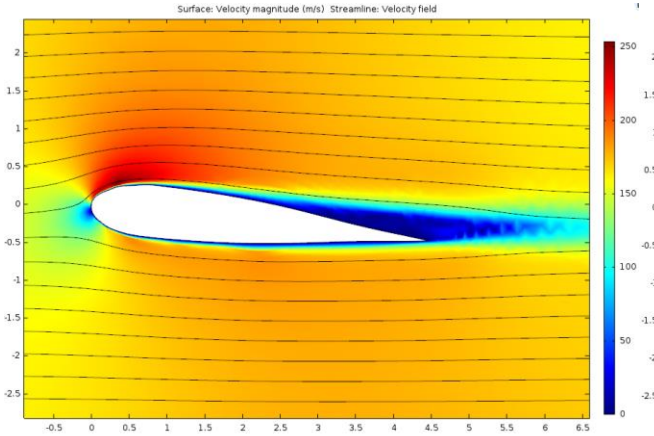
40m/s



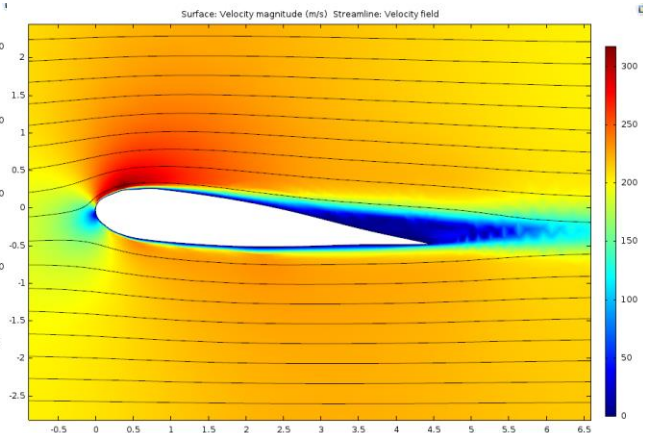
80m/s



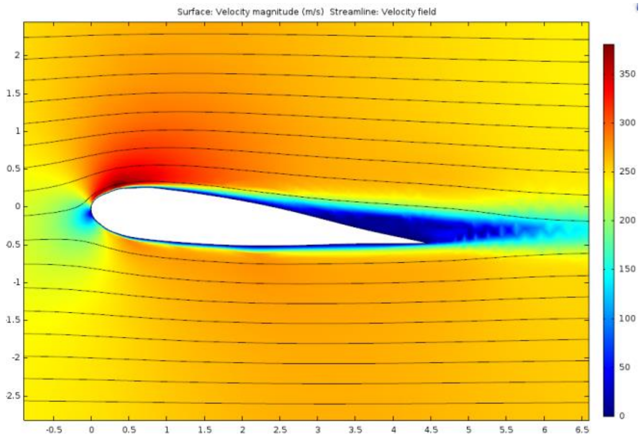
120m/s



160m/s



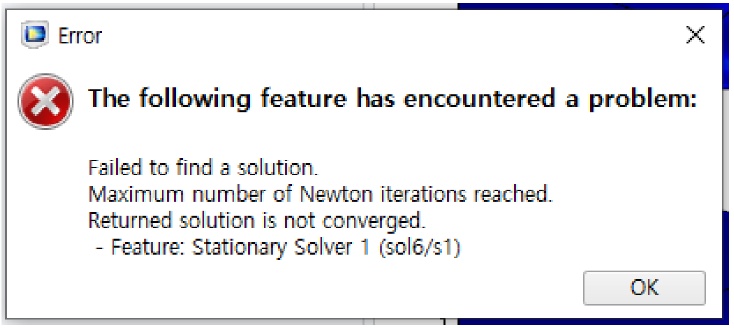
200m/s

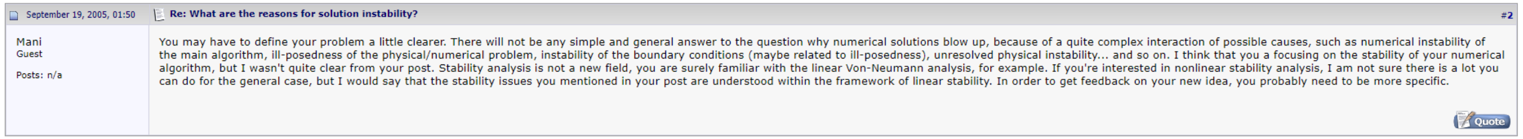


240m/s

CAE Project

Stationary vs Time Dependent



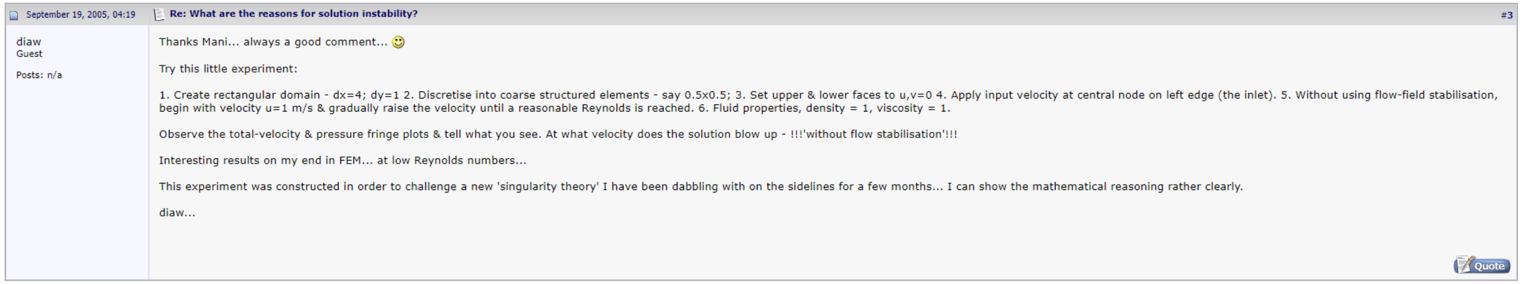


September 19, 2005, 01:50 — **Re: What are the reasons for solution instability?** #2

Mani
Guest
Posts: n/a

You may have to define your problem a little clearer. There will not be any simple and general answer to the question why numerical solutions blow up, because of a quite complex interaction of possible causes, such as numerical instability of the main algorithm, ill-posedness of the physical/numerical problem, instability of the boundary conditions (maybe related to ill-posedness), unresolved physical instability... and so on. I think that you a focusing on the stability of your numerical algorithm, but I wasn't quite clear from your post. Stability analysis is not a new field, you are surely familiar with the linear Von-Neumann analysis, for example. If you're interested in nonlinear stability analysis, I am not sure there is a lot you can do for the general case, but I would say that the stability issues you mentioned in your post are understood within the framework of linear stability. In order to get feedback on your new idea, you probably need to be more specific.

Quote



September 19, 2005, 04:19 — **Re: What are the reasons for solution instability?** #3

diaw
Guest
Posts: n/a

Thanks Mani... always a good comment... 🙂

Try this little experiment:

1. Create rectangular domain - dx=4; dy=1 2. Discretise into coarse structured elements - say 0.5x0.5; 3. Set upper & lower faces to u,v=0 4. Apply input velocity at central node on left edge (the inlet). 5. Without using flow-field stabilisation, begin with velocity u=1 m/s & gradually raise the velocity until a reasonable Reynolds is reached. 6. Fluid properties, density = 1, viscosity = 1.

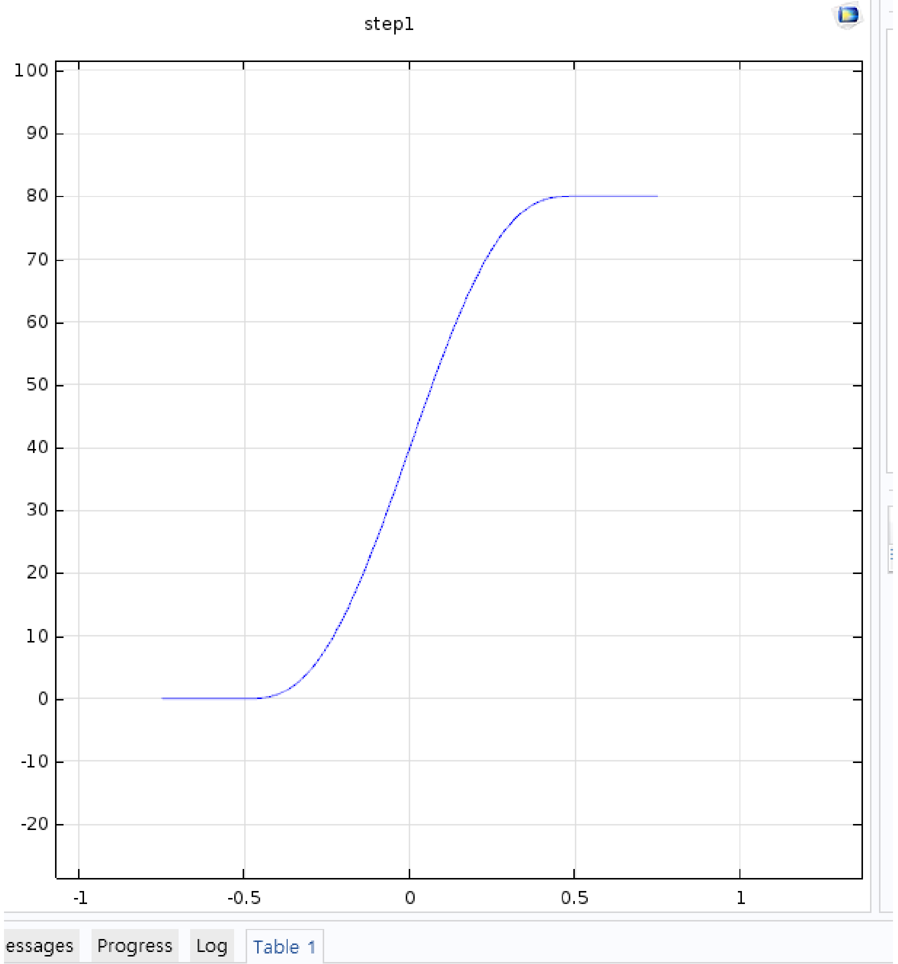
Observe the total-velocity & pressure fringe plots & tell what you see. At what velocity does the solution blow up - !!!'without flow stabilisation'!!!

Interesting results on my end in FEM... at low Reynolds numbers...

This experiment was constructed in order to challenge a new 'singularity theory' I have been dabbling with on the sidelines for a few months... I can show the mathematical reasoning rather clearly.

diaw...

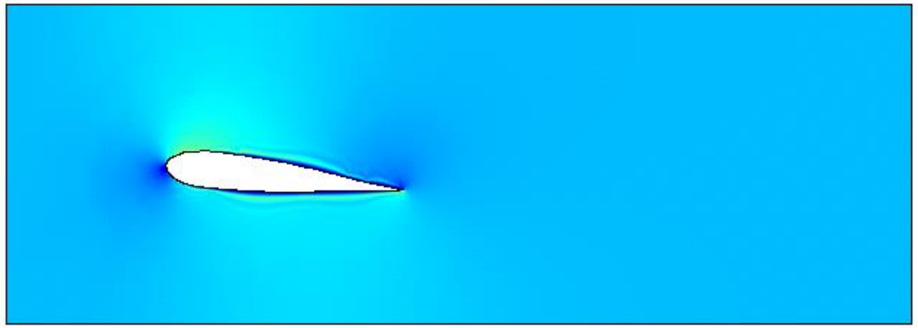
Quote

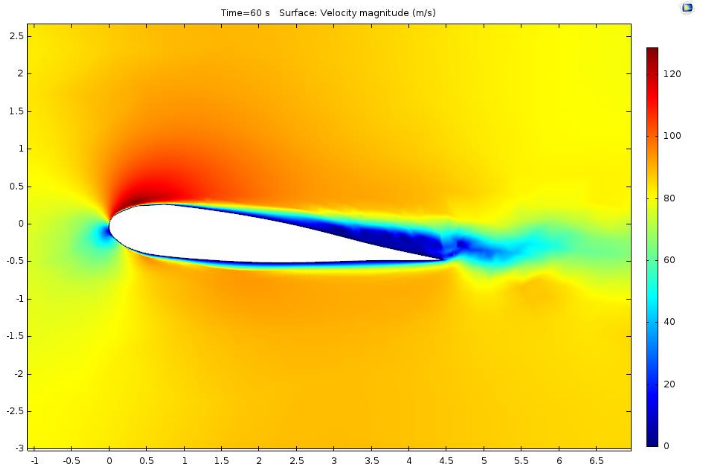


CAE Project

Stationary vs Time Dependent

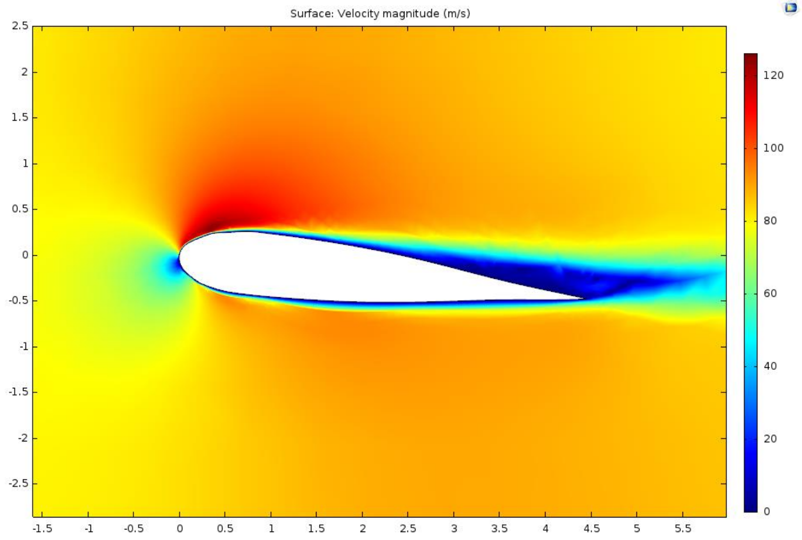
Step function * 80 m/s





vs

Stationary 80 m/s



Average of reasonable data ≈ Stationary result

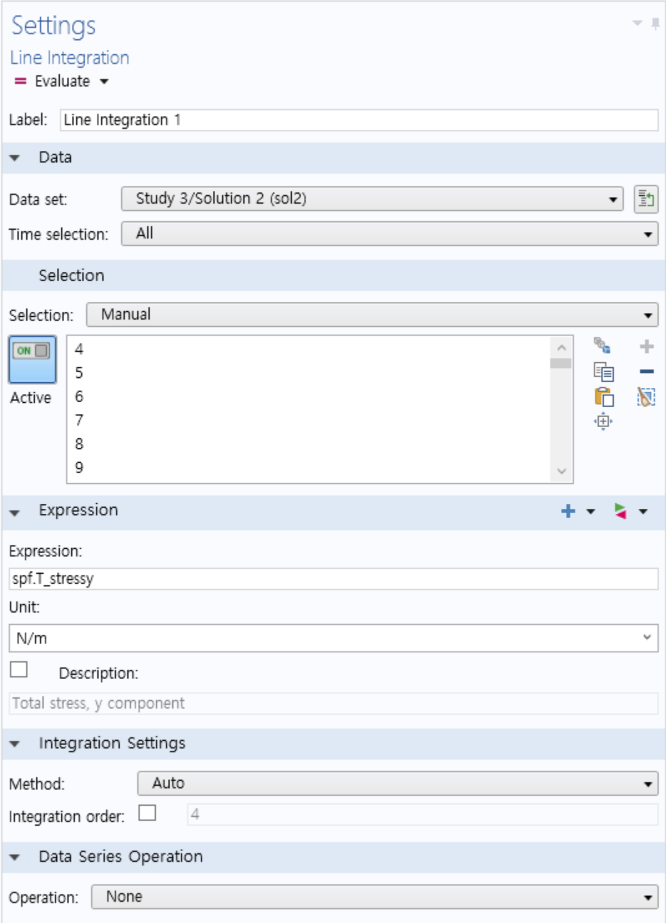
CAE Project

Calculating Lift & Drag Force

Momentum Equations

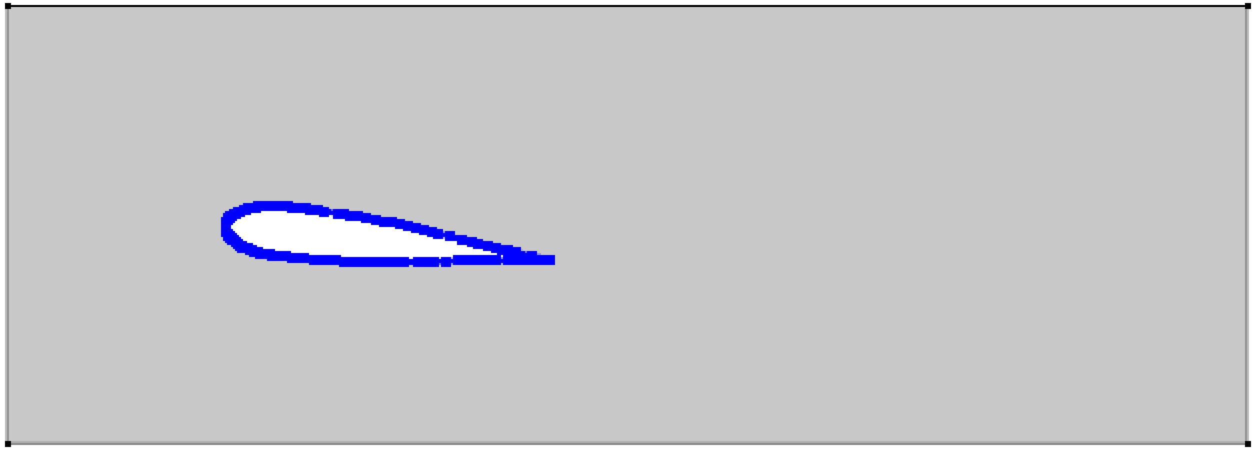
$$\rho \frac{D\vec{V}}{Dt} = -\nabla p + \rho \vec{g} + \mu \nabla^2 \vec{V}$$

Total derivative, Pressure gradient, Body force term, Diffusion term



spf.T_stressy

spf.T_stressx



Lift force per unit length / Drag force per unit length

Total stress, y component (N/m)	Total stress, x component (N/m)
-5748.0	-725.65

Why minus?

: This calculates reaction force from airfoil boundary

AoA 6deg Lift & Drag

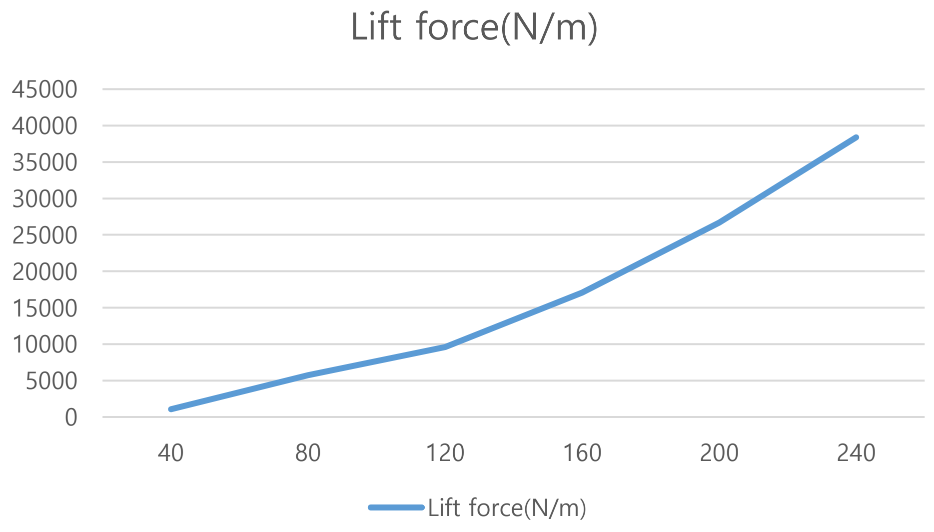
Velocity (m/s)	Lift force (N/m)	Drag force(N/m)
40	1066	183.6
80	5748	725.65
120	9598.2	1651.2
160	17064	2935
200	26669	4585.8
240	38388.7	6603.9

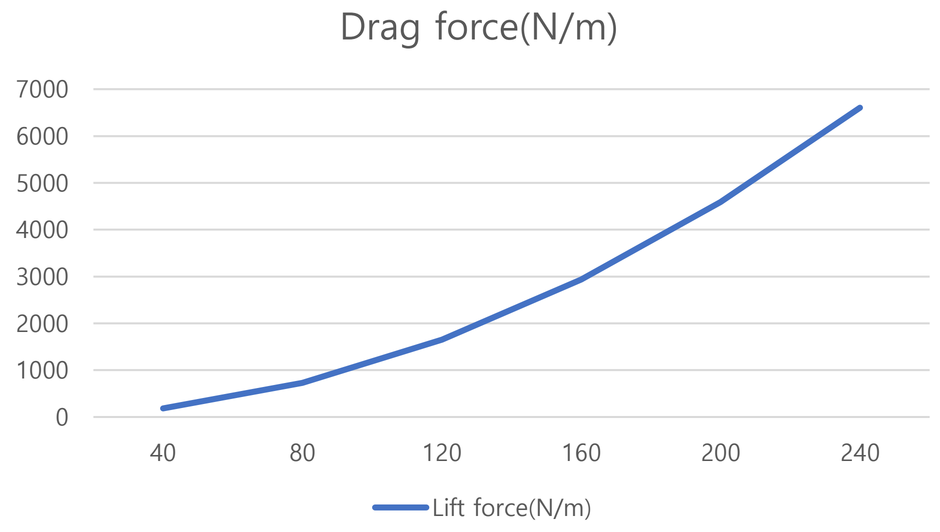
L*wing span	D*wing span	L/v^2	D/v^2
26650	4590	0.66625	0.11475
143700	18141.25	0.898125	0.113383
239955	41280	0.666542	0.114667
426600	73375	0.666563	0.114648
666725	114645	0.666725	0.114645
959717.5	165097.5	0.66647	0.114651

$$\frac{1}{2} \cdot C_L \cdot \rho \cdot A = 0.705\ /span$$

$$\frac{1}{2} \cdot C_D \cdot \rho \cdot A = 0.114\ /span$$

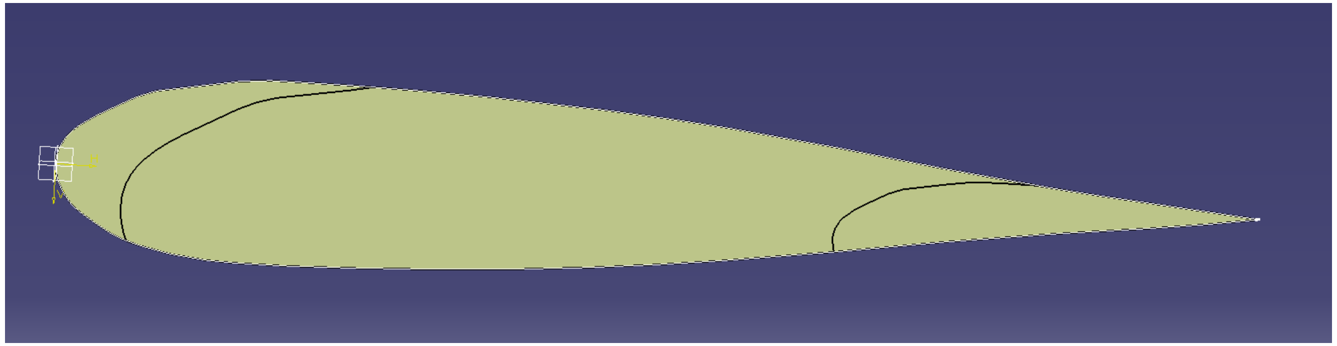
$$C_L/C_D = 6.184$$

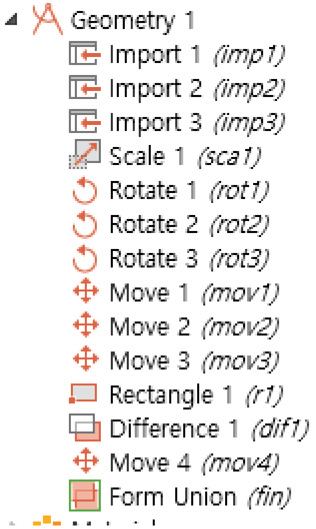


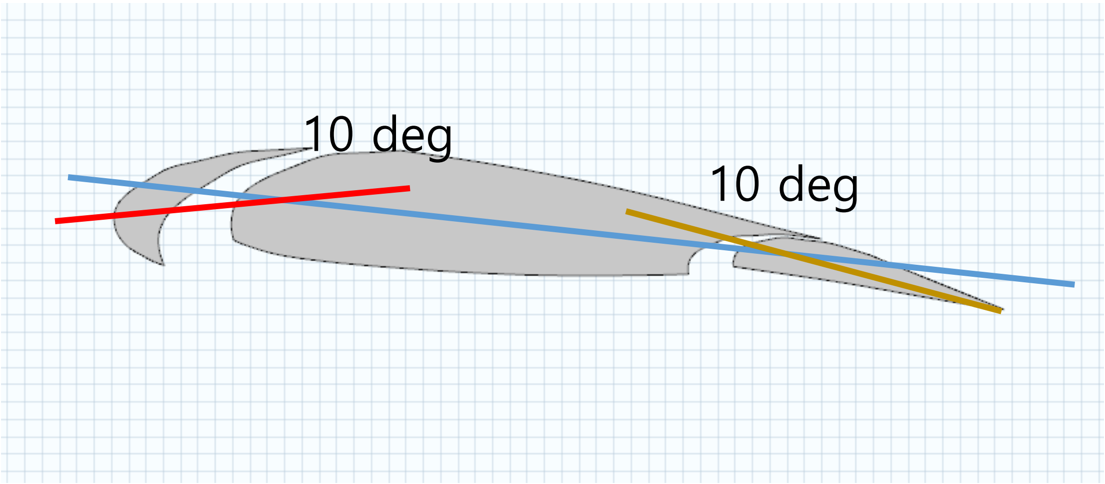


Slat & Flap 2D CFD

Slat & Flap 2D Model



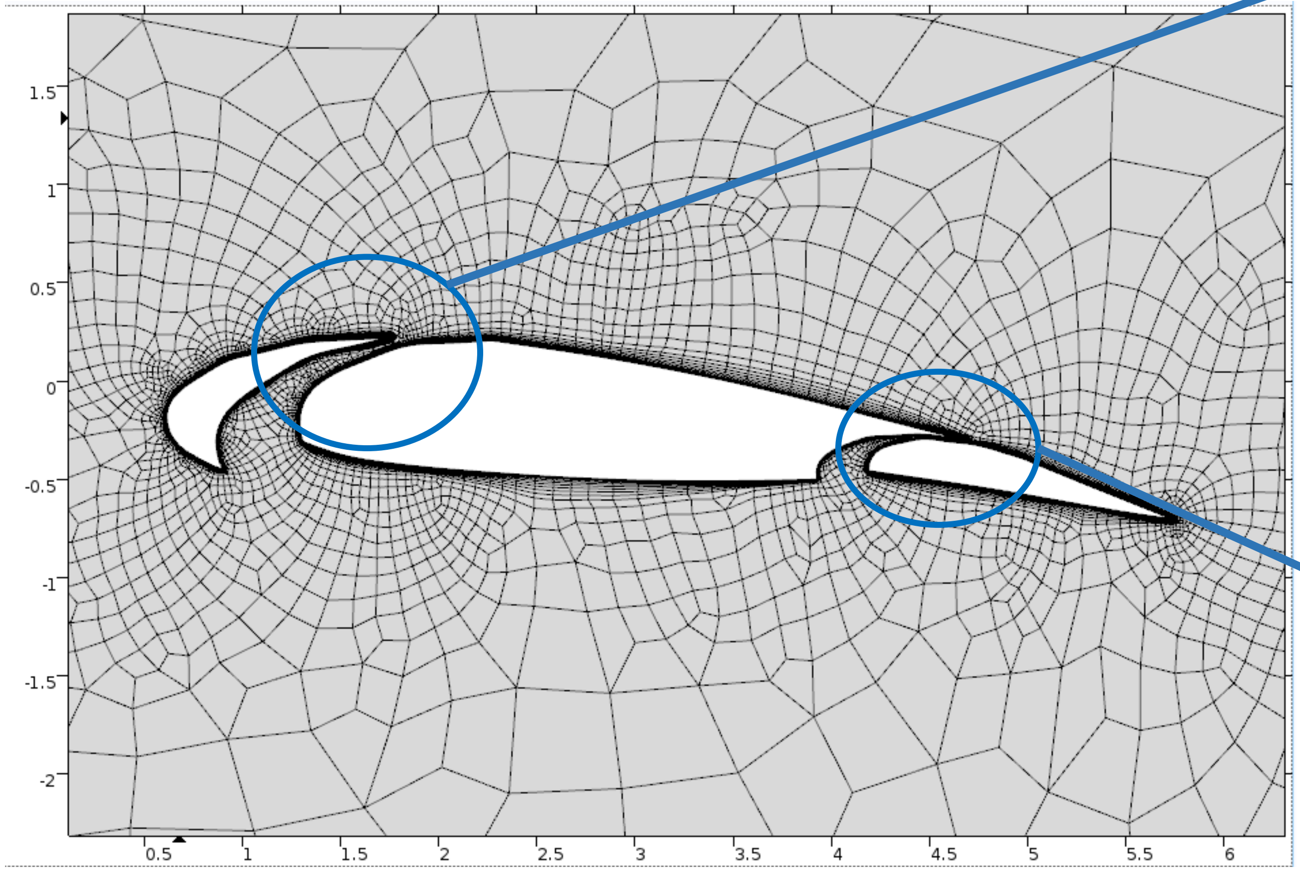


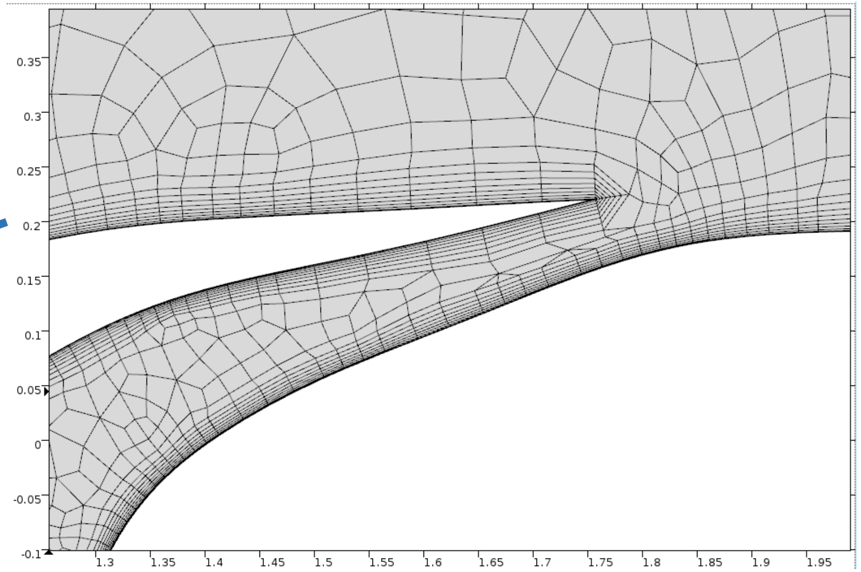


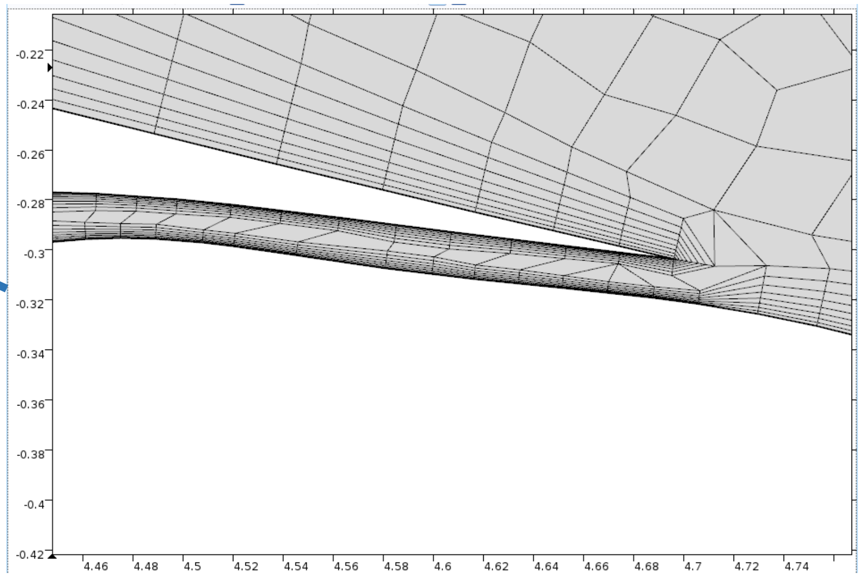
CAE Project

Slat & Flap 2D Mesh

Boundary Layer for Mesh Quality



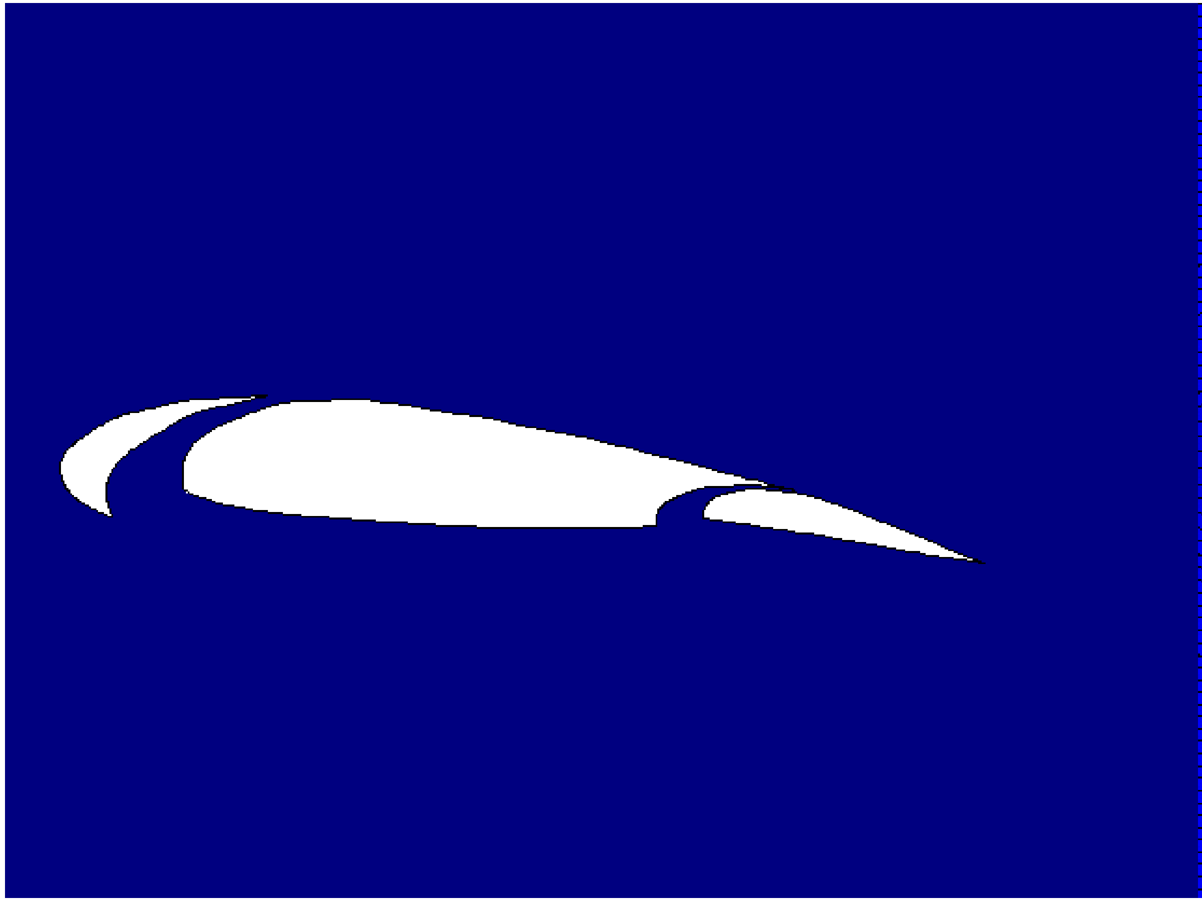




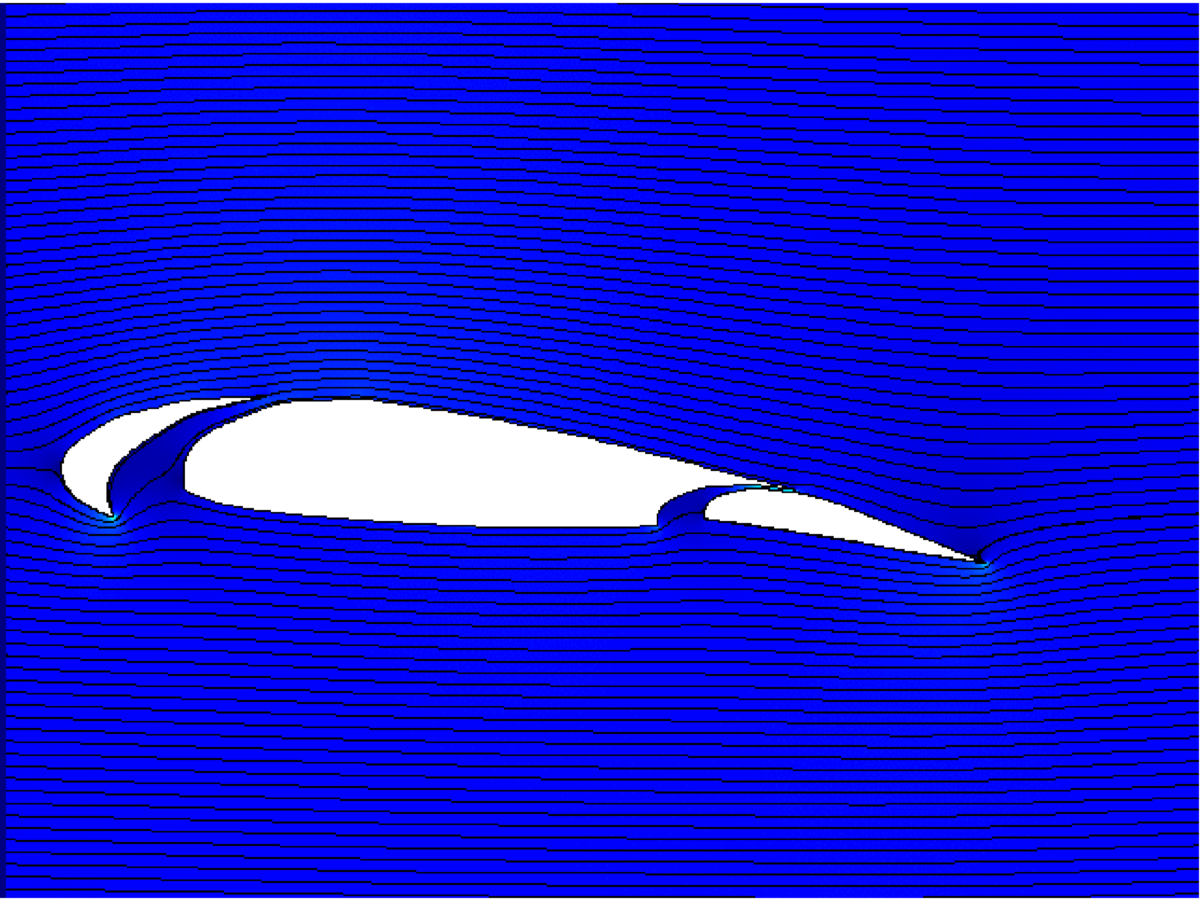
CAE Project

Slat & Flap 2D CFD

Acceleration from 0 m/s to 240 m/s

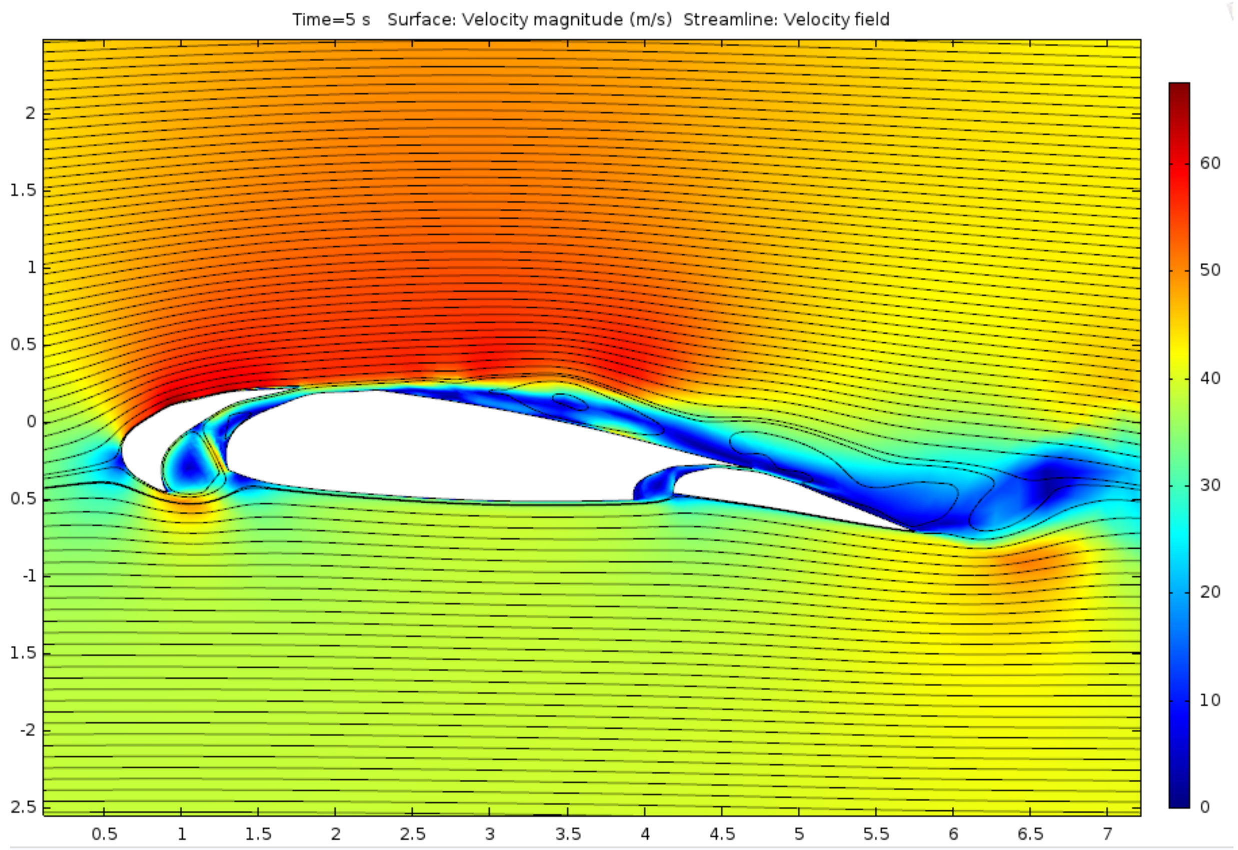


40 m/s steady flow



CAE Project

Slat & Flap 2D CFD



Lift (N/m)	Drag(N/m)	Lift/v^2	Drag/v^2
4988.564948	424.2375945	3.117853093	0.265148497

$$\frac{1}{2} \cdot C_L \cdot \rho \cdot A = 0.705 \; /span$$

$$\Rightarrow \frac{1}{2} \cdot C_L \cdot \rho \cdot A = 3.118 \; /span$$

$$\frac{1}{2} \cdot C_D \cdot \rho \cdot A = 0.114 \; /span$$

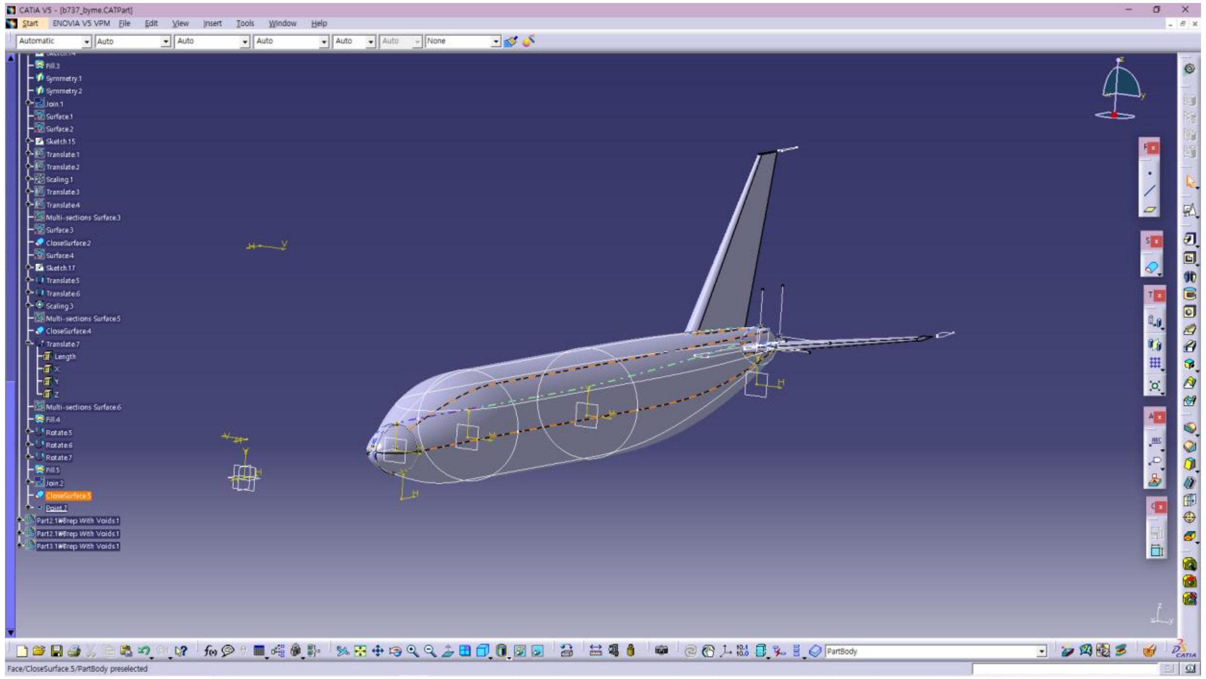
$$\Rightarrow \frac{1}{2} \cdot C_D \cdot \rho \cdot A = 0.265 \; /span$$

$$C_L/C_D = 6.184 \Rightarrow C_L/C_D = 11.77$$

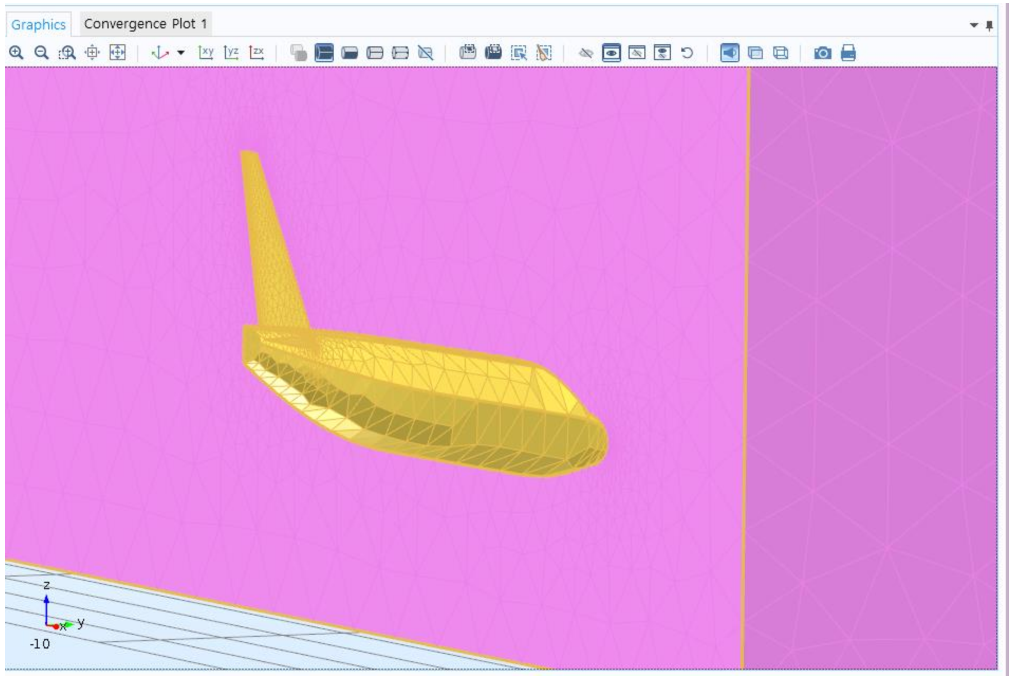
Body & Tailwing 3D CFD

Body & Tail 3D Model

Full Model

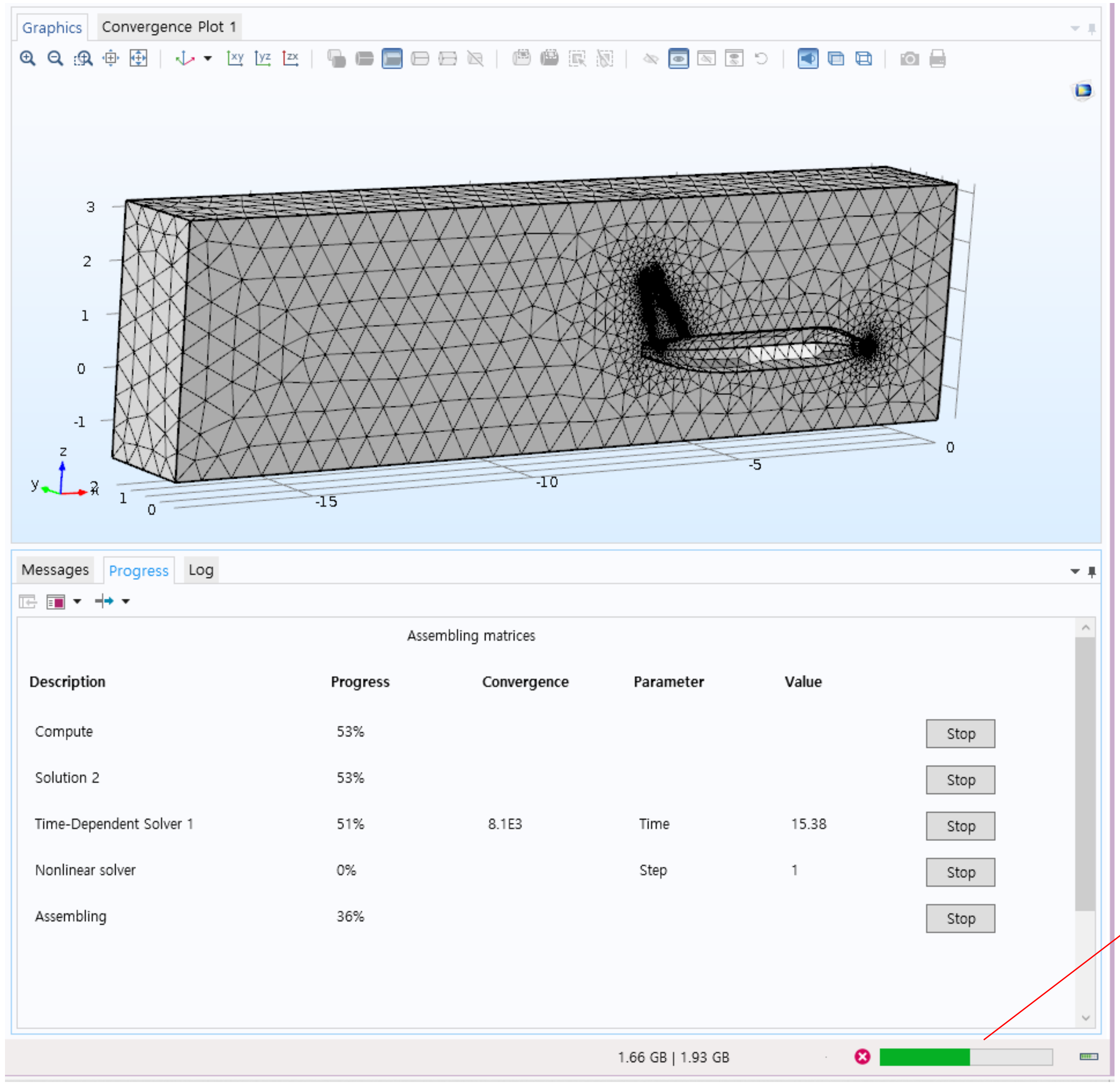


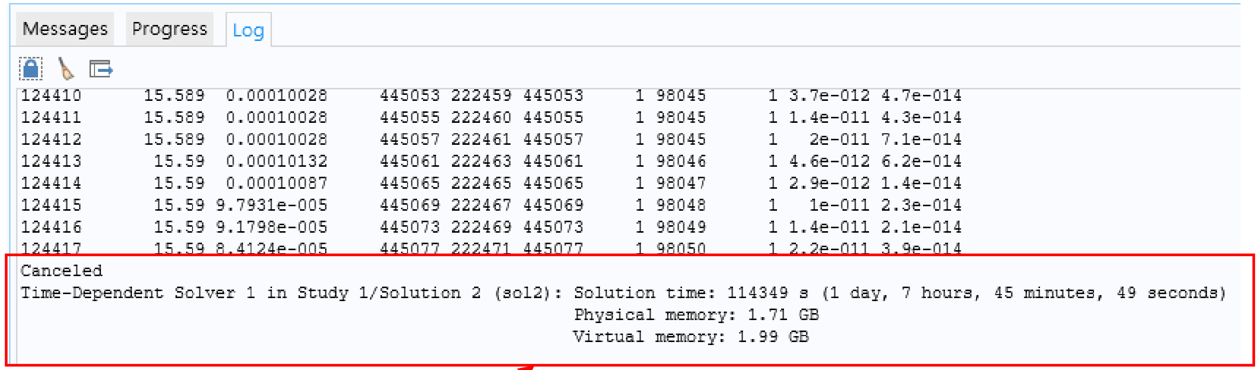
Half Model



CAE Project

Body & Tail 3D CFD





1 day, 7 hours, 45 minutes

Solution time issue!!

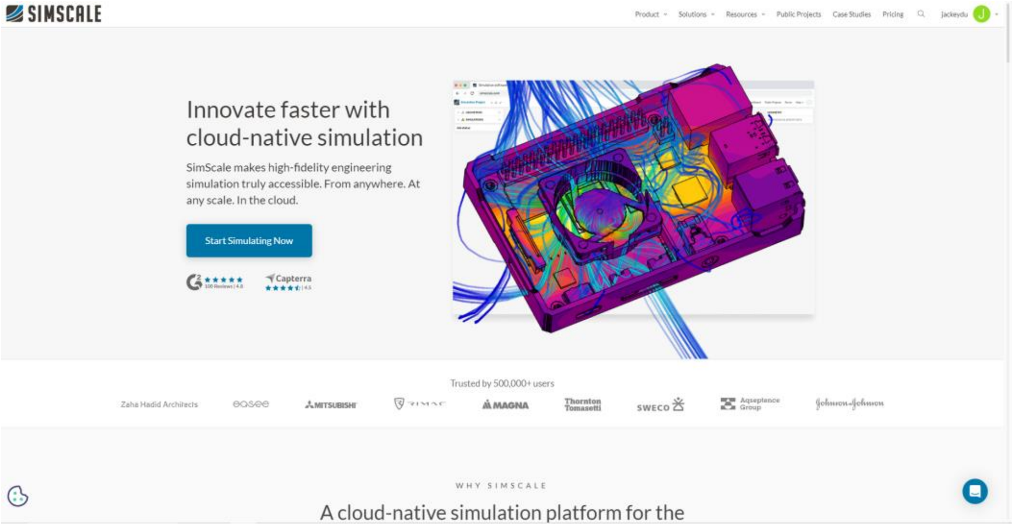
50 %

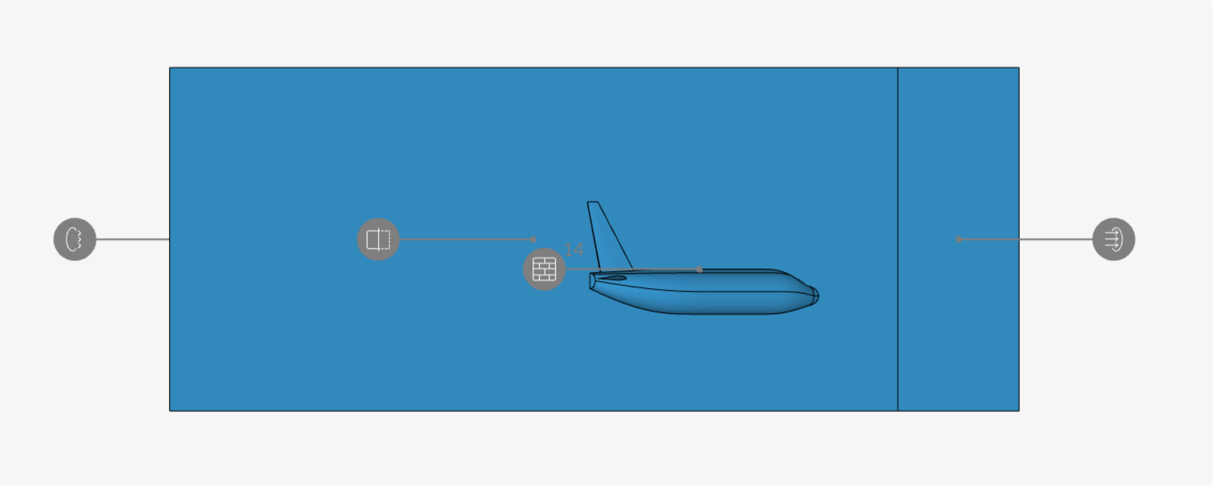
CAE Project

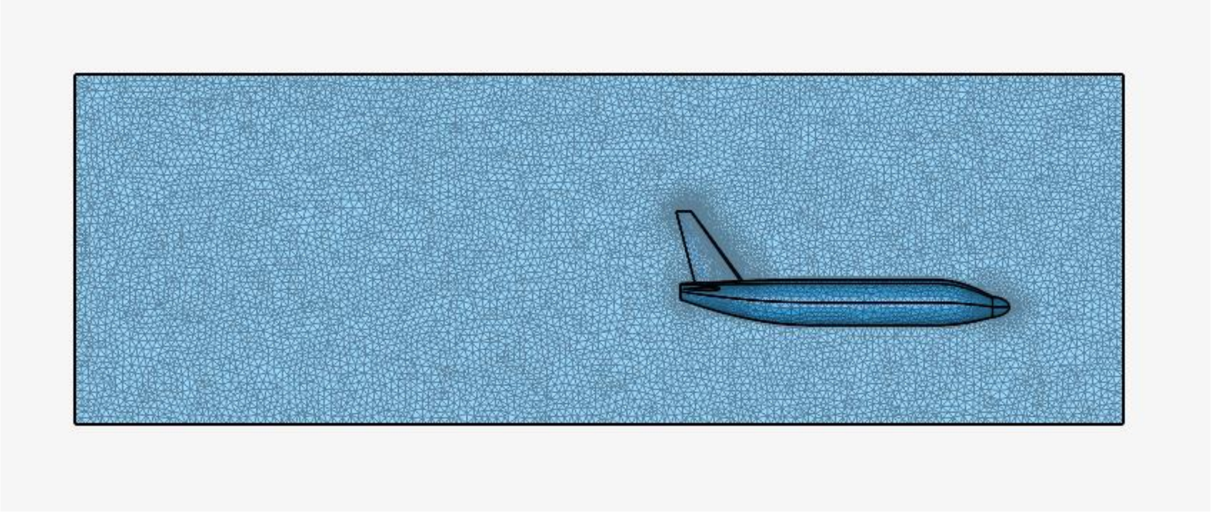
Body & Tail 3D CFD

SIMSCALE

Full-cloud CAE simulation software
Specialized in CFD

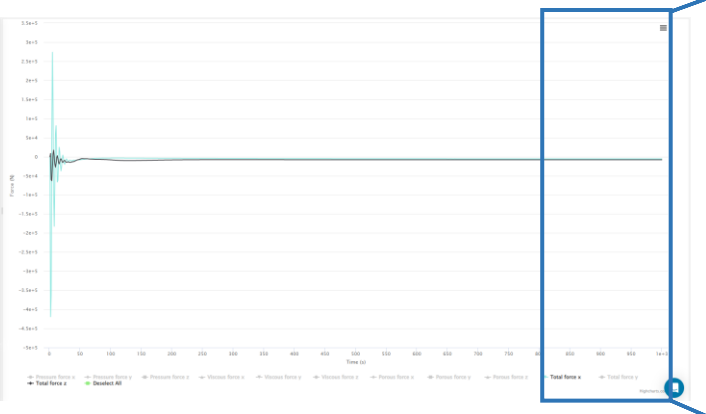


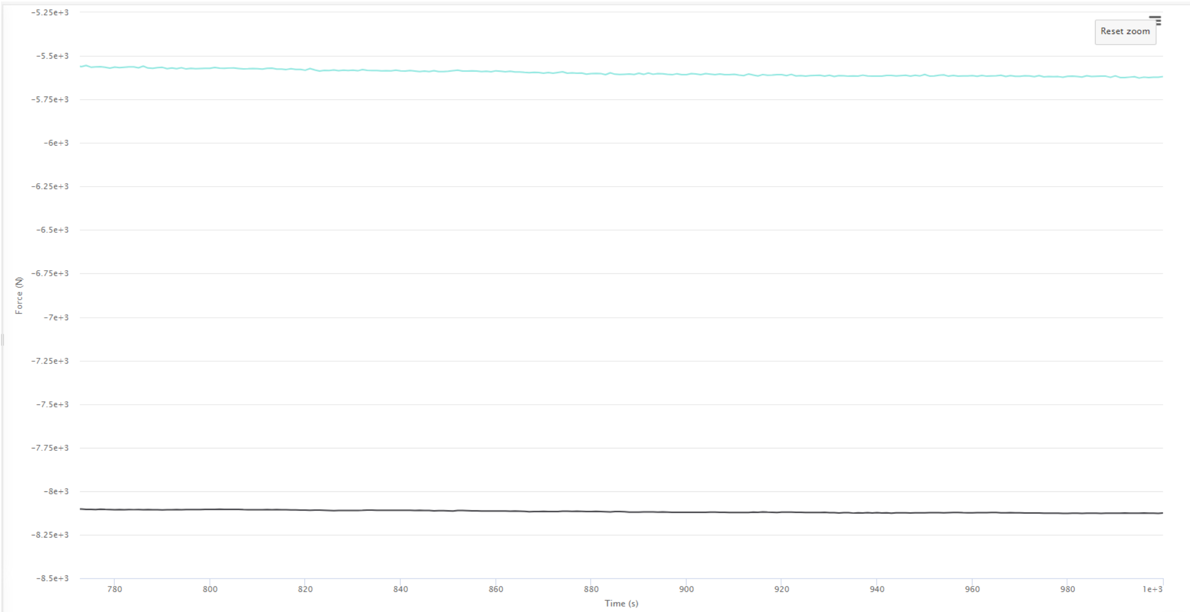


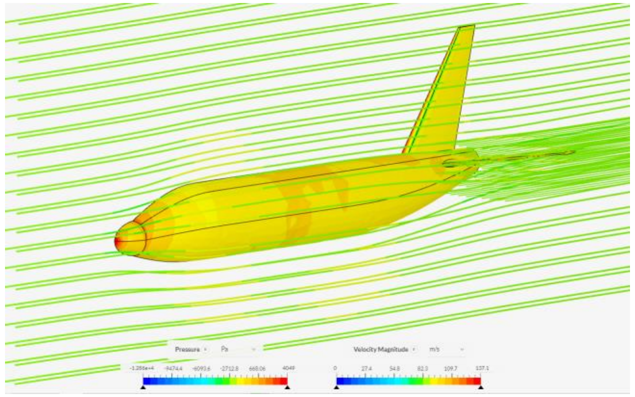


CAE Project

Body & Tail 3D CFD







CAE Project

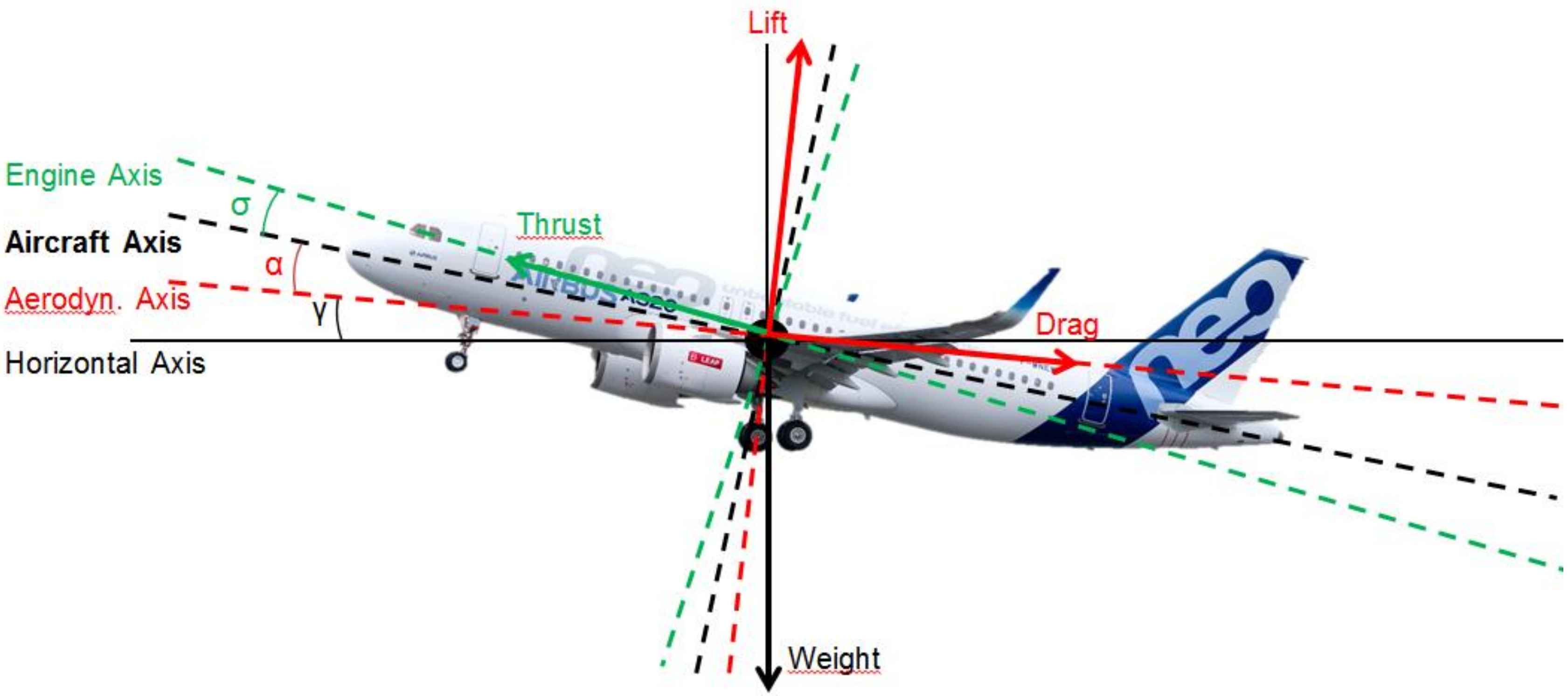
Body & Tail 3D CFD

Time (s)	POROUS_	POROUS_	POROUS_FORCE_Z	PRESSURE_FORCE_X	PRESSURE_FORCE_Y	PRESSURE_FORCE_Z	TOTAL_FORCE_X	TOTAL_FORCE_Y	TOTAL_FORCE_Z	VISCOUS_FORCE_X	VISCOUS_FORCE_Y	VISCOUS_FORCE_Z
1	0.00E+00	0.00E+00	0.00E+00	-9.38E+01	-4.83E+03	-1.08E+01	-9.41E+01	-4.83E+03	-1.08E+01	-3.17E-01	-5.61E-03	-1.92E-0
2	0.00E+00	0.00E+00	0.00E+00	-4.19E+05	-2.29E+07	8.41E+03	-4.19E+05	-2.29E+07	8.42E+03	-9.18E+01	2.53E+00	2.43E+0
3	0.00E+00	0.00E+00	0.00E+00	-3.61E+05	-1.77E+07	-6.00E+04	-3.61E+05	-1.77E+07	-6.00E+04	-9.20E+01	3.40E+00	2.53E+0
4	0.00E+00	0.00E+00	0.00E+00	4.78E+04	4.10E+06	-6.31E+04	4.77E+04	4.10E+06	-6.31E+04	-9.35E+01	3.59E+00	2.31E+00
5	0.00E+00	0.00E+00	0.00E+00	2.74E+05	1.53E+07	-2.84E+04	2.74E+05	1.53E+07	-2.84E+04	-9.19E+01	3.76E+00	2.04E+00
6	0.00E+00	0.00E+00	0.00E+00	1.49E+05	7.84E+06	7.19E+03	1.49E+05	7.84E+06	7.20E+03	-8.91E+01	3.84E+00	1.90E+00
7	0.00E+00	0.00E+00	0.00E+00	-9.55E+04	-5.16E+06	1.73E+04	-9.56E+04	-5.16E+06	1.73E+04	-8.74E+01	3.92E+00	1.88E+00
8	0.00E+00	0.00E+00	0.00E+00	-1.82E+05	-9.17E+06	5.83E+02	-1.82E+05	-9.17E+06	5.85E+02	-8.68E+01	3.98E+00	1.91E+00
9	0.00E+00	0.00E+00	0.00E+00	-7.49E+04	-2.86E+06	-2.06E+04	-7.50E+04	-2.86E+06	-2.06E+04	-8.63E+01	4.02E+00	1.90E+00
10	0.00E+00	0.00E+00	0.00E+00	6.00E+04	4.32E+06	-2.75E+04	5.99E+04	4.32E+06	-2.75E+04	-8.47E+01	4.03E+00	1.81E+00
11	0.00E+00	0.00E+00	0.00E+00	8.13E+04	5.02E+06	-1.69E+04	8.12E+04	5.02E+06	-1.69E+04	-8.23E+01	4.03E+00	1.71E+00
12	0.00E+00	0.00E+00	0.00E+00	3.57E+03	5.11E+05	-2.12E+03	3.49E+03	5.11E+05	-2.12E+03	-7.99E+01	4.09E+00	1.63E+00
13	0.00E+00	0.00E+00	0.00E+00	-6.63E+04	-3.14E+06	2.65E+03	-6.64E+04	-3.14E+06	2.65E+03	-7.80E+01	4.22E+00	1.64E+00
14	0.00E+00	0.00E+00	0.00E+00	-6.20E+04	-2.55E+06	-4.95E+03	-6.21E+04	-2.55E+06	-4.95E+03	-7.66E+01	4.36E+00	1.67E+00
15	0.00E+00	0.00E+00	0.00E+00	-1.00E+04	4.12E+05	-1.55E+04	-1.01E+04	4.12E+05	-1.55E+04	-7.51E+01	4.46E+00	1.69E+00
16	0.00E+00	0.00E+00	0.00E+00	2.45E+04	2.10E+06	-1.89E+04	2.45E+04	2.10E+06	-1.89E+04	-7.34E+01	4.49E+00	1.71E+00
17	0.00E+00	0.00E+00	0.00E+00	1.29E+04	1.21E+06	-1.34E+04	1.28E+04	1.21E+06	-1.34E+04	-7.16E+01	4.47E+00	1.72E+00
18	0.00E+00	0.00E+00	0.00E+00	-2.04E+04	-5.74E+05	-6.72E+03	-2.05E+04	-5.74E+05	-6.71E+03	-6.99E+01	4.43E+00	1.73E+00
19	0.00E+00	0.00E+00	0.00E+00	-3.72E+04	-1.23E+06	-5.28E+03	-3.72E+04	-1.23E+06	-5.28E+03	-6.87E+01	4.38E+00	1.75E+00
20	0.00E+00	0.00E+00	0.00E+00	-2.59E+04	-4.41E+05	-9.19E+03	-2.60E+04	-4.41E+05	-9.18E+03	-6.76E+01	4.33E+00	1.74E+00
21	0.00E+00	0.00E+00	0.00E+00	-4.85E+03	5.76E+05	-1.38E+04	-4.92E+03	5.76E+05	-1.38E+04	-6.65E+01	4.26E+00	1.71E+00
22	0.00E+00	0.00E+00	0.00E+00	4.24E+03	7.57E+05	-1.50E+04	4.18E+03	7.57E+05	-1.50E+04	-6.53E+01	4.18E+00	1.67E+00
23	0.00E+00	0.00E+00	0.00E+00	-3.22E+03	1.74E+05	-1.26E+04	-3.28E+03	1.74E+05	-1.26E+04	-6.43E+01	4.05E+00	1.62E+00
24	0.00E+00	0.00E+00	0.00E+00	-1.54E+04	-3.70E+05	-1.00E+04	-1.55E+04	-3.70E+05	-1.00E+04	-6.33E+01	3.91E+00	1.57E+00
25	0.00E+00	0.00E+00	0.00E+00	-1.99E+04	-3.50E+05	-9.83E+03	-2.00E+04	-3.50E+05	-9.83E+03	-6.24E+01	3.76E+00	1.52E+00
26	0.00E+00	0.00E+00	0.00E+00	-1.52E+04	4.40E+04	-1.17E+04	-1.53E+04	4.40E+04	-1.17E+04	-6.17E+01	3.61E+00	1.46E+00
27	0.00E+00	0.00E+00	0.00E+00	-8.24E+03	3.16E+05	-1.39E+04	-8.30E+03	3.16E+05	-1.39E+04	-6.10E+01	3.46E+00	1.40E+00
28	0.00E+00	0.00E+00	0.00E+00	-5.71E+03	2.36E+05	-1.48E+04	-5.78E+03	2.36E+05	-1.48E+04	-6.04E+01	3.32E+00	1.34E+00
29	0.00E+00	0.00E+00	0.00E+00	-8.26E+03	-1.28E+04	-1.43E+04	-8.32E+03	-1.28E+04	-1.43E+04	-5.98E+01	3.18E+00	1.29E+00
30	0.00E+00	0.00E+00	0.00E+00	-1.18E+04	-1.40E+05	-1.36E+04	-1.19E+04	-1.40E+05	-1.36E+04	-5.92E+01	3.06E+00	1.24E+00
31	0.00E+00	0.00E+00	0.00E+00	-1.28E+04	-5.91E+04	-1.35E+04	-1.29E+04	-5.91E+04	-1.35E+04	-5.88E+01	2.94E+00	1.20E+00
32	0.00E+00	0.00E+00	0.00E+00	-1.11E+04	8.70E+04	-1.42E+04	-1.12E+04	8.70E+04	-1.42E+04	-5.84E+01	2.82E+00	1.15E+00
33	0.00E+00	0.00E+00	0.00E+00	-9.15E+03	1.36E+05	-1.48E+04	-9.21E+03	1.36E+05	-1.48E+04	-5.81E+01	2.70E+00	1.11E+00
34	0.00E+00	0.00E+00	0.00E+00	-8.78E+03	6.72E+04	-1.49E+04	-8.84E+03	6.72E+04	-1.49E+04	-5.79E+01	2.59E+00	1.07E+00
35	0.00E+00	0.00E+00	0.00E+00	-9.88E+03	-1.85E+04	-1.45E+04	-9.94E+03	-1.85E+04	-1.45E+04	-5.77E+01	2.49E+00	1.04E+00

Total Lift	Total Drag
-8.12E+03	5.60E+03
Lift/v^2	Drag/v^2
-1.27E+00	8.75E-01

Take off situation

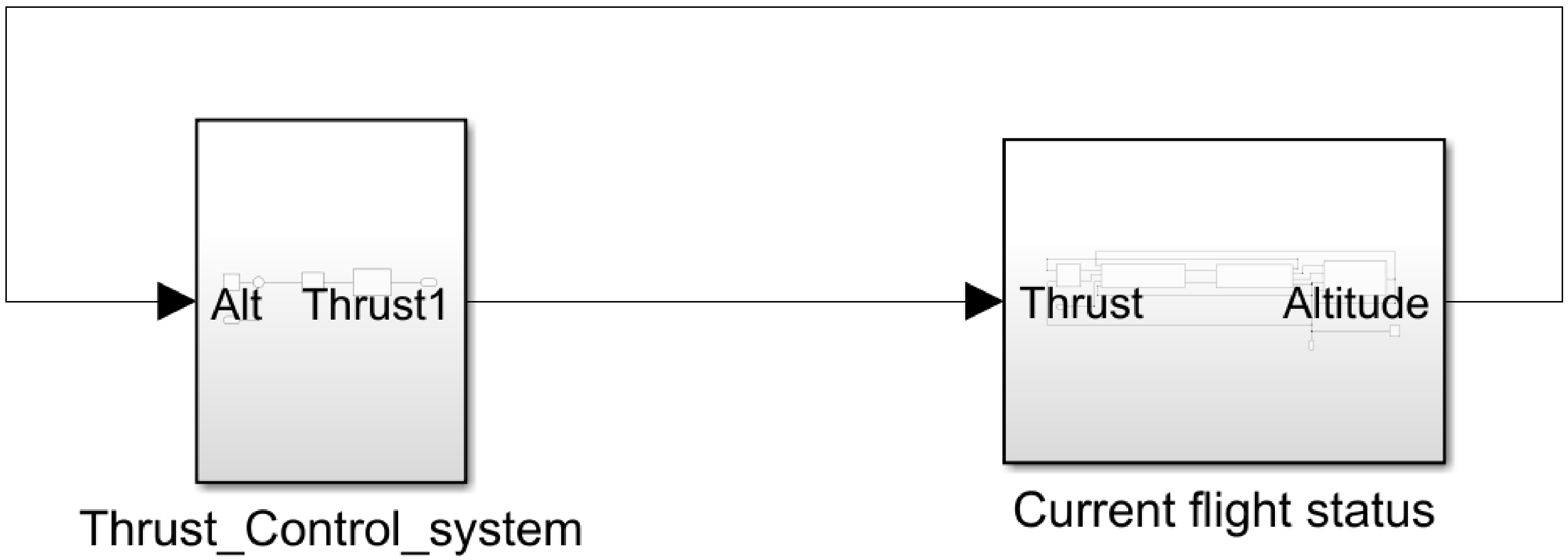
Take – off



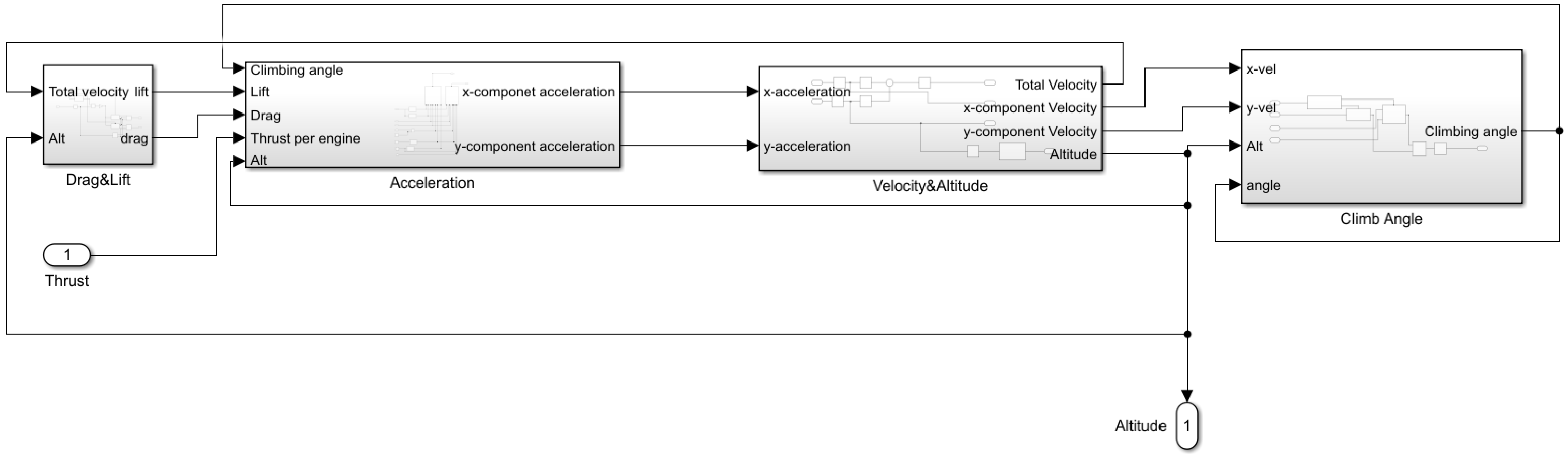
Engine Axis = Aircraft Axis
To negelect an effect of Aerodyn. Axis, set a climb angle by velocity
Maintaining altitude at 12000m

Build a system using simulink

System component



Current flight status



Drag&Lift

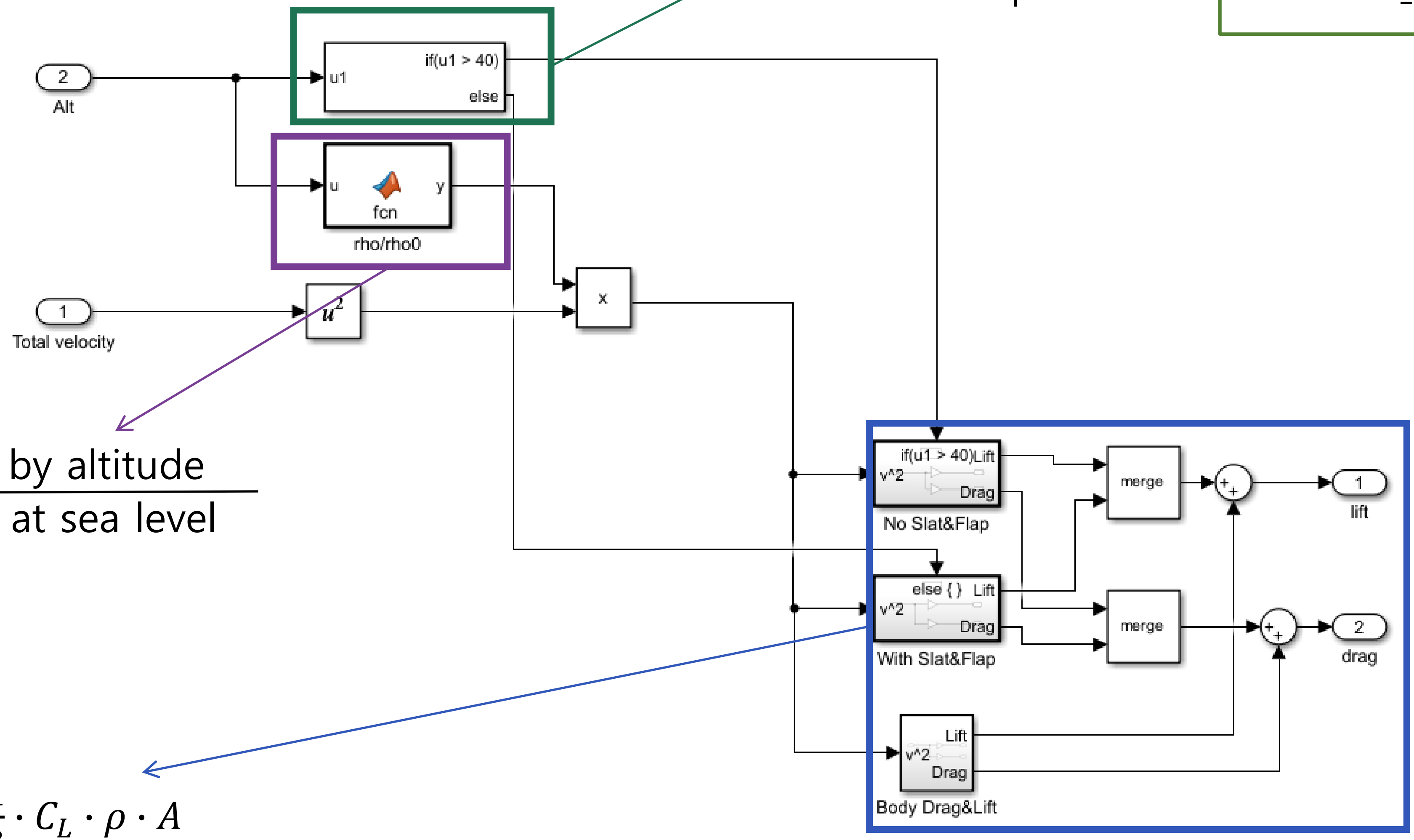
CAE Project

Drag&Lift

Input -Altitude
-Total velocity

Output - Lift
- Drag

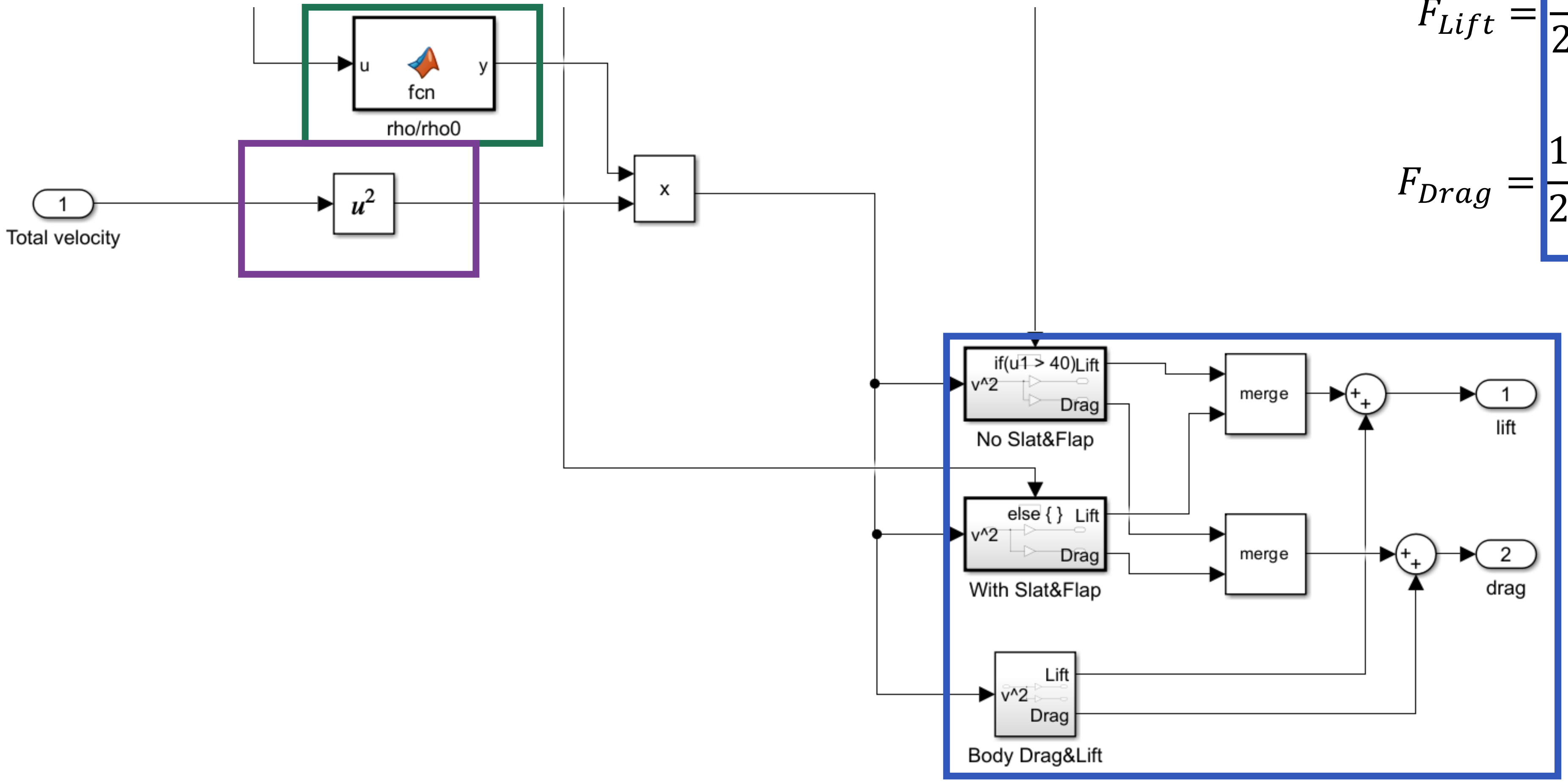
If altitude > 40m
No slat&flap



$$\frac{\text{Air density by altitude}}{\text{Air density at sea level}}$$

Multiply $\frac{1}{2} \cdot C_L \cdot \rho \cdot A$
Merge all drag&lift

Drag&Lift



$$F_{Lift} = \frac{1}{2} \cdot C_L \cdot \rho \cdot A \cdot v^2$$

$$F_{Drag} = \frac{1}{2} \cdot C_D \cdot \rho \cdot A \cdot v^2$$

CAE Project

Air density by altitude / Air density at sea level

rho/rho0 | Acceleration

Boeing_737_500_5 ▸ Current flight status ▸ Drag&Lift ▸ rho/rho0

```matlab
function y = fcn(u)

p0 = 101325;
M = 0.0289652;
R = 8.31446;
T0 = 288.15;
L = 0.0065;
h = u;
g = 9.81;
T = T0-L*h

rho_0 = (p0*M)/(R*T0)*(1)^(-(g*M)/(R*L));

rho = (p0*M)/(R*T)*(1+(L*h)/T0)^(-(g*M)/(R*L));

y = rho/rho_0;
```

p0 = 101325	Pa	sea level standard atmospheric pressure
T0 = 288.15	K	sea level standard temperature
g = 9.80665	m/s2	Earth-surface gravitational acceleration
L = 0.0065	K/m	temperature lapse rate
R = 8.31447	J/(mol·K)	universal gas constant
M = 0.0289644	kg/mol	molar mass of dry air

$$p = p_0 \cdot \left(1 + \frac{L \cdot h}{T_0}\right)^{\frac{g \cdot M}{-R \cdot L}}$$

$$\rho = \frac{p}{R \cdot T}$$

Acceleration

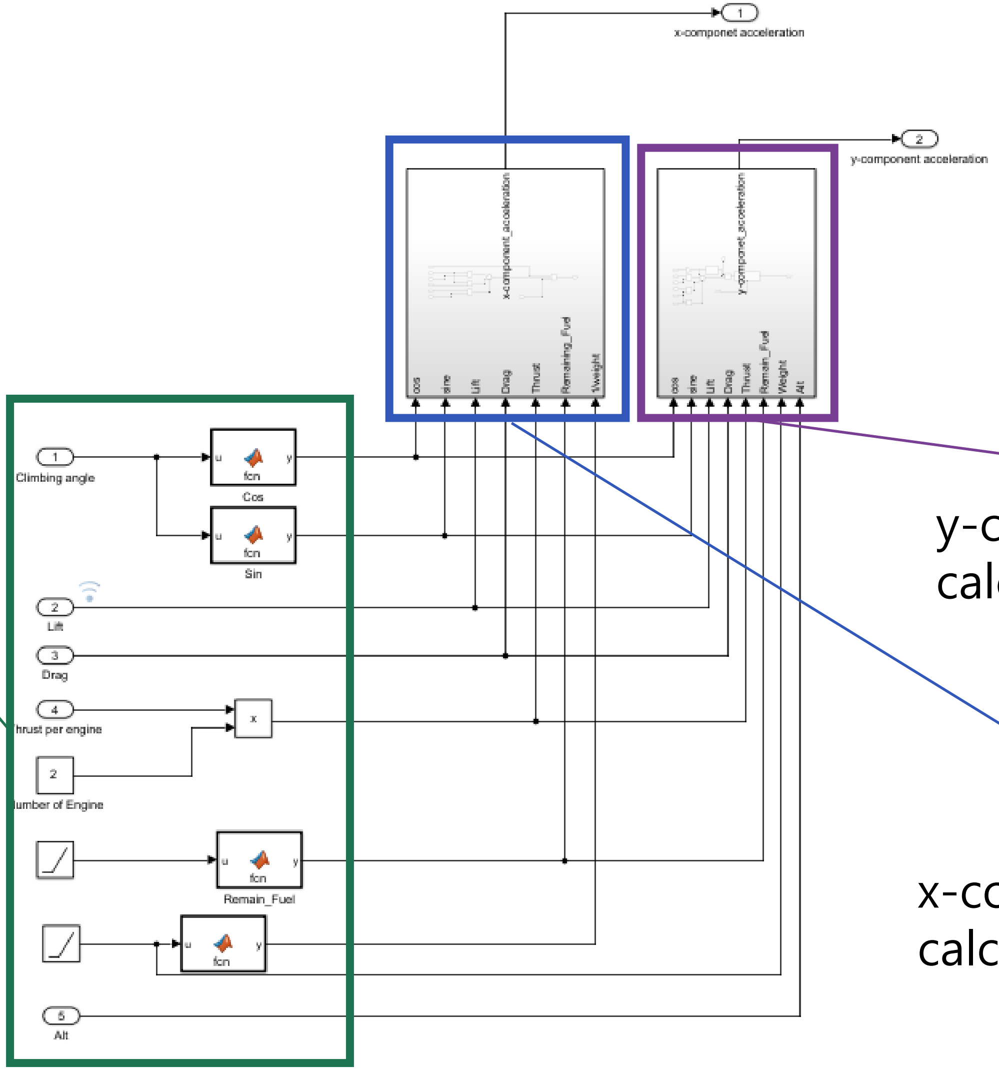
CAE Project

Acceleration

Input
- -Climb angle
- -Lift
- -Drag
- -Thrust per engine
- -Altitude

Output
- - x-component acceleration
- - y-component acceleration

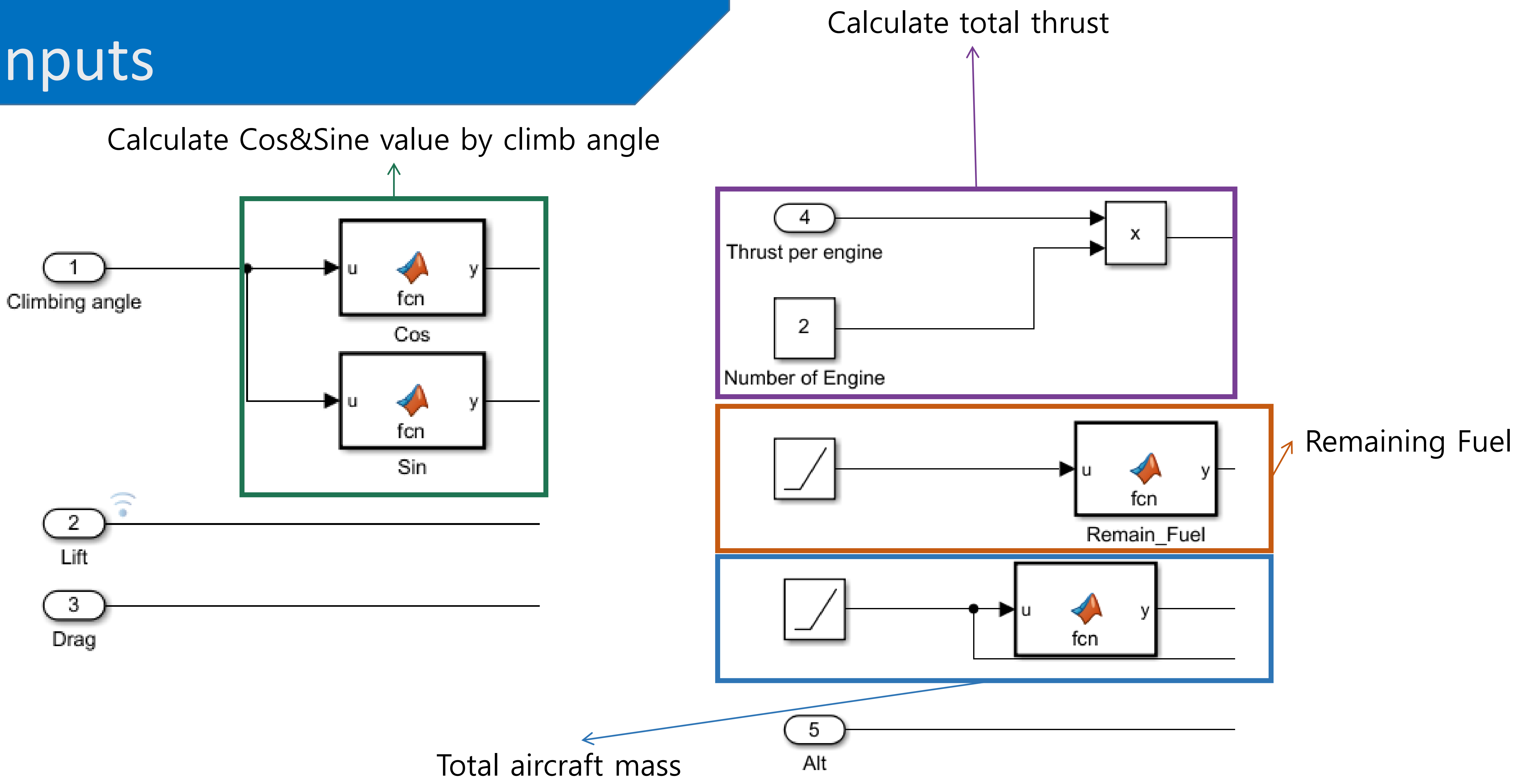
Inputs



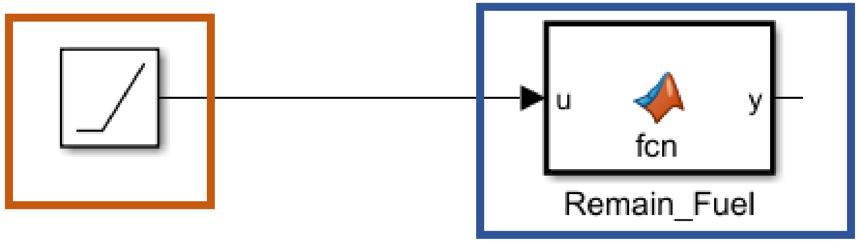
y-component acceleration calculation sub-system

x-component acceleration calculation sub-system

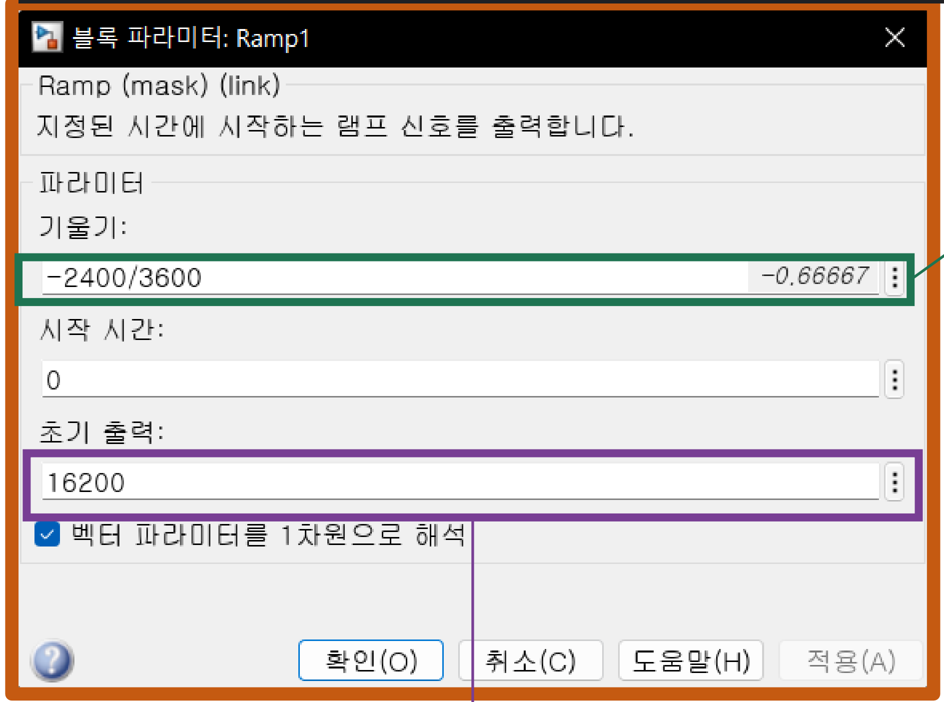
Inputs


Calculate Cos&Sine value by climb angle
1
Climbing angle
u
y
fcn
Cos
u
y
fcn
Sin
2
Lift
3
Drag
Calculate total thrust
4
Thrust per engine
x
2
Number of Engine
Remaining Fuel
u
y
fcn
Remain_Fuel
u
y
fcn
Total aircraft mass
5
Alt

Inputs – Remaing Fuel



Manufacturer	Aircraft	Fuel Consumption (kilogram/hour)
Boeing	737-500	2400

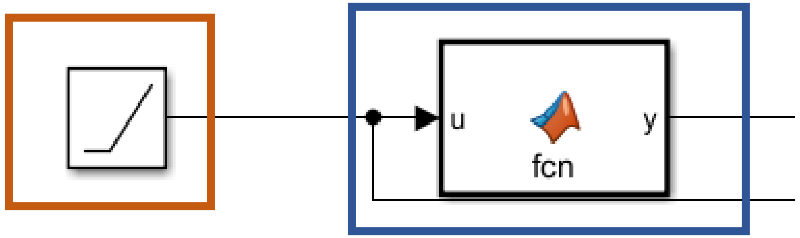


Fuel consumption 2400kg/hour

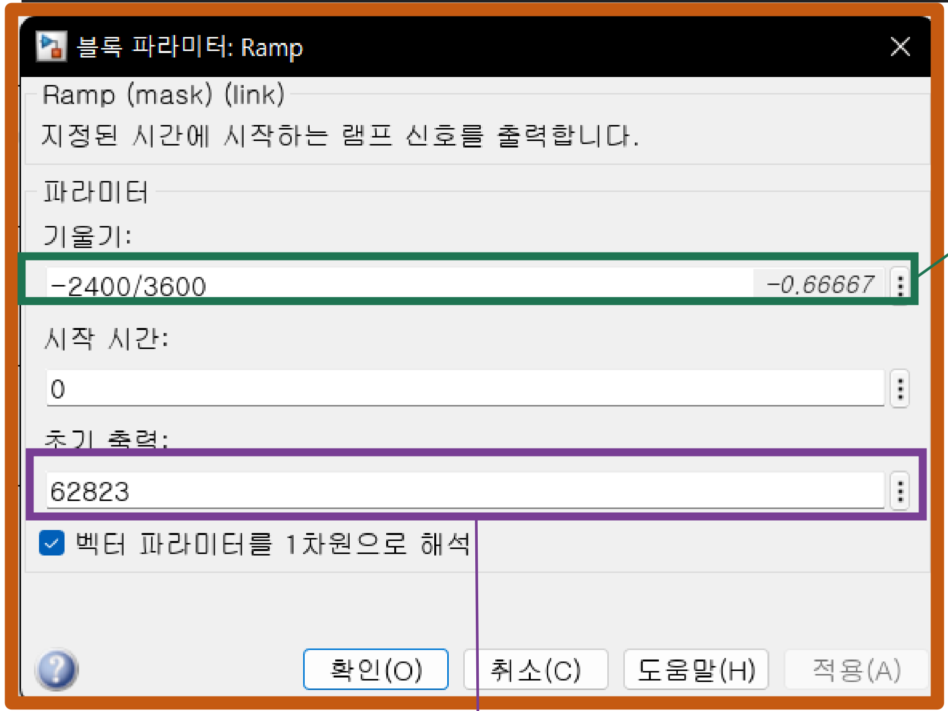
Max fuel mass

```
function y = fcn(u)

%연료 소비가 다 되었을 때 가속도를 0으로 만들어주는 함수

if u <= 0;
    y = 0;

else
    y = 1;

end
```

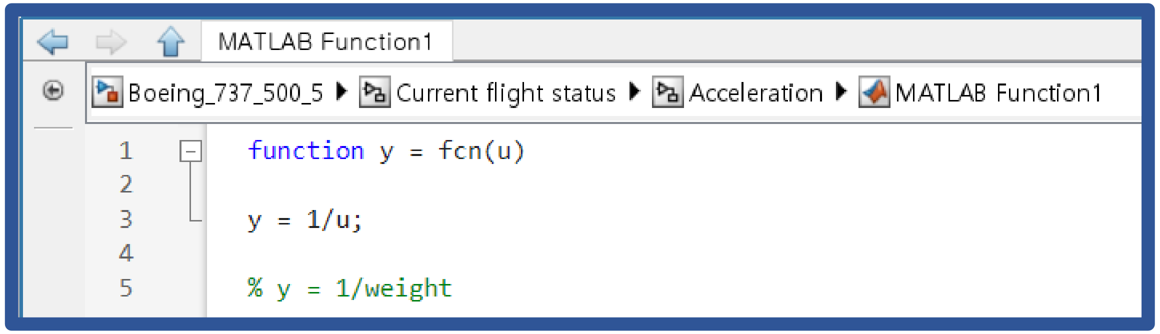
Inputs – Total aircraft weight



Manufacturer	Aircraft	Fuel Consumption (kilogram/hour)
Boeing	737-500	**2400**

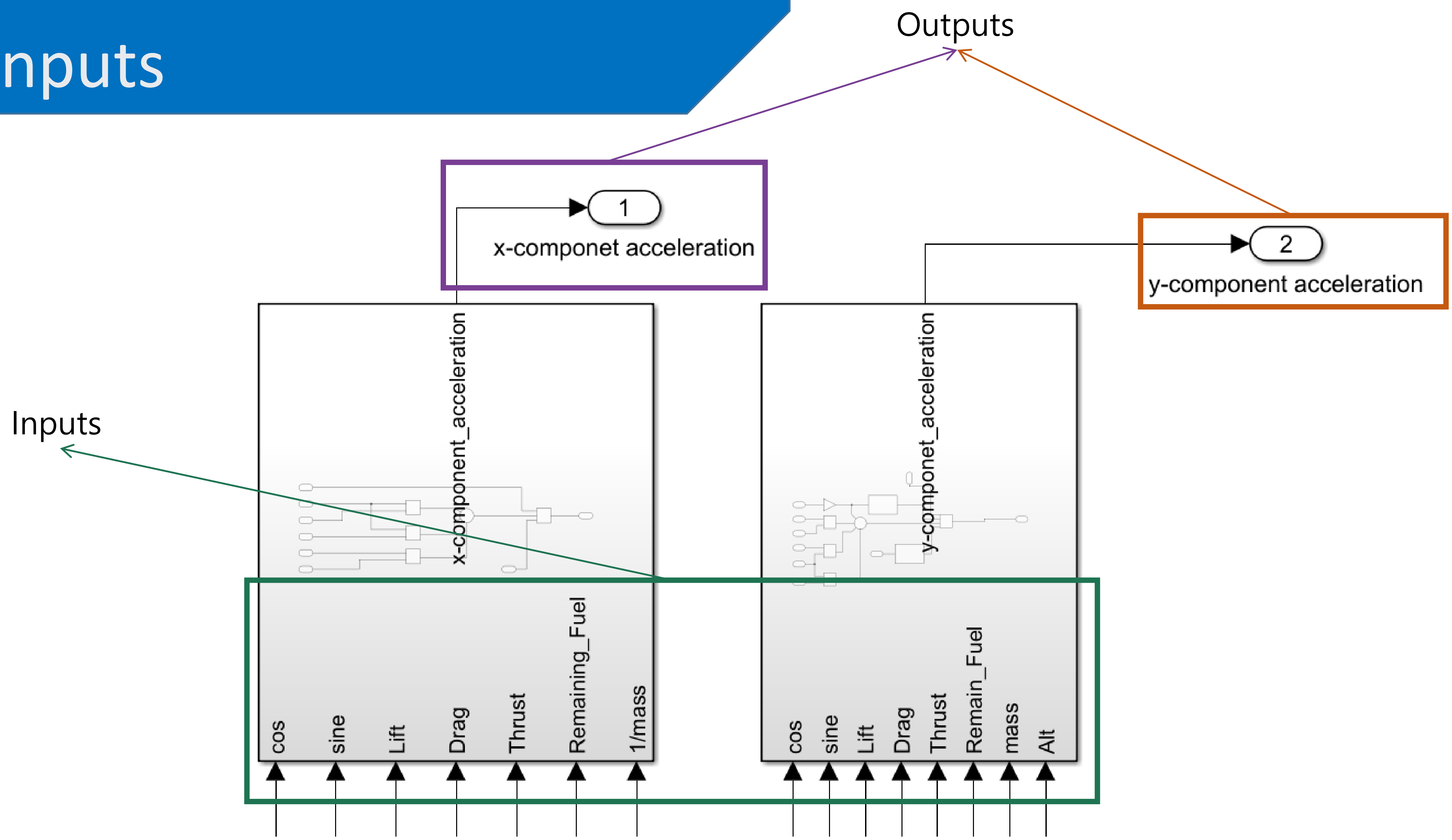


Fuel consumption 2400kg/hour

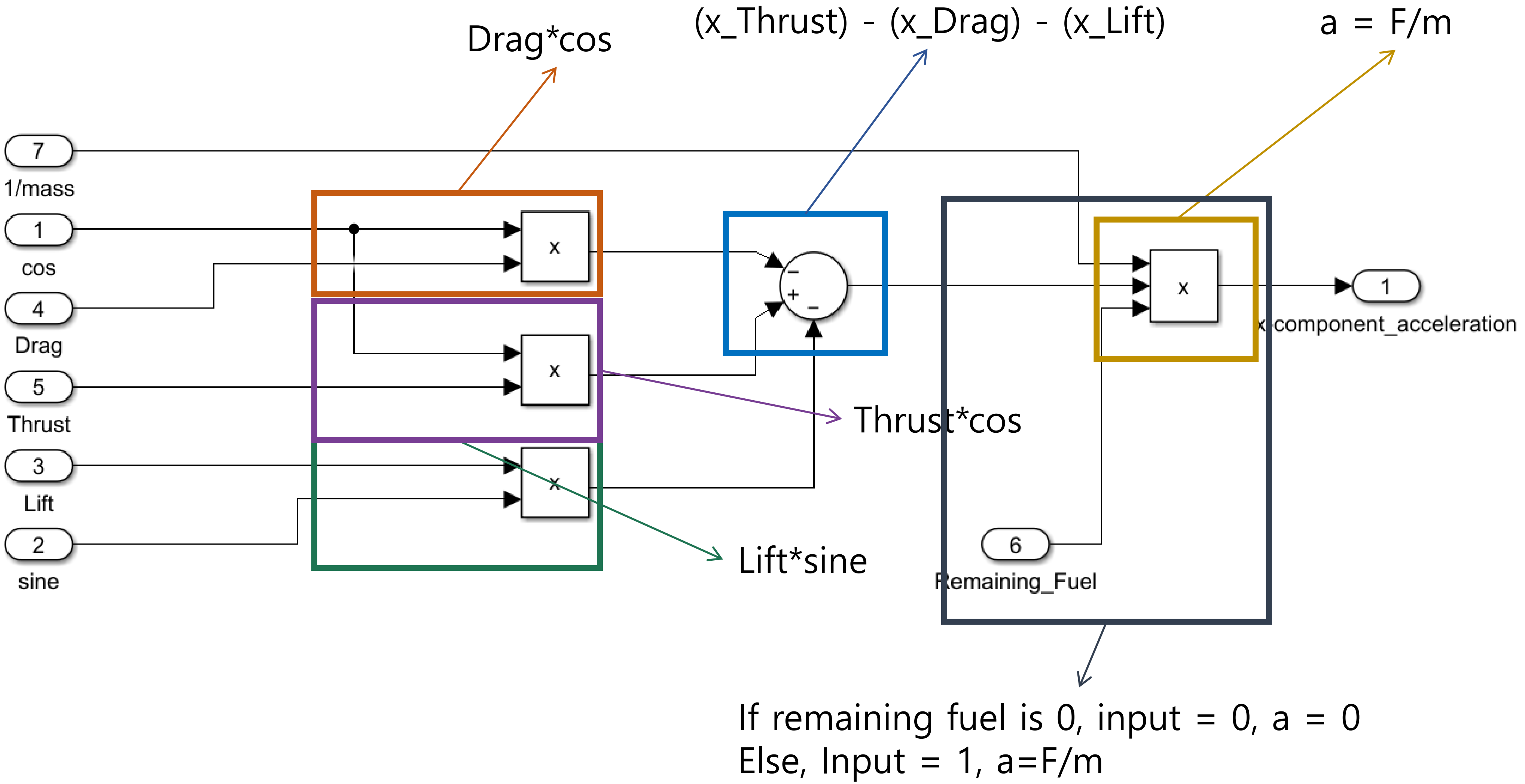


Max take-off mass

Inputs


Outputs
1
x-componet acceleration
2
y-component acceleration
x-component_acceleration
y-component_acceleration
Inputs
cos
sine
Lift
Drag
Thrust
Remaining_Fuel
1/mass
cos
sine
Lift
Drag
Thrust
Remain_Fuel
mass
Alt

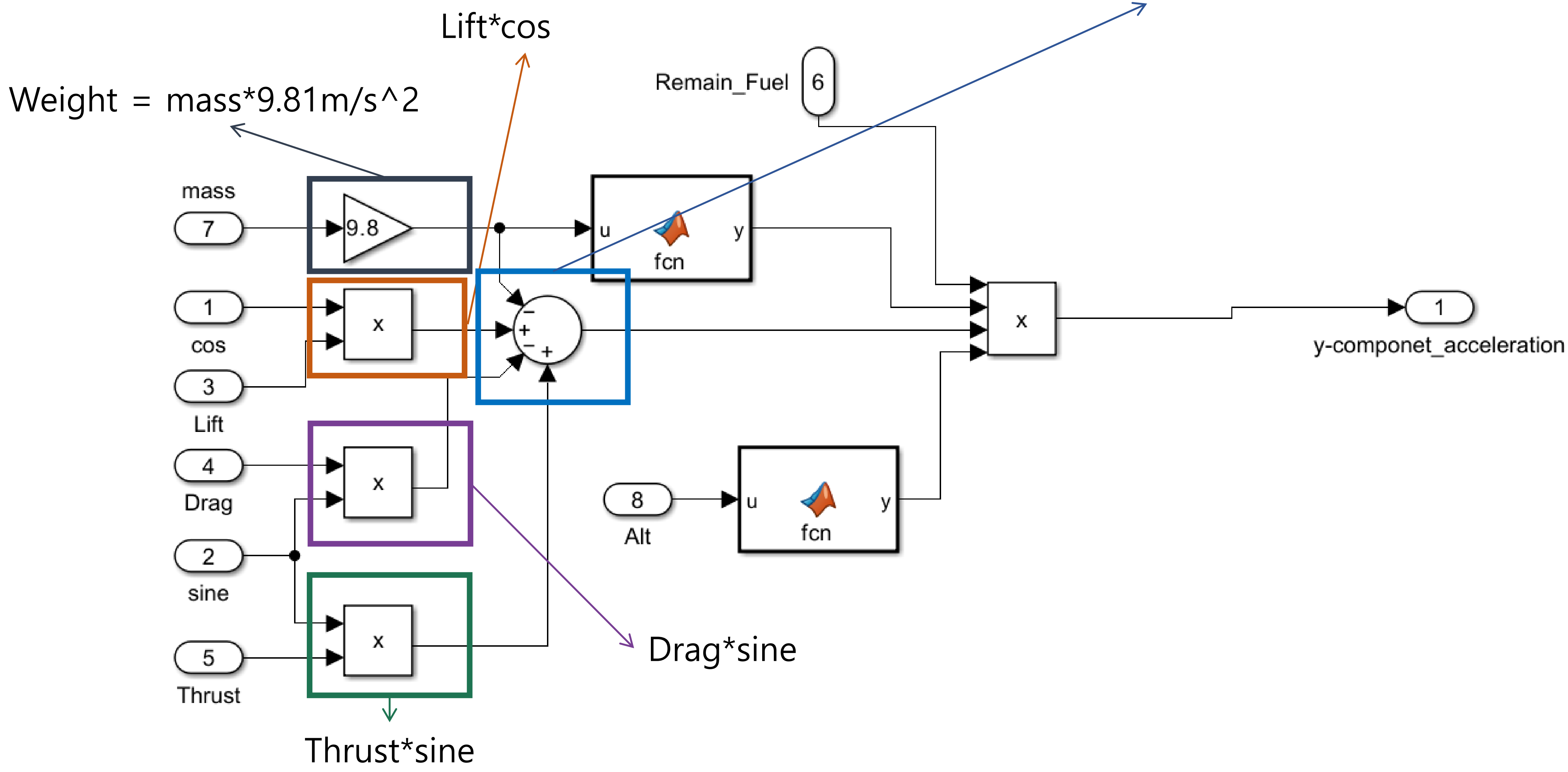
x-component acceleration calculation sub-system



CAE Project

y-component acceleration calculation sub-system

(y_Thrust) + (x_Lift)- (x_Drag) - Weight

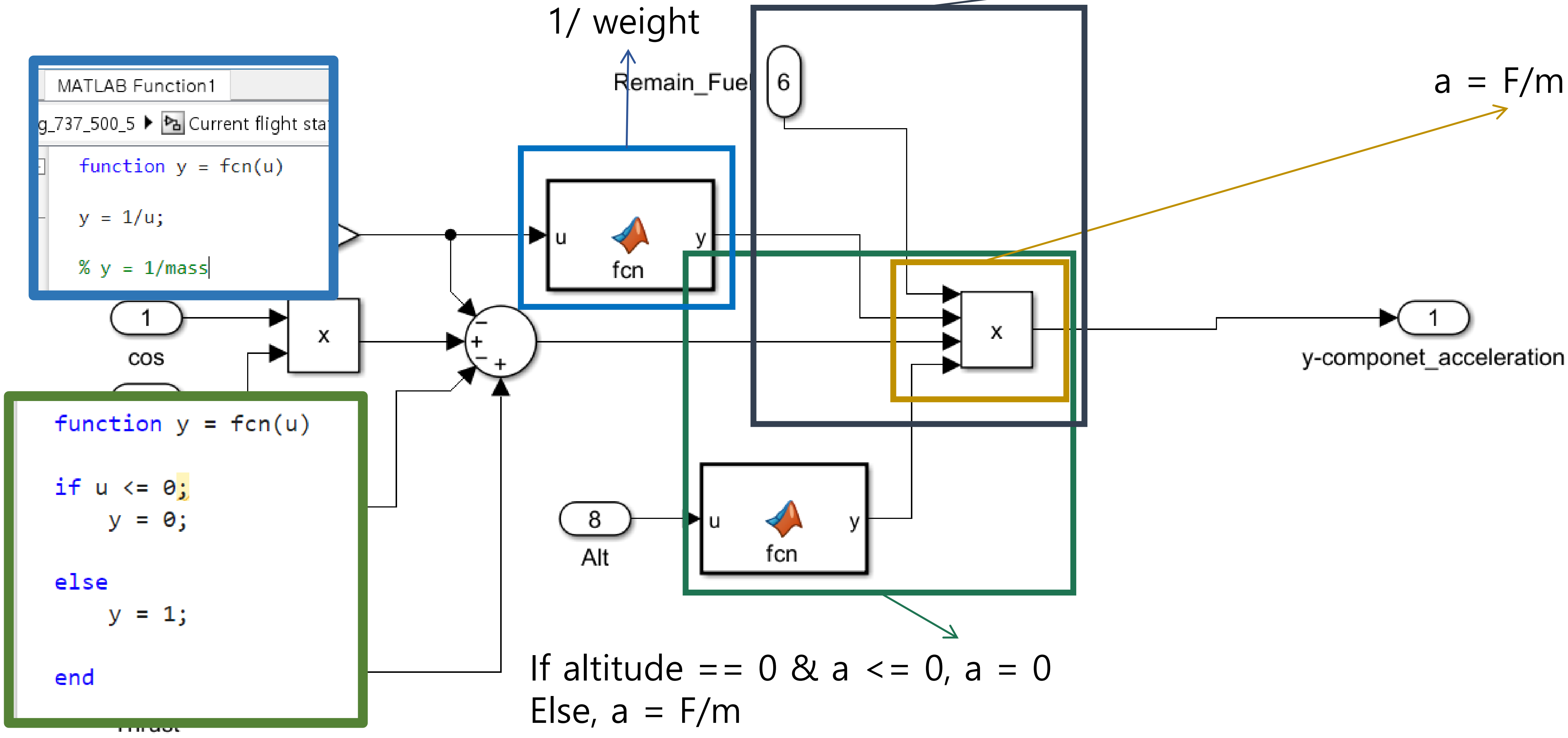


y-component acceleration calculation sub-system

If remaining fuel <= 0, input = 0, a = 0
Else, Input = 1, a=F/m

1/ weight

a = F/m

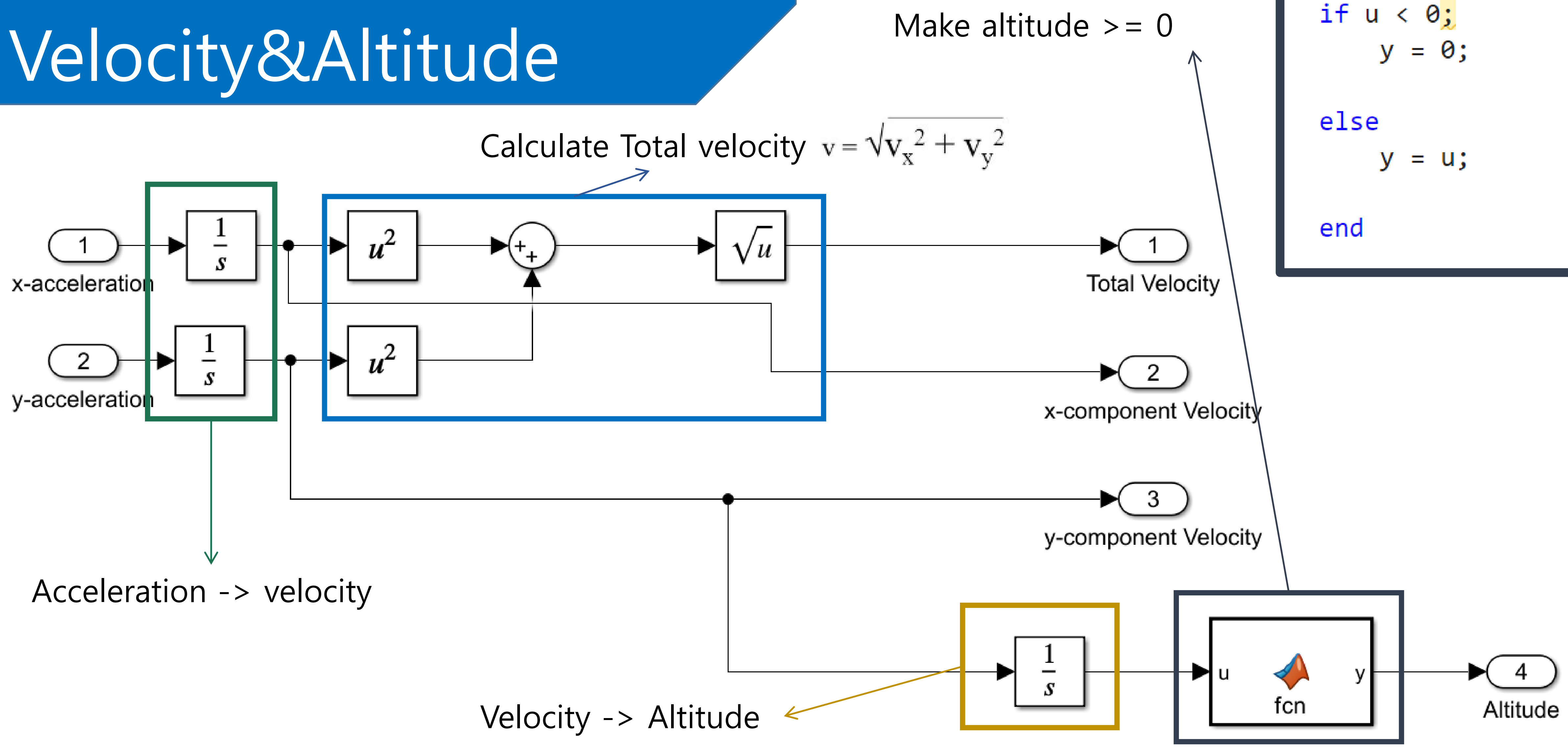


If altitude == 0 & a <= 0, a = 0
Else, a = F/m
With out this function, y-accel is negative when the altitude is 0. So, altitude will have a negative value.

Velocity&Altitude

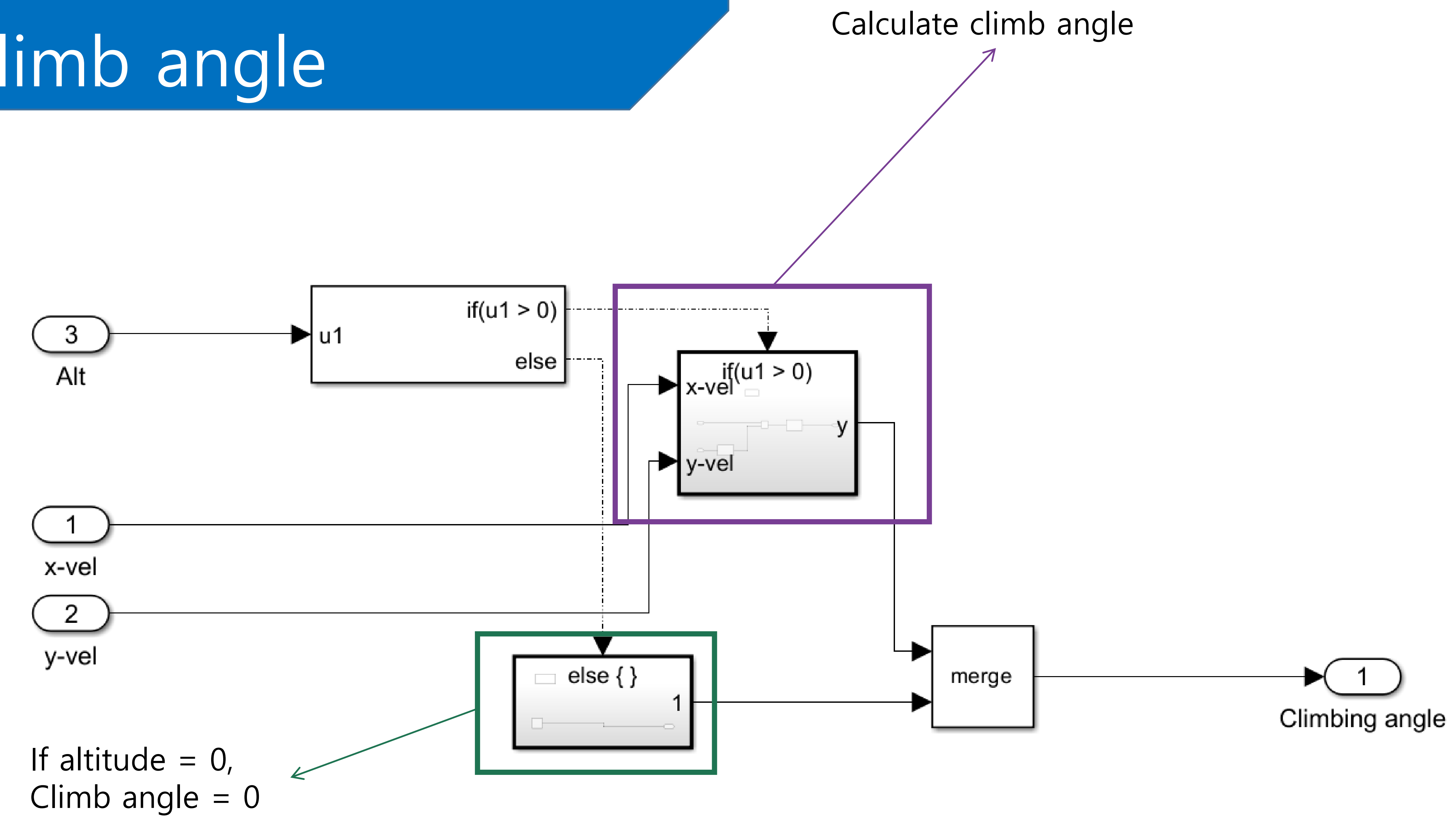
Velocity&Altitude

```
function y = fcn(u)

if u < 0;
    y = 0;

else
    y = u;

end
```



Climb Angle

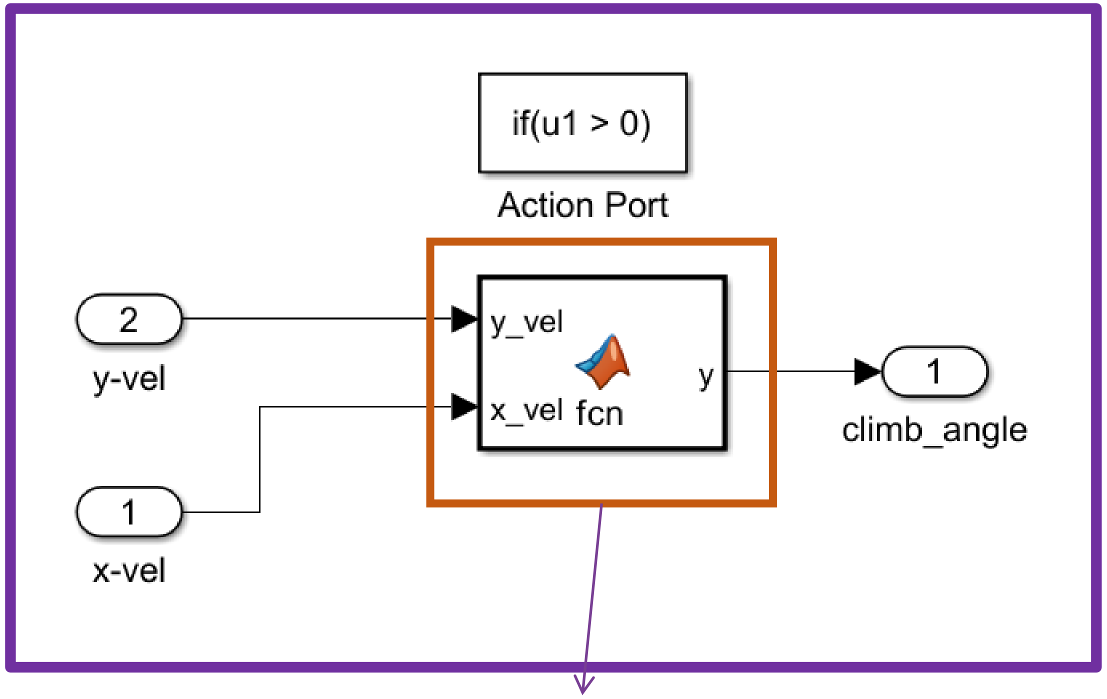
Climb angle


Calculate climb angle
3
Alt
u1
if(u1 > 0)
else
if(u1 > 0)
x-vel
y
y-vel
1
x-vel
2
y-vel
else { }
1
merge
1
Climbing angle
If altitude = 0,
Climb angle = 0

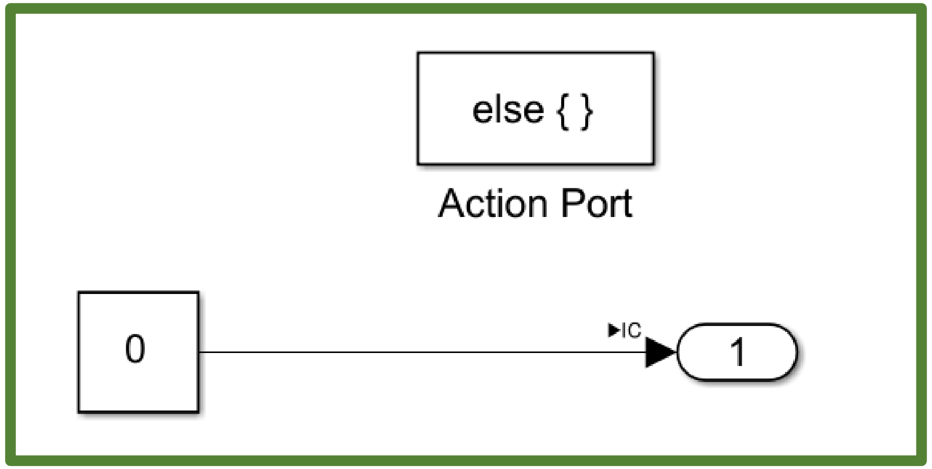
Climb angle

```
function y = fcn(y_vel, x_vel)

u = y_vel/x_vel;

y = atan(u);
```

By using this function, we can align aircraft axis and aerodyn. axis



Calculate climb angle

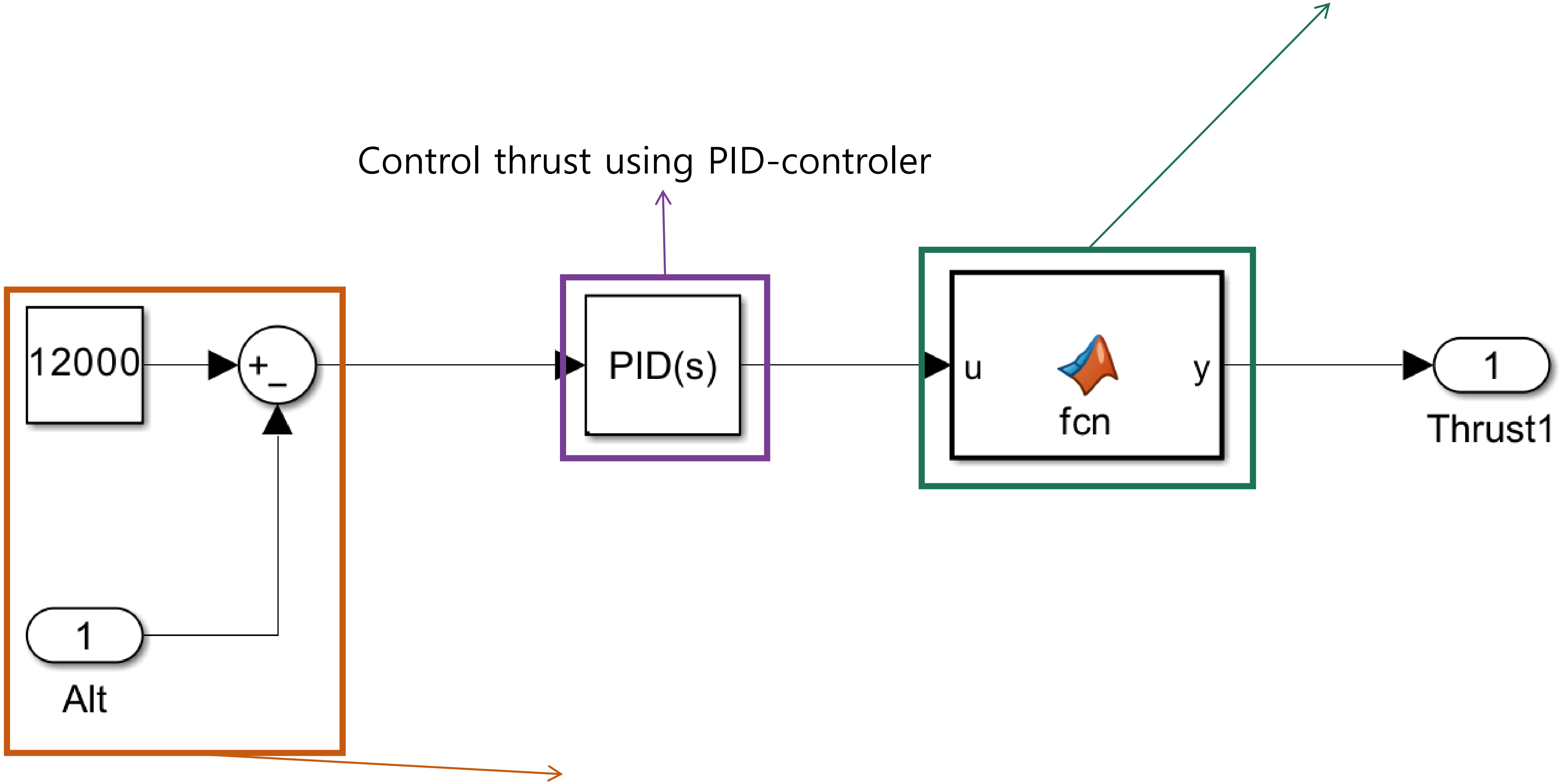


Thrust Control System

Thrust control system

Making the controlled thrust does not exceed the max thrust of the engine

Control thrust using PID-controler

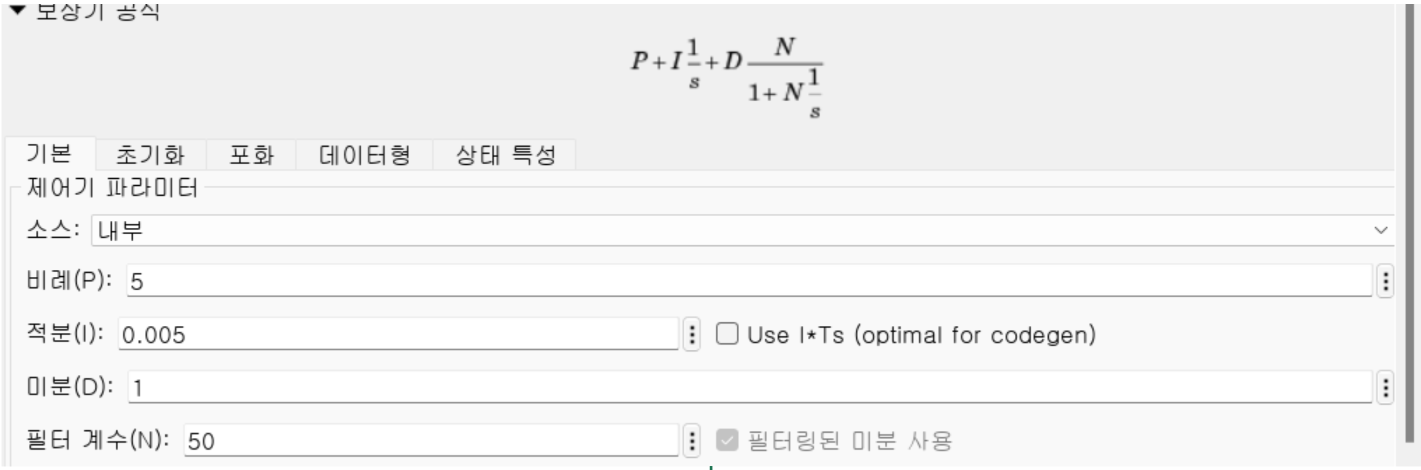


Calculating the difference between target altitude and current altitude

Thrust control system

the max thrust of the engine = 82.3kN

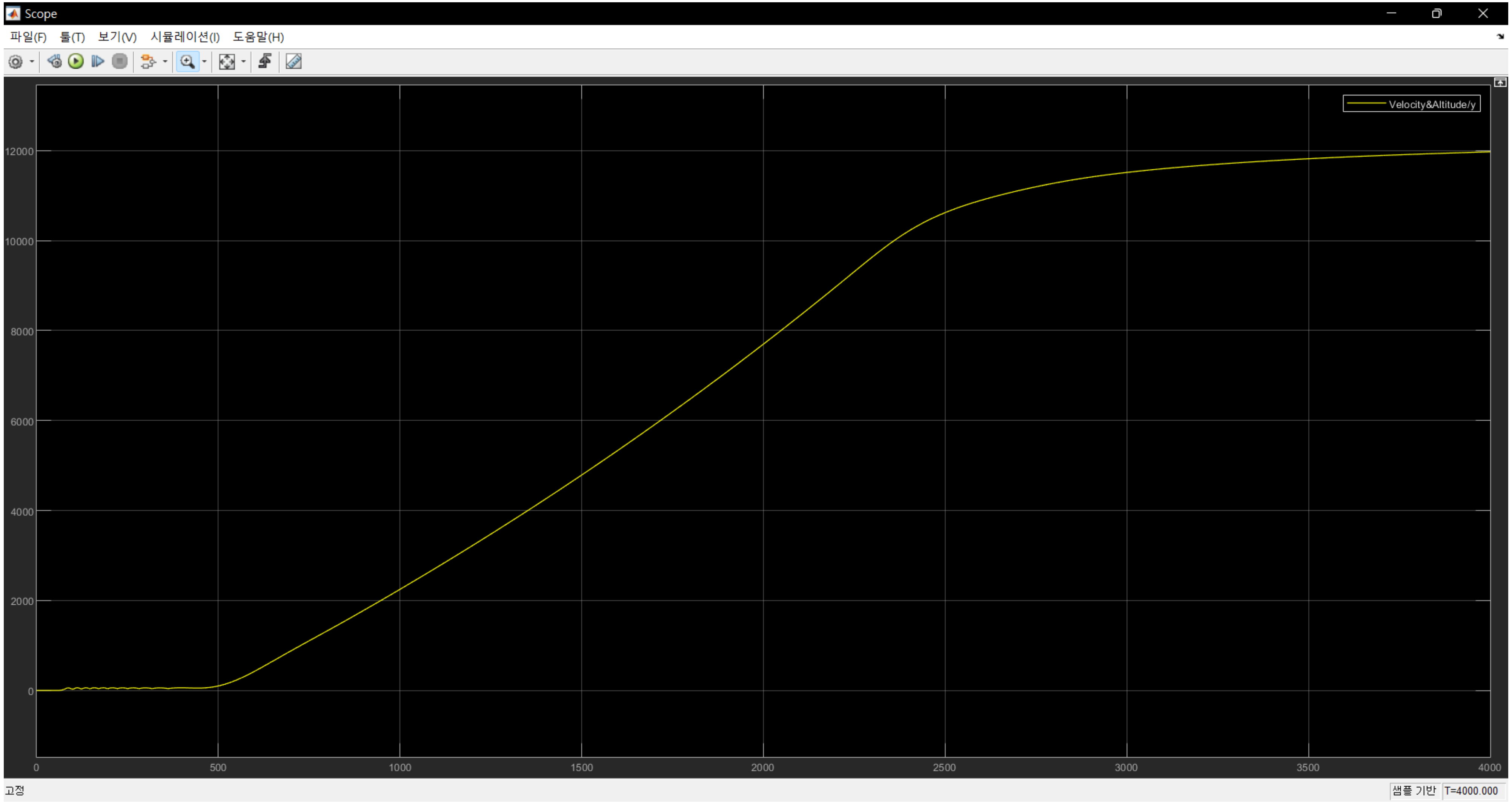
```
function y = fcn(u)

if u <= 82300
     y = u;

else
     y = 82300;
end
```



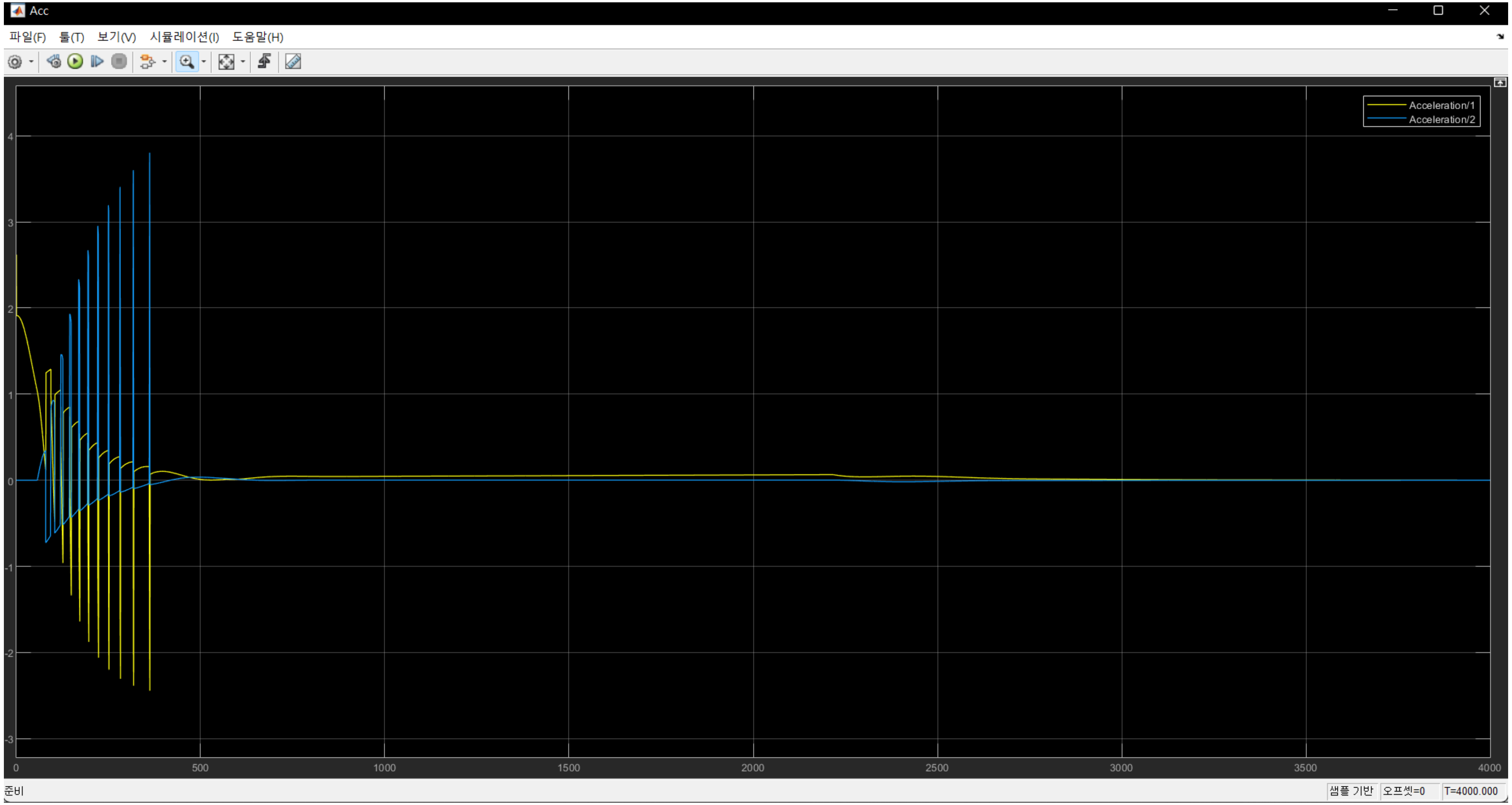
After trial and error, we adjusted the gain values.

Simulation

Altitude

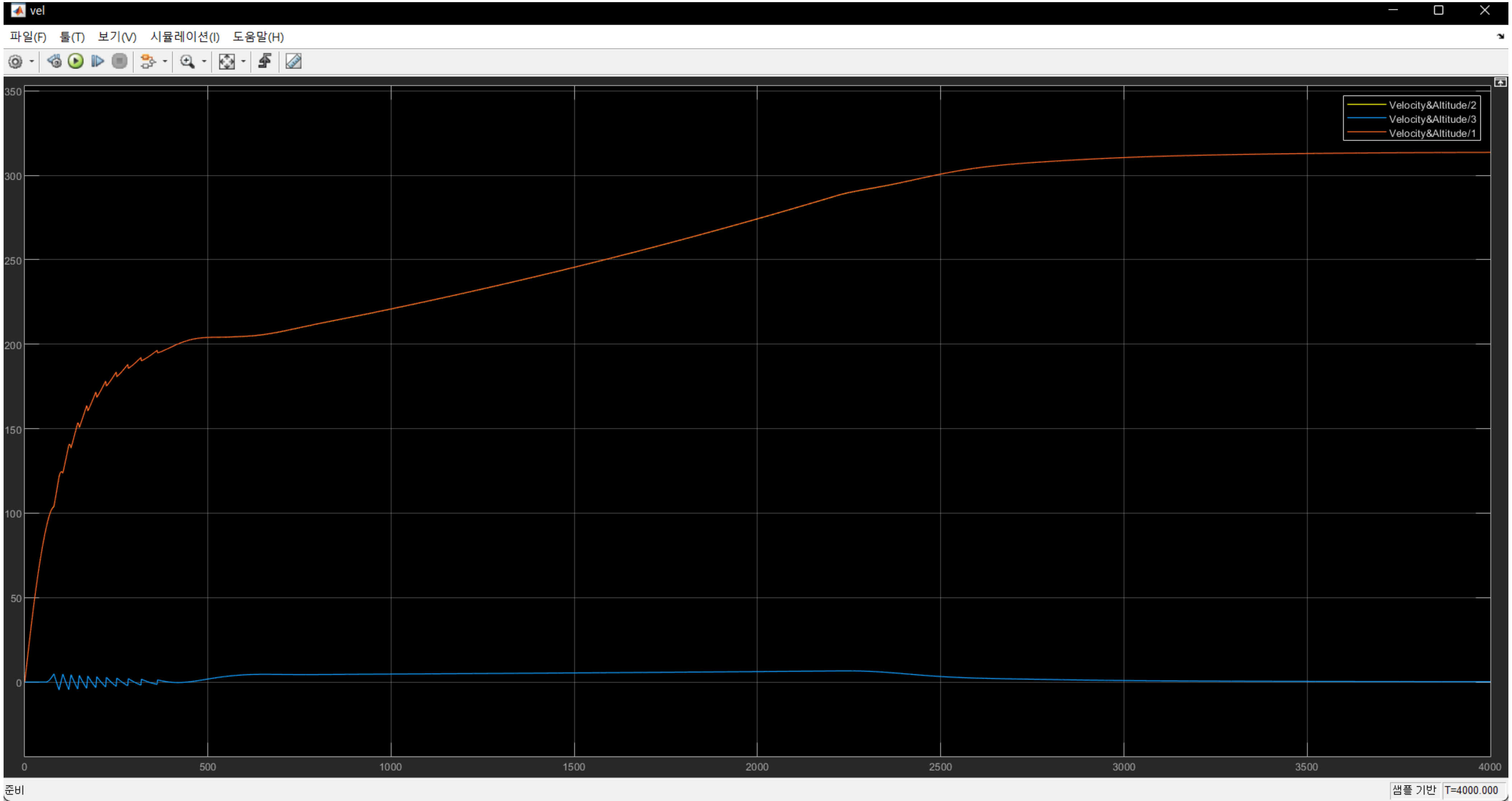

Scope
파일(F) 툴(T) 보기(V) 시뮬레이션(I) 도움말(H)
Velocity&Altitude/y
12000
10000
8000
6000
4000
2000
0
0
500
1000
1500
2000
2500
3000
3500
4000
고정
샘플 기반
T=4000.000

Acceleration

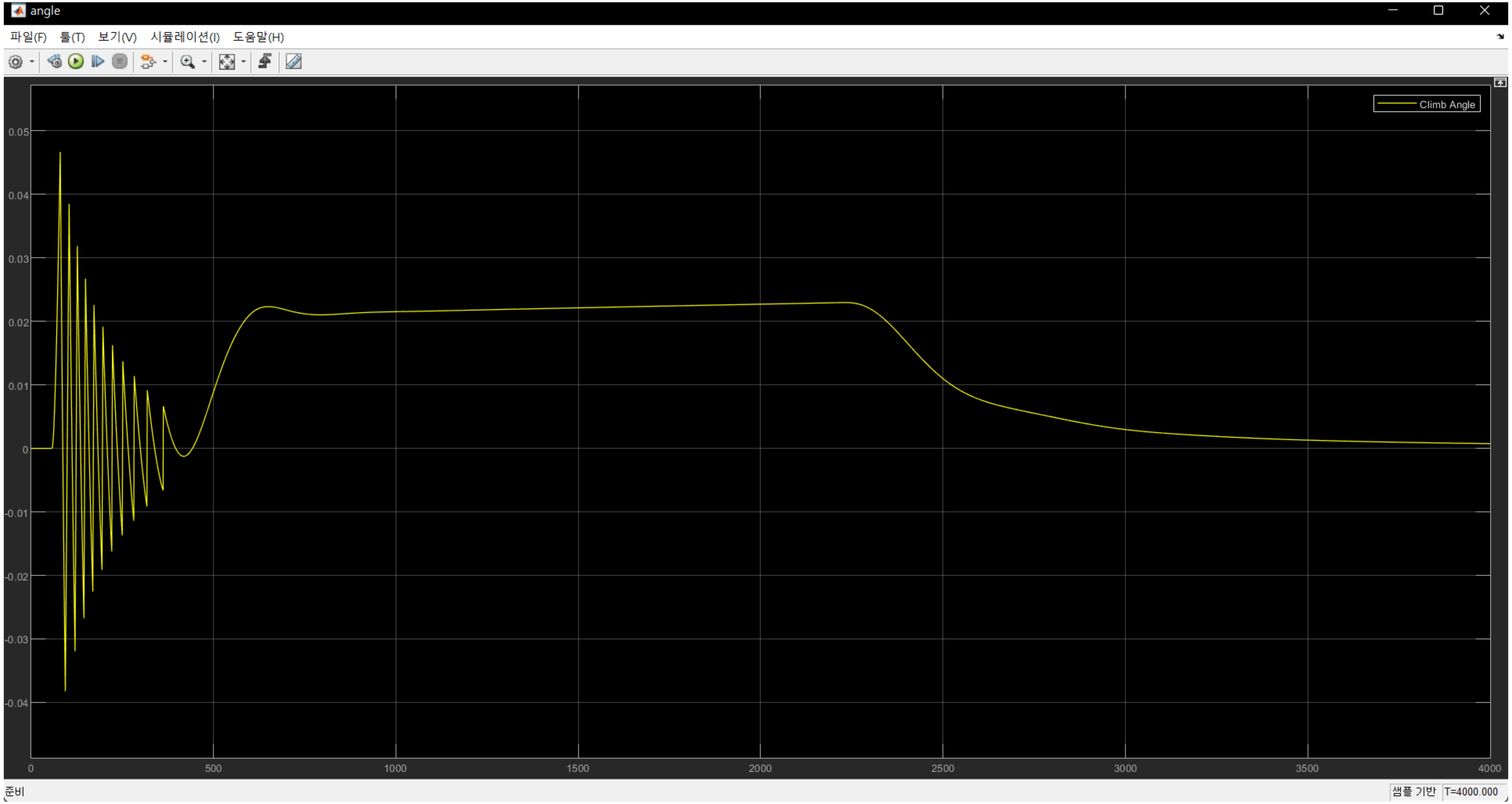


CAE Project

Velocity



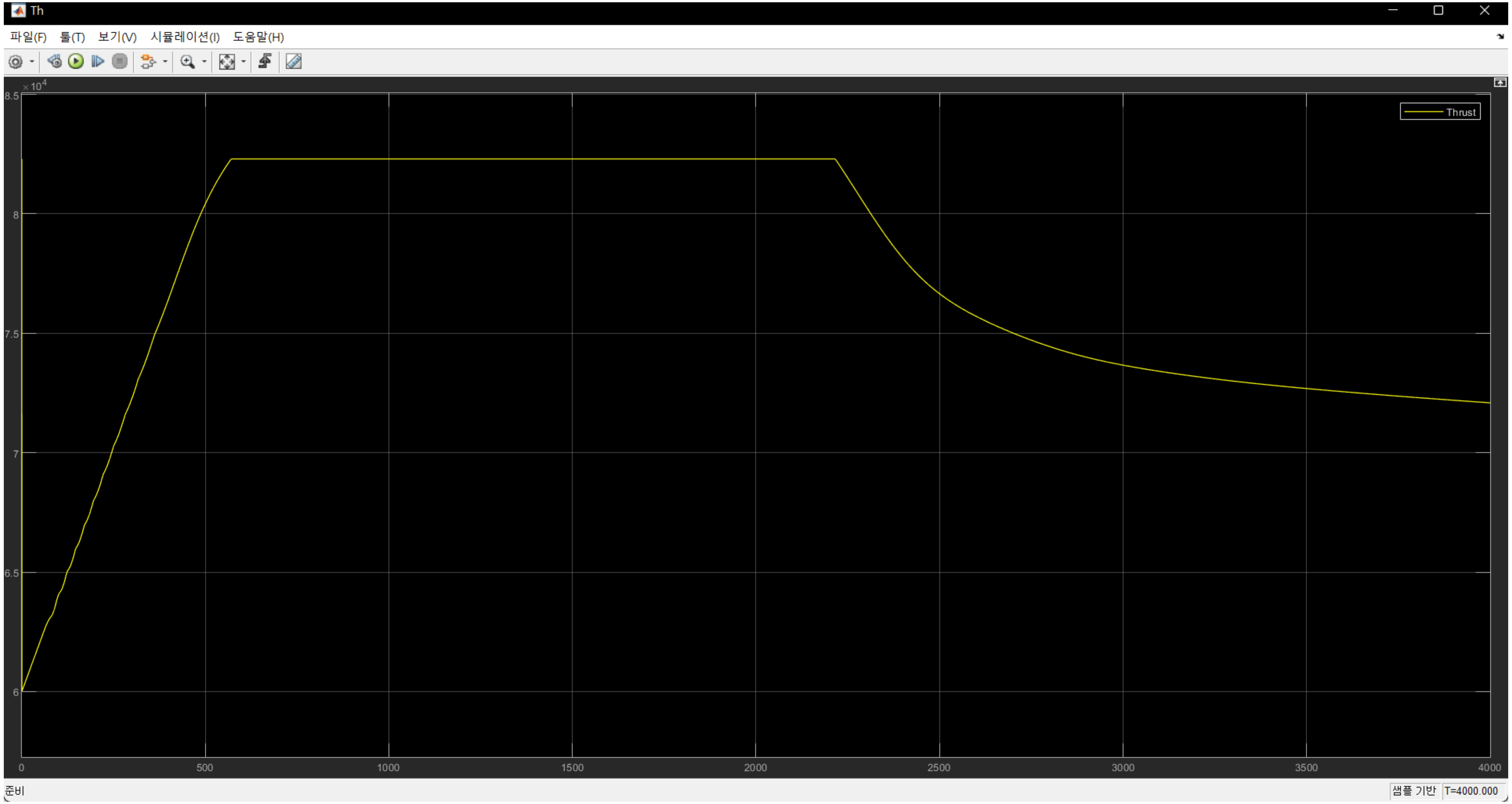
Climb Angle



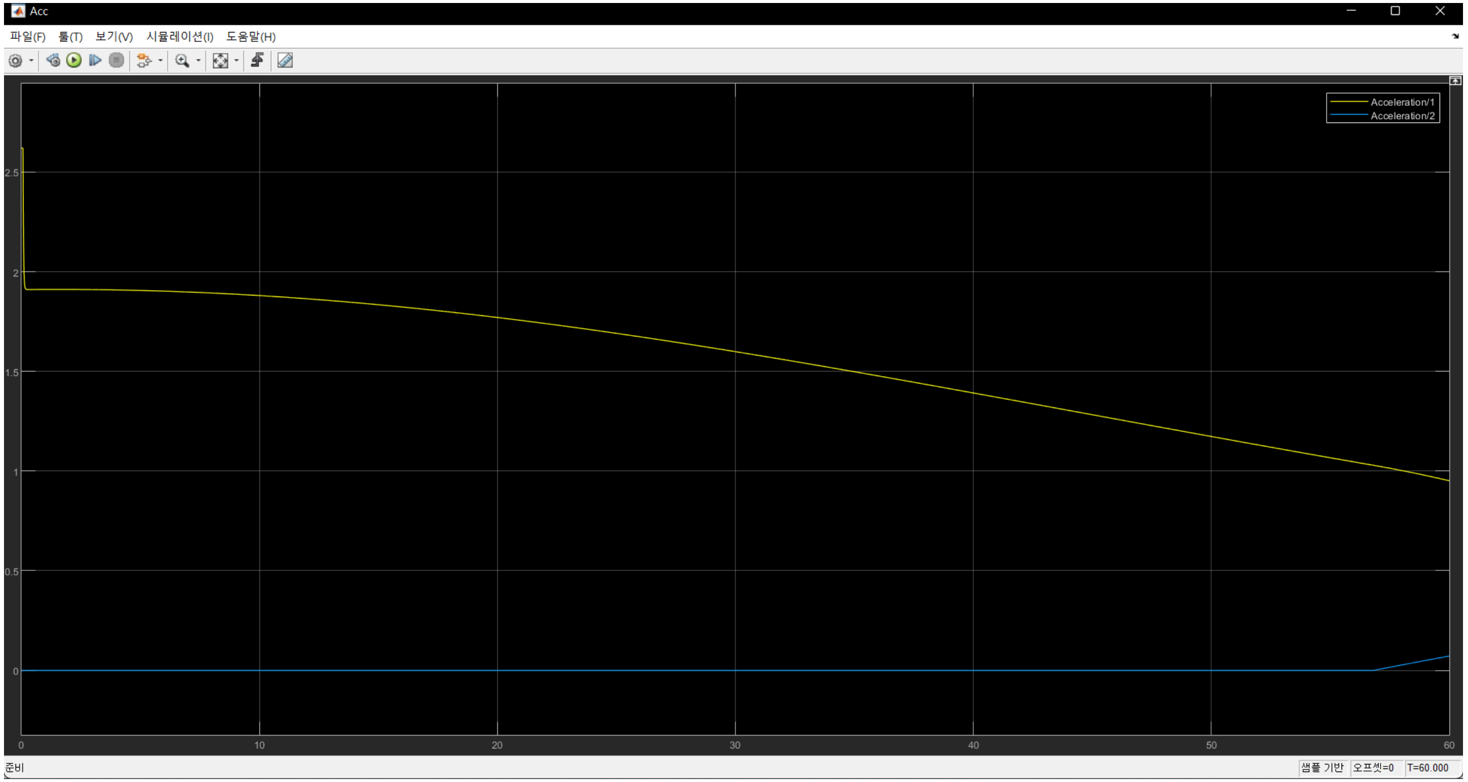
angle
파일(F) 툴(T) 보기(V) 시뮬레이션(I) 도움말(H)
Climb Angle
0.05
0.04
0.03
0.02
0.01
0
-0.01
-0.02
-0.03
-0.04
0
500
1000
1500
2000
2500
3000
3500
4000
준비
샘플 기반
T=4000.000

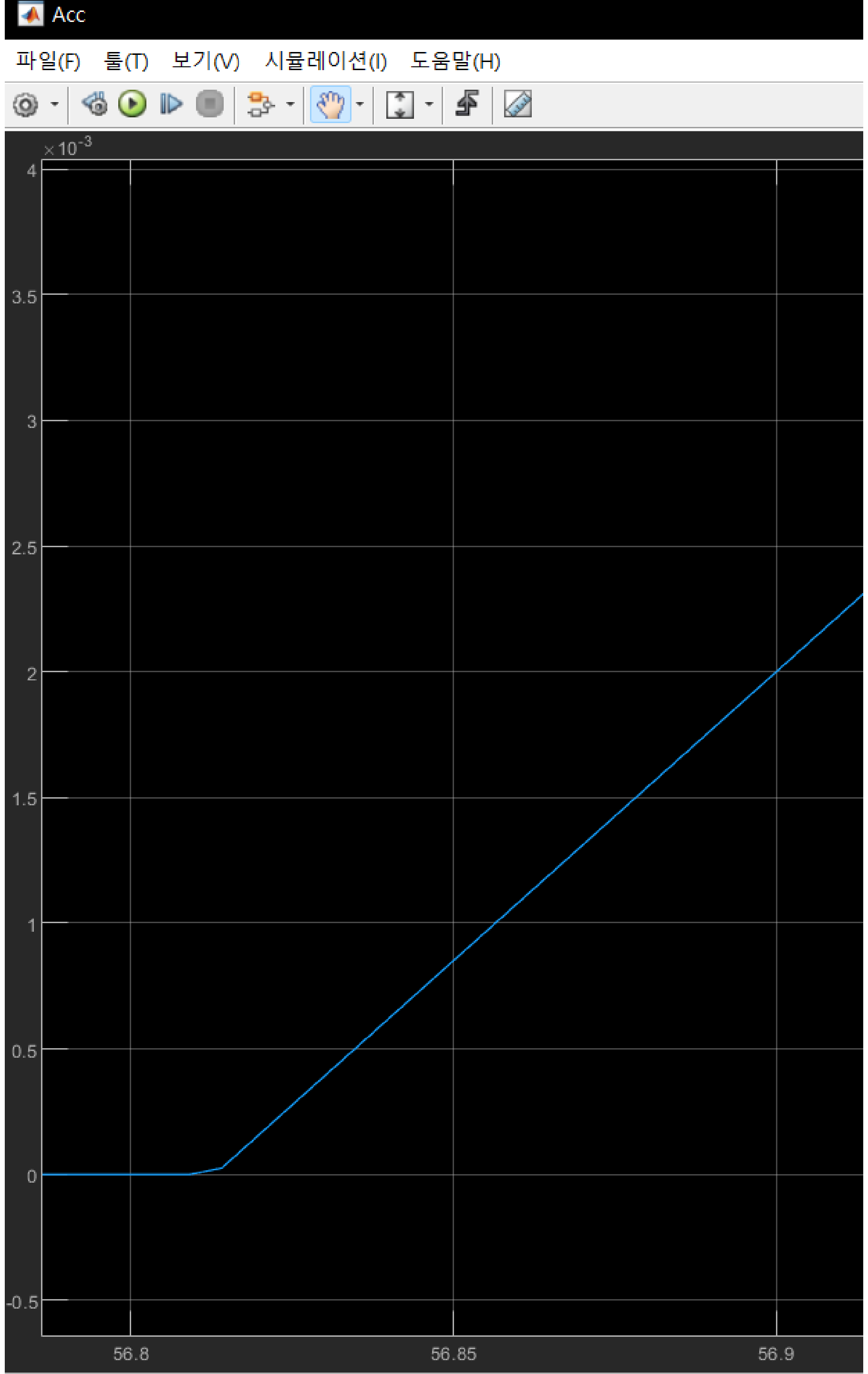
CAE Project

Thrust



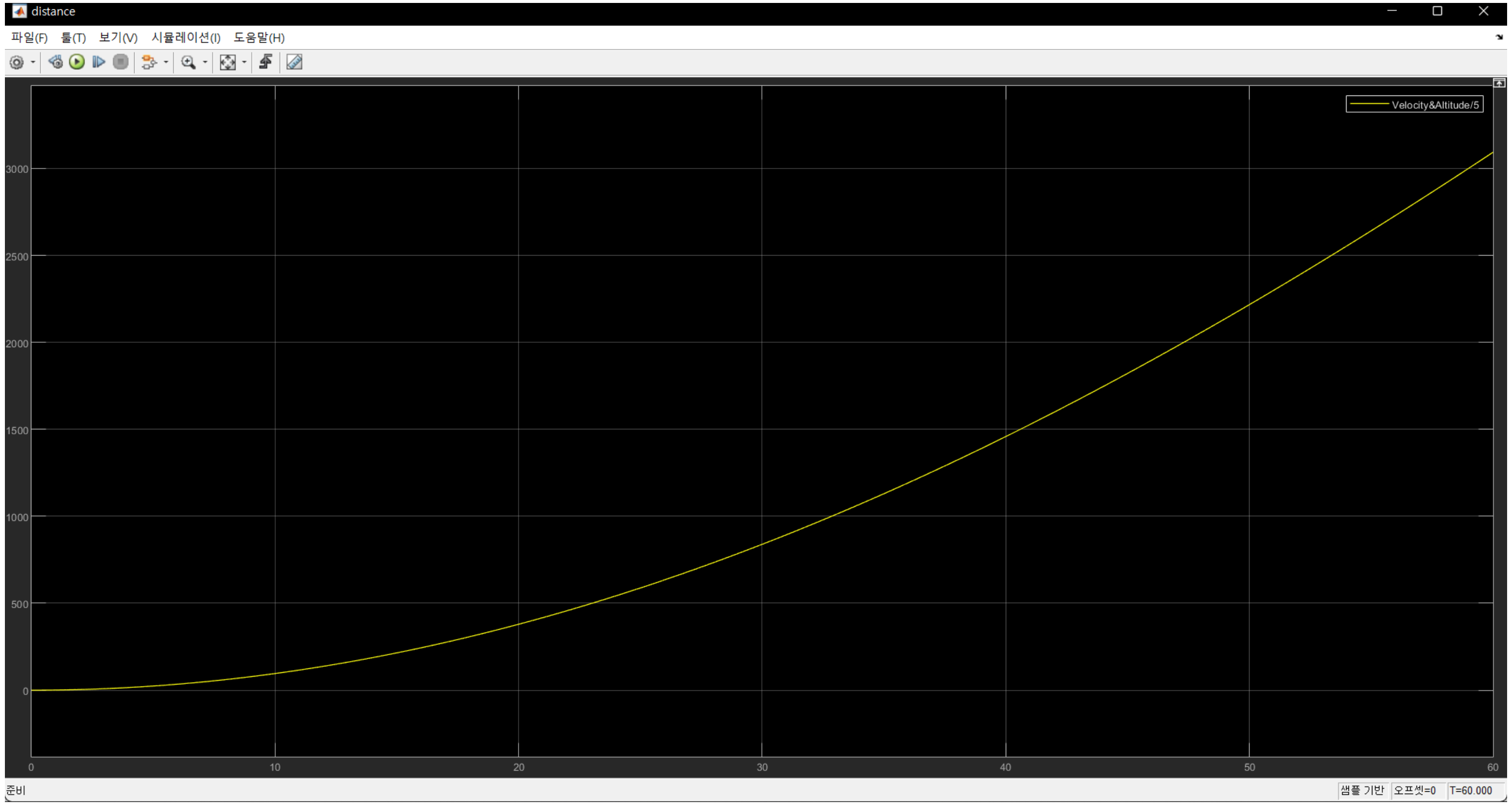
Take-off Time

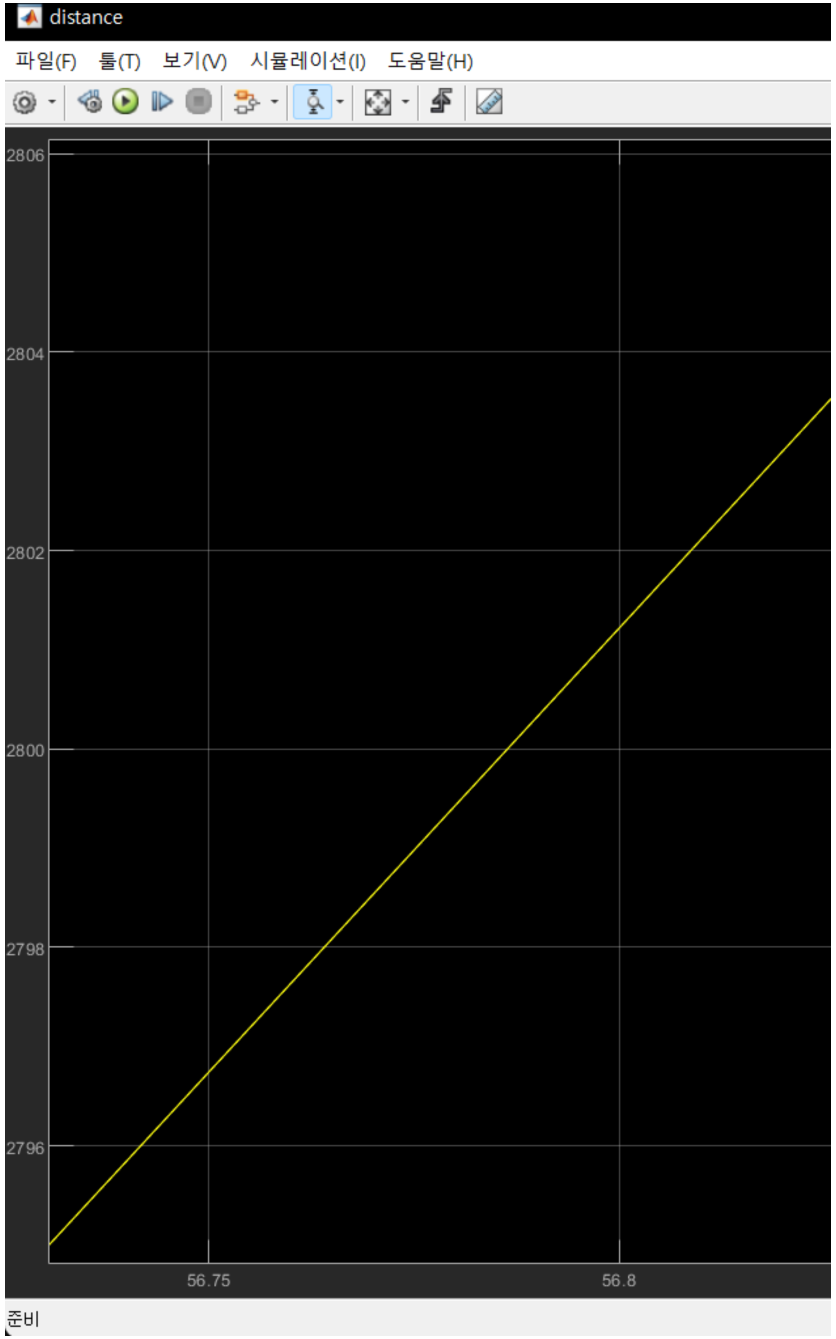




About 56.8 sec

Take-off distance



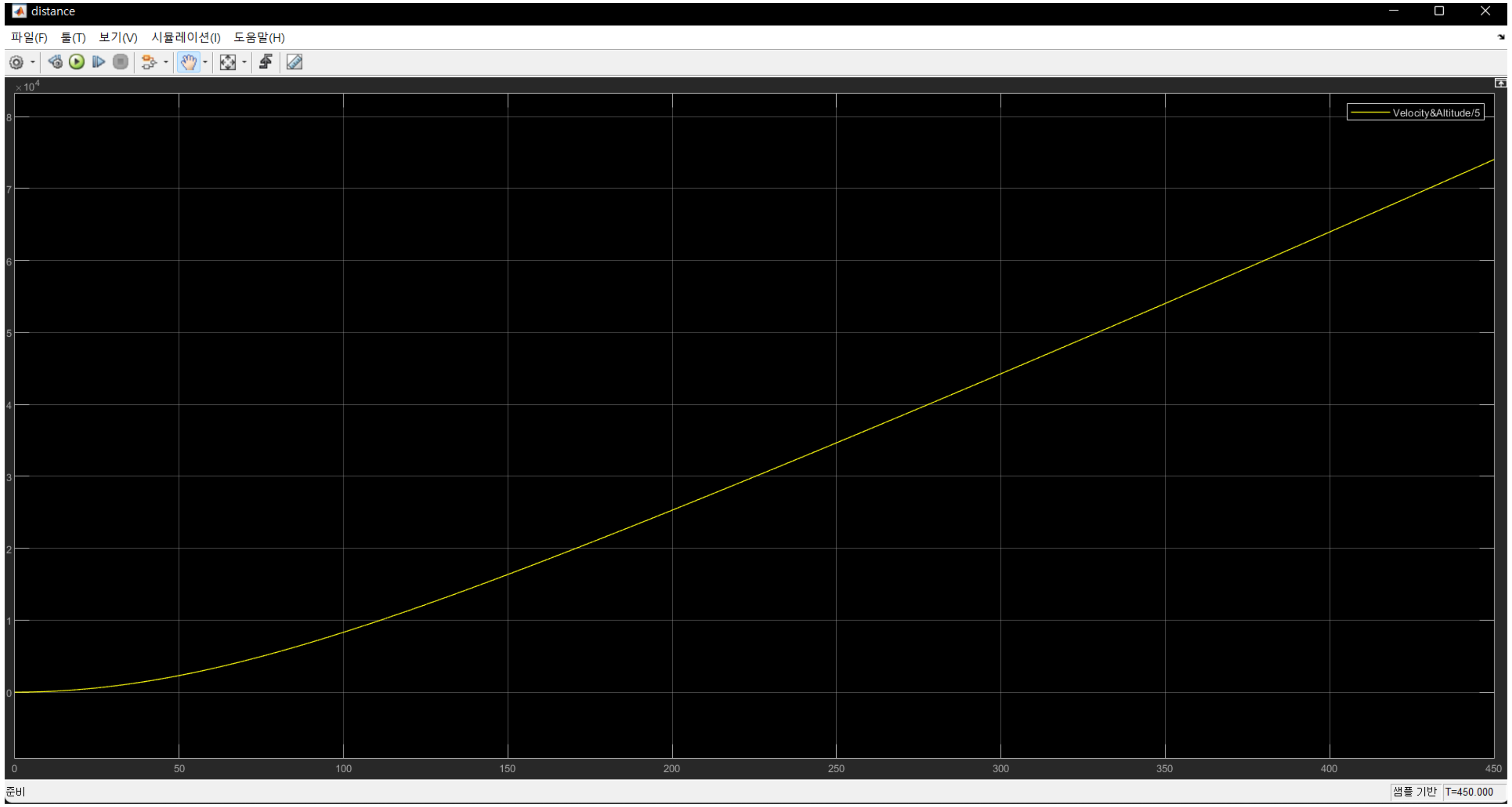


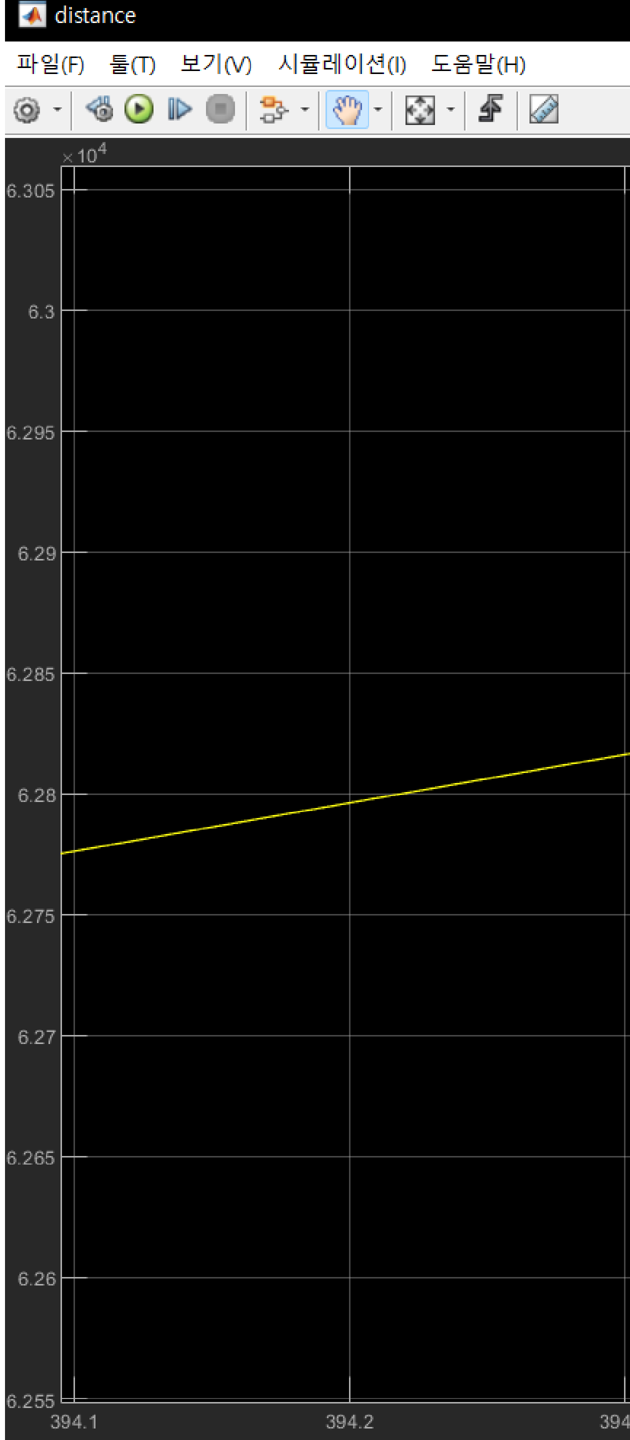
About 2801m

Comparison between “With slat&flap” and “No slat&Flap”

CAE Project

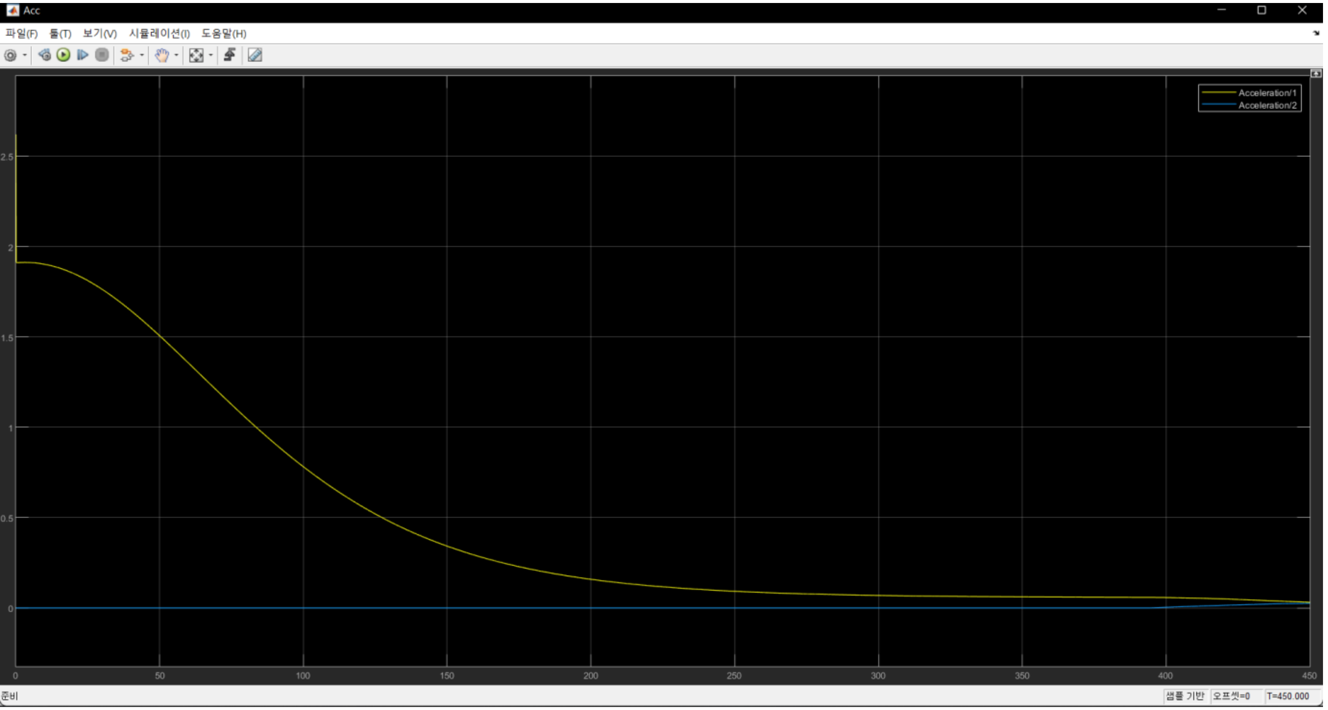
Take-off distance

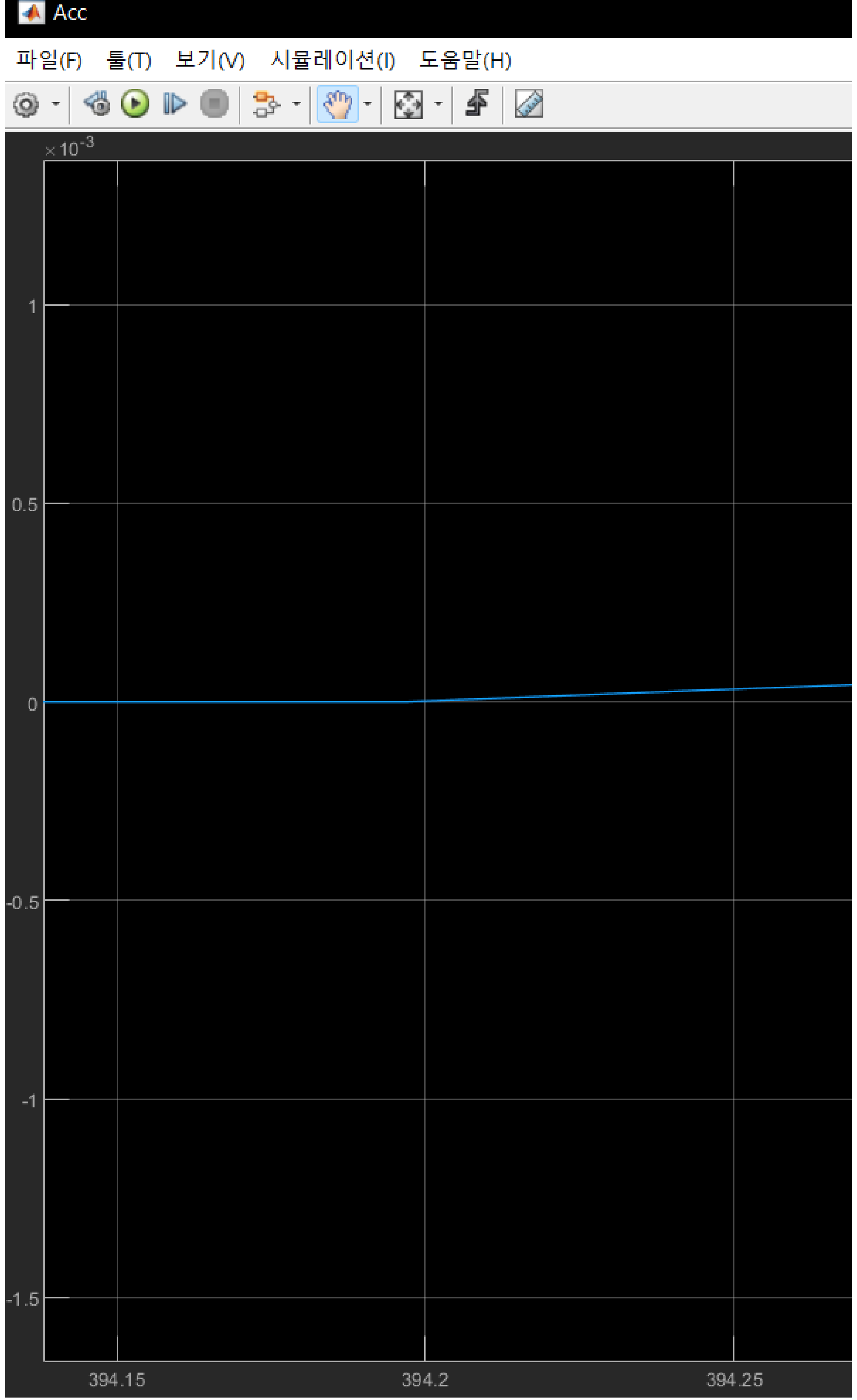




About 62800m >>>>>>>2801m

Take-off Time

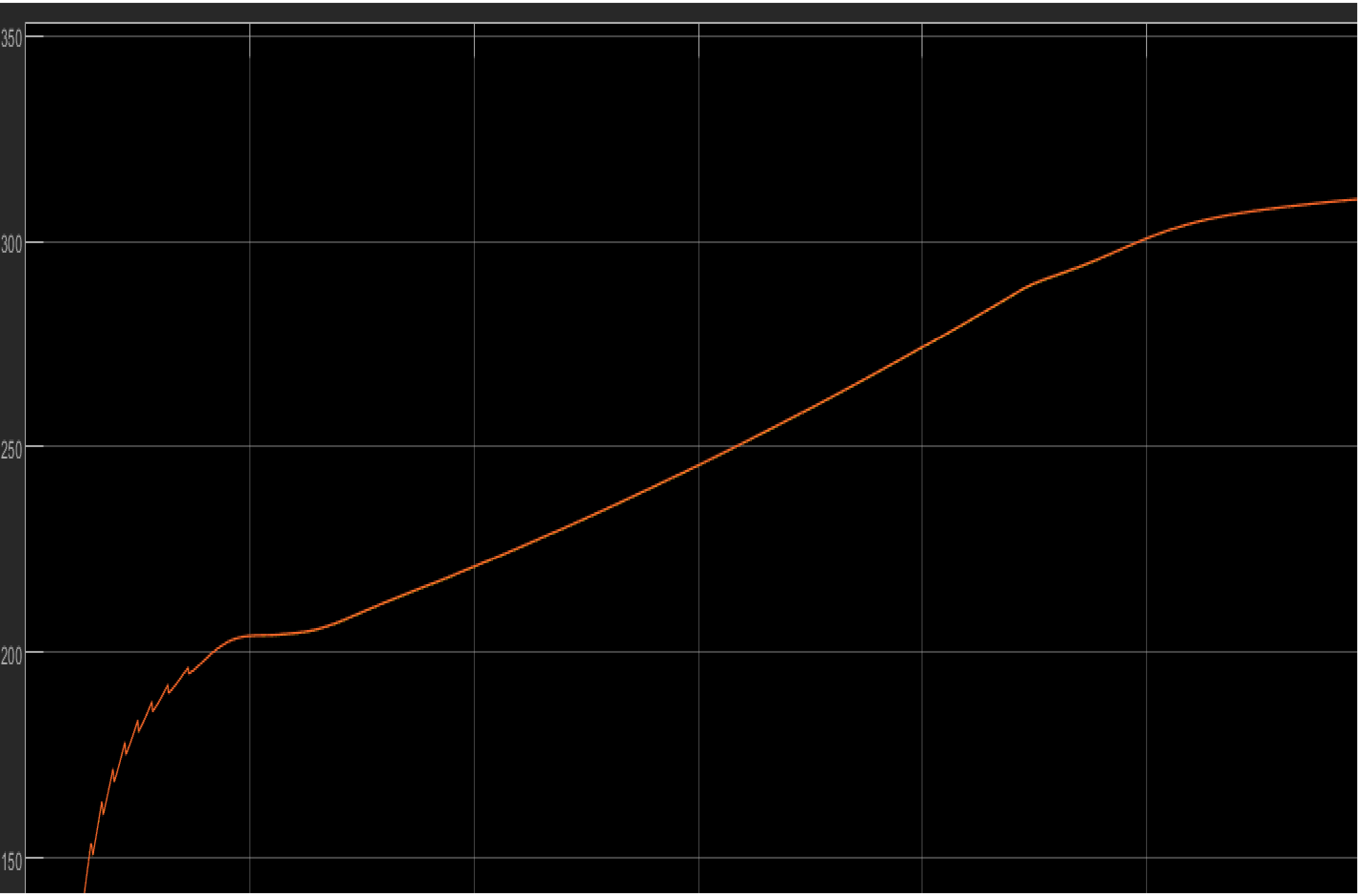




About 394.2 sec>>>>>>>>56.8 sec

Improvements

Climb angle control



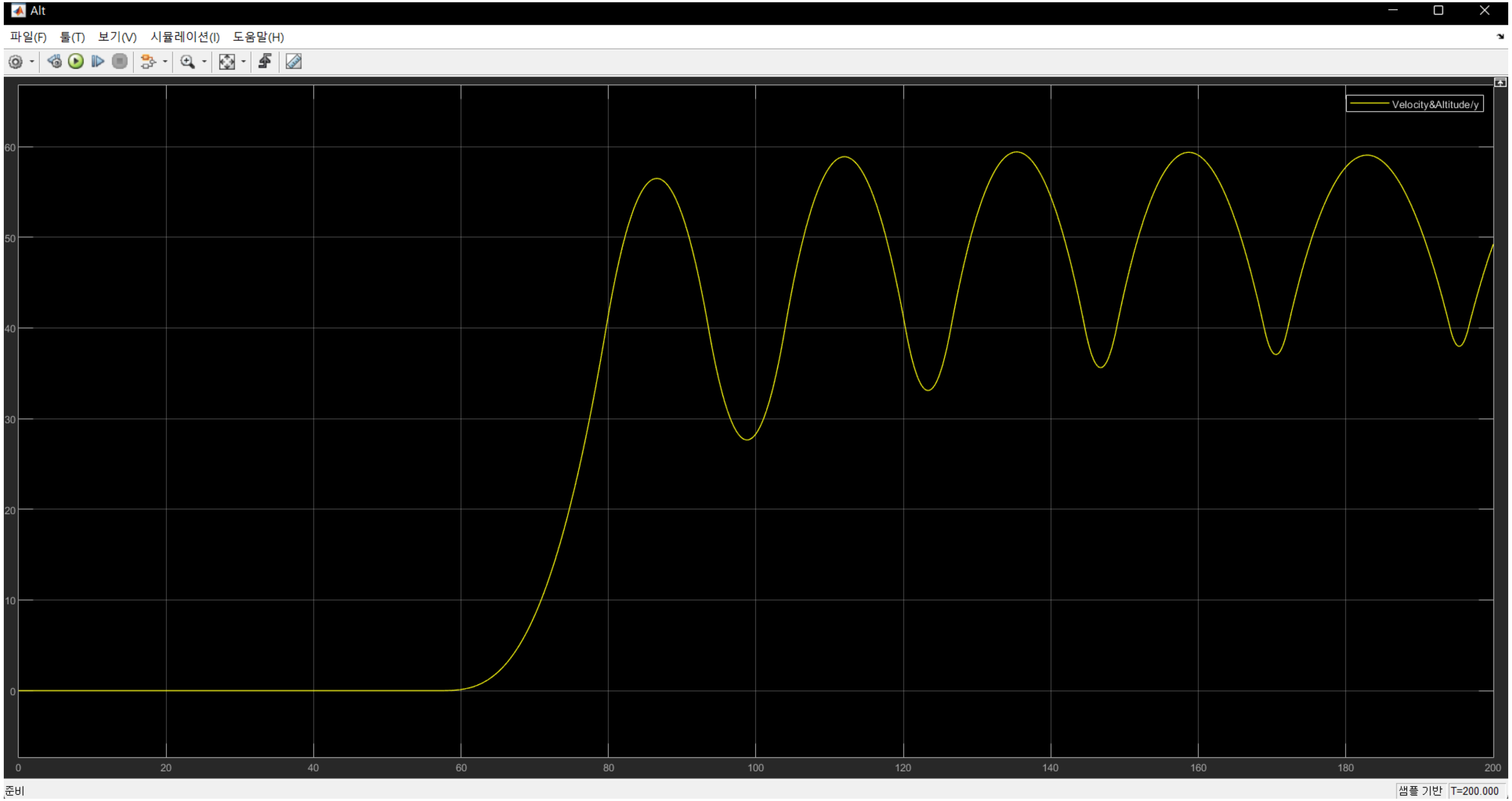
다음 그래프는 시간에 따른 속도 그래프인데, 속도가 300m/s를 넘는 것을 볼 수 있다. 비행기의 순항 속도가 보통 216m/s인 것을 감안하면 매우 빠른 속도이다.

원인은 비행기의 각도 제어가 들어가 있지 않아 오로지 추력으로만 고도를 조절하기 때문에다. 비행기의 각도 제어를 한다면 원하는 속도로 고도를 유지하는 것이 가능할 것이다. 다만 수많은 AoA에 대한 Cl&Cd 값이 필요할 것이다.

CAE Project

Various slat and flap angles

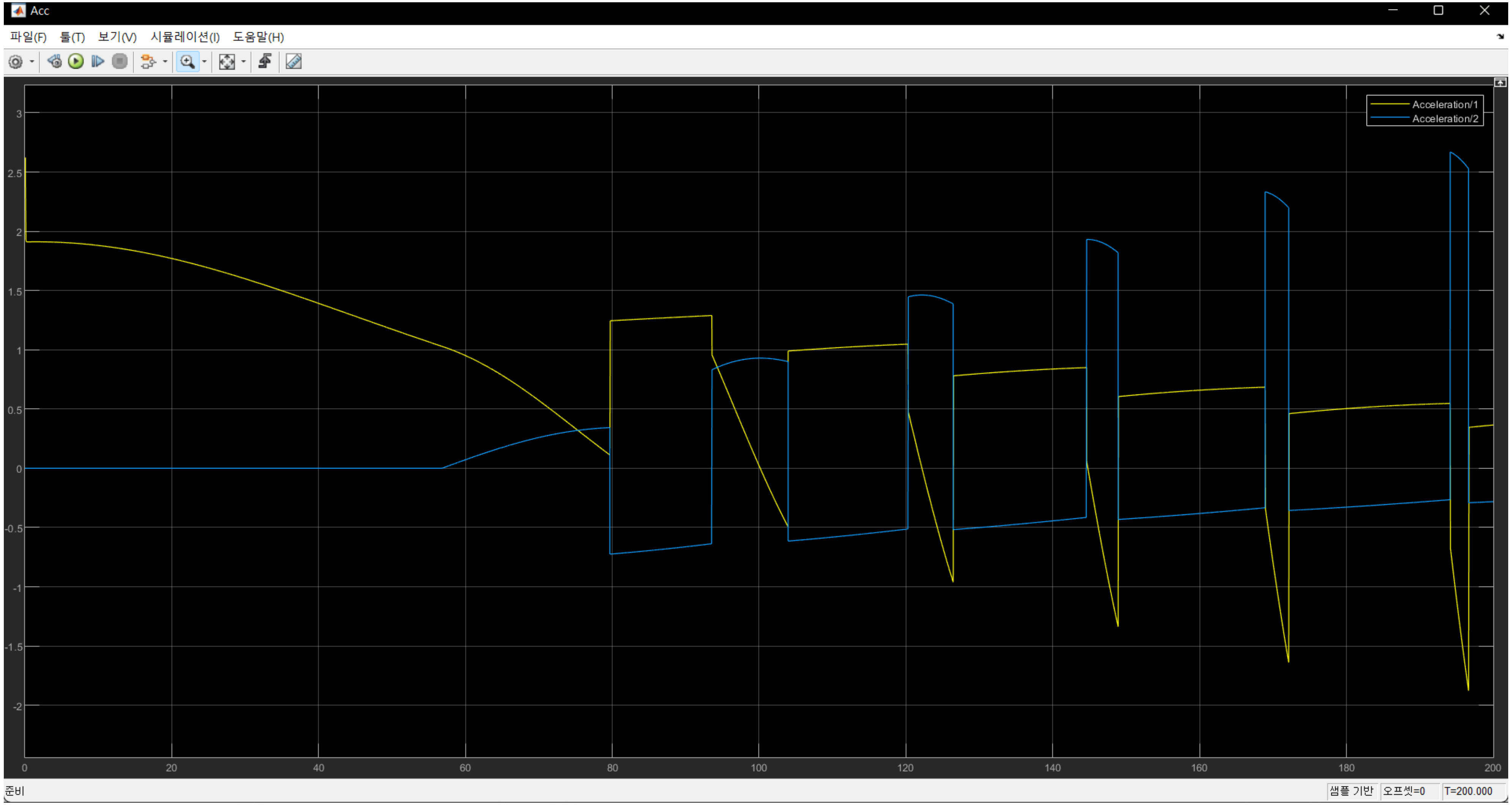
Altitude



CAE Project

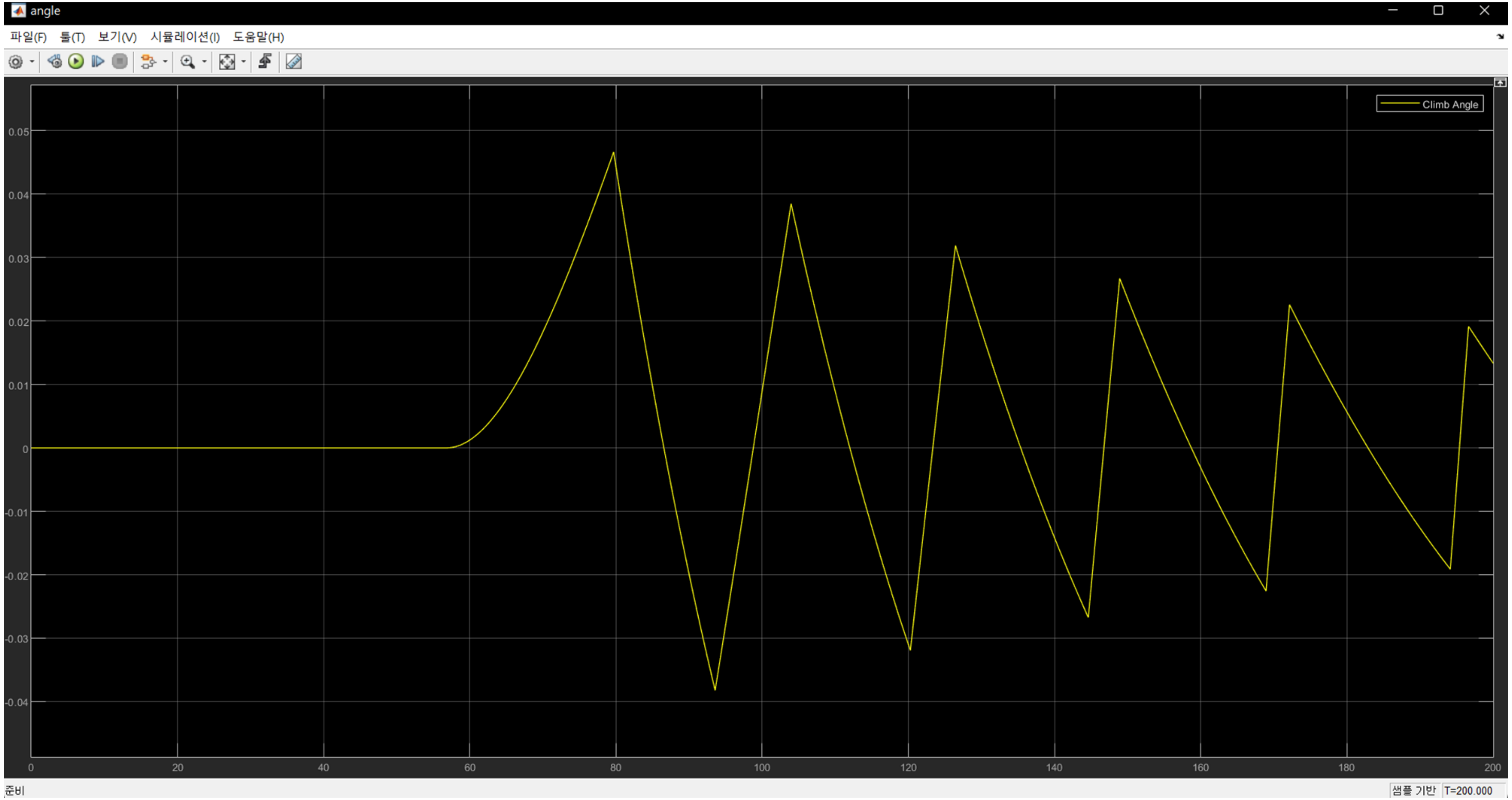
Various slat and flap angles

Accerlation



Various slat and flap angles

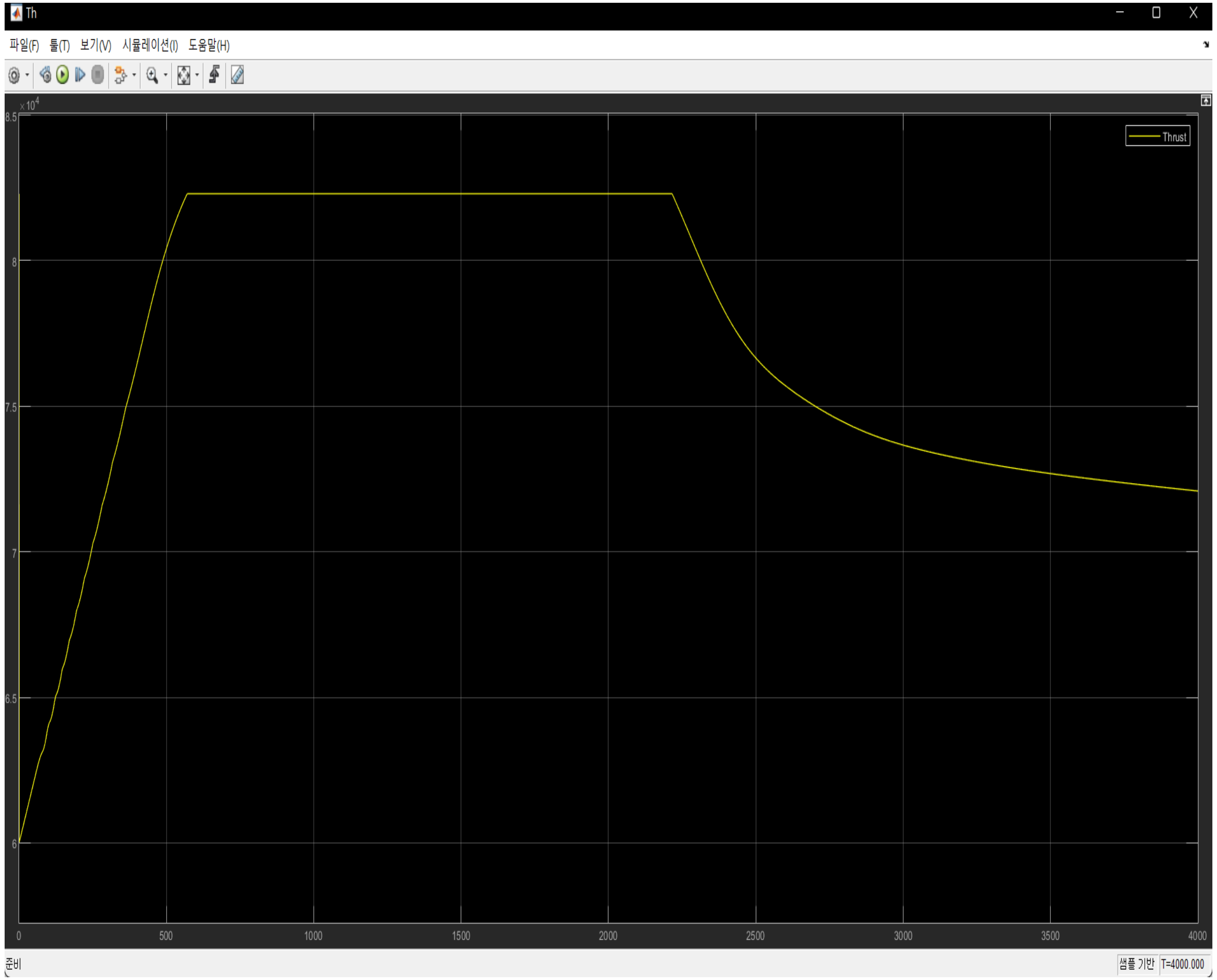
Climb angle



Various slat and flap angles

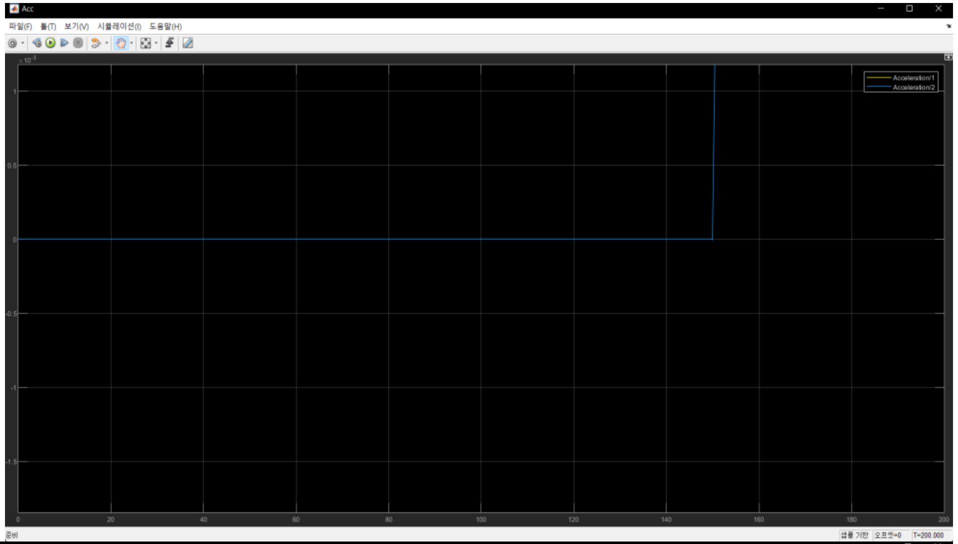
Slat&flap이 접힐 때 갑자기 접히기 때문에 Lift와 Drag의 값이 급격하게 변화하고 이에 따라 심한 진동이 발생한다. 이는 세분화된 slat&flap 별 Lift와 Drag의 값을 알 수 있다면 해소될 수 있는 문제라고 생각한다.

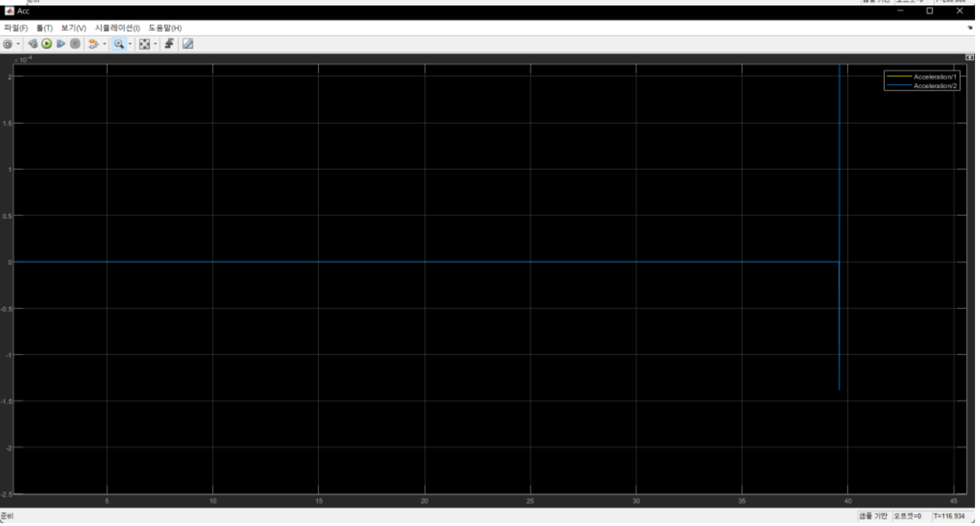
More accurate control system



초반 최대 추력으로 빠르게 속도를 올린 후 이륙해야하지만 PID 제어의 한계로 초반에 추력을 최대로 쓰지 않아 이륙 시간과 거리가 증가하게 되었다.

More accurate control system

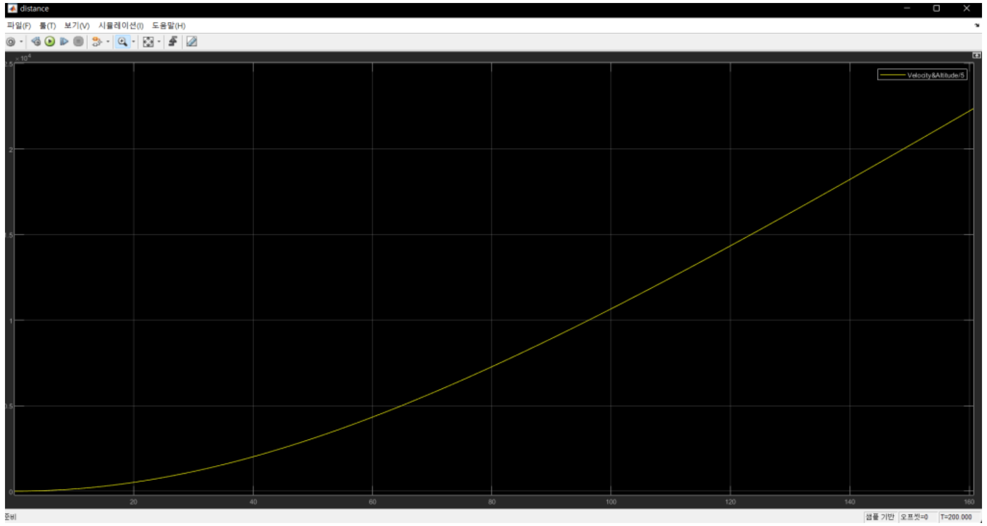


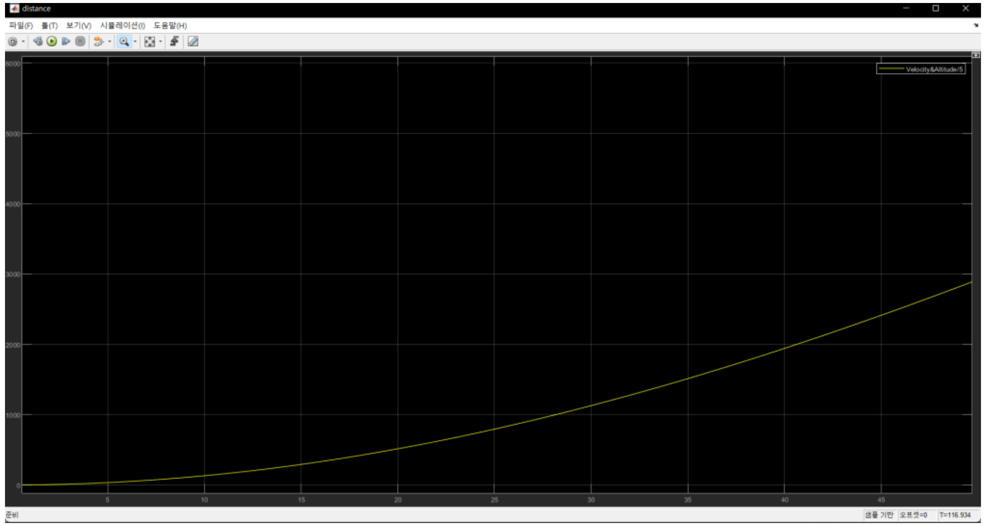


실재로 추력을 82.3kN인 상수로 대입한 경우

Slat&flap이 접힌 상태는 150초에서,
펼쳐진 상태에선 39.6초에 이룩했으며

More accurate control system





실재로 추력을 82.3kN인 상수로 대입한 경우

Slat&flap이 접힌 상태는 약 20000m에서,
펼쳐진 상태에선 1700m만에 이룩했다.

References

References

https://m-selig.ae.illinois.edu/ads/aircraft.html#conventional
-Airfoil-information

http://www.b737.org.uk/techspecsdetailed.htm
-Boeing 737-500 Detailed Technical Data

https://www.boeing.com/resources/boeingdotcom/company/about_bca/startup/pdf/historical/737-classic-passenger.pdf
-Boeing 737-500 official brochure