Simulating a 3-D Vertical Axis Wind Turbine using Actuator Method

Bachelor of Technology, Mechanical Engineering

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OVERVIEW

An unmanned aerial vehicle (UAV), commonly known as a drone, is an aircraft without any human pilot, crew, or passengers on board. UAVs are a component of an unmanned aircraft system (UAS), which has three basic components:

1. An autonomous or human-operated control system which is usually on the ground or a ship but may be on another airborne platform.

2. An Unmanned Aerial Vehicle (UAV)

3. A command and control (C2) system - sometimes referred to as a communication, command, and control (C3) system - to link the two.

These systems include but are not limited to remotely piloted air systems (RPAS) in which the UAV is controlled by a 'pilot' using a radio data link from a remote location. UAS can also include an autonomously controlled UAV or, more likely, a semi-autonomous UAV.

NOTE: In recent years, the tendency to refer to any UAV as a "drone" has developed but the term is not universally considered appropriate. UAVs can vary in size from those which can be hand launched to purpose-built or adapted vehicles the size of conventional fixed or rotary wing aircraft.

In this given task, we have designed and developed two individual UAS systems (Fixed-Wing sUAS and multi-rotor drone) with respect to the given configuration. The design and analysis of the UAS system are done with respect to the selection of parameters that suit the given configuration. A linearized dynamics model was developed and subjected to multiple simulations and test conditions in order to estimate the stability of the system. The control system is implemented by incorporating PID control over the control points of the aircraft and the test results are established. The software platform used for the analysis of the aircraft system is MATLAB and Simulink.

Keywords: sUAS, Multirotor, Control System, Fixed-wing

Chapter 1

Fixed Wind sUAS Development

1.1 Range & Endurance

The fixed wing sUAS we used for this project was the Aerosonde UAV. The parameters as given for the Aerosonde UAV are in the Figure below.

1.1.1 Range

The range of an aircraft is technically defined as the total distance measured with respect to the ground traversed by the aircraft on one full tank of fuel. It is used to estimate engine performance in terms of fuel consumption. The maximum total range is the maximum distance an aircraft can fly between takeoff and landing, as limited by fuel capacity in powered aircraft, or cross-country speed and environmental conditions in unpowered aircraft. In this project, the maximum range of the given aircraft is calculated with respect to the propeller data obtained from the UIUC database for the given propeller (APC 16X8).

In order to calculate the range of the aircraft with respect to the different powertrain systems, the following specifications are adopted:

1. **Specific Fuel Consumption (SFC) (for propeller)** : Specific fuel consumption is one of the important metrics in determining the performance of a propeller-driven aircraft. SFC is defined as the weight of the fuel consumed by the reciprocating engine per unit power per unit time.

2. **Thrust specific fuel consumption (TSFC) (for jet engine)** : The thrust specific fuel consumption of a jet engine is defined as the fuel efficiency of an engine design with respect to thrust output. TSFC is the mass of fuel burned by an engine in one hour divided by the thrust that the engine produces.

Geometric		Longitudinal		Lateral	
Parameter	Value	Coef.	Value	Coef.	Value
m	13.5 kg	C_{L_0}	0.28	C_{Y_0}	0
I_{xx}	0.8244 kg m^2	C_{D_0}	0.03	C_{l_0}	0
I_{yy}	1.135 kg m^2	C_{m_0}	-0.02338	C_{n_0}	0
I_{zz}	1.759 kg m^2	C_{L_α}	3.45	C_{Y_β}	-0.98
I_{xz}	0.1204 kg m^2	C_{D_α}	0.30	C_{l_β}	-0.12
S	0.55 m^2	C_{m_α}	-0.38	C_{n_β}	0.25
b	2.8956 m	C_{L_q}	0	C_{Y_p}	0
c	0.18994 m	C_{D_q}	0	C_{l_p}	-0.26
S_{prop}	0.2027 m^2	C_{m_q}	-3.6	C_{n_p}	0.022
e	0.9	$C_{L_{\delta_e}}$	-0.36	C_{Y_r}	0
C_T	$0.7155 - 0.3927J^2$	$C_{D_{\delta_e}}$	0	C_{l_r}	0.14
C_Q	$0.0056 - 0.0052J$	$C_{m_{\delta_e}}$	-0.5	C_{n_r}	-0.35
Ω_{max}	7000 RPM	ϵ	0.1592	$C_{Y_{\delta_a}}$	0
Fuel Capacity	5.7 L			$C_{l_{\delta_a}}$	0.08
				$C_{n_{\delta_a}}$	0.06
				$C_{Y_{\delta_r}}$	-0.17
				$C_{l_{\delta_r}}$	0.105
				$C_{n_{\delta_r}}$	-0.032

Figure 1.1: Given Parameters

The maximum range for the APC 16X8 propeller is calculated using the given formula:

$$R = \int_{W_0}^{W_1} \frac{\eta}{c_p} \frac{C_L}{C_D} \frac{dW}{W} = \frac{\eta}{c_p} \frac{C_L}{C_D} ln(\frac{W_0}{W_1}) \tag{1.1}$$

$$R_{max} = \frac{\eta}{c_p} \frac{C_L}{C_D} ln(\frac{W_0}{W_1}) = \frac{\eta/c_p}{2\sqrt{KC_{D_0}}} ln(\frac{W_0}{W_1}) \tag{1.2}$$

Note : To cover the longest distance, common sense says that we must use the minimum fuel consumption per unit distance (e.g., km or mile).

1.1.2 Endurance

The endurance of an aircraft is defined as the total time that an aircraft stays in the air on a tank of fuel. Like the range characteristics, to achieve maximum endurance, it is advised to use the minimum thrust per unit time. For a steady level flight, the lift is equal to weight and the thrust is equal to the drag (L = W, T = D).

To calculate the maximum endurance for a propeller-driven aircraft, the following parameters are required :

1. Maximum weight loss (Wf = W0 – W1)
2. Maximum $C_l^{\frac{3}{2}}/C_d$

The maximum endurance for a propeller-driven aircraft is calculated using the given formula :

$$E = \int_{W_0}^{W_1} \frac{\eta}{c_p V} \frac{C_L}{C_D} \frac{dW}{W} \tag{1.3}$$

$$= \int_{W_0}^{W_1} \frac{\eta}{c_p} \sqrt{\frac{\rho S}{2}} \frac{C_L^{1.5}}{C_D} \frac{dW}{W^{1.5}} \tag{1.4}$$

$$E = \frac{\eta}{c_p} \sqrt{2\rho S} (\frac{C_L^{1.5}}{C_D})(\frac{1}{\sqrt{W_1}} - \frac{1}{\sqrt{W_0}}) \tag{1.5}$$

Based on the above-mentioned procedures, the results of the multi-rotor drone (quadcopter) are as follows:

Max. Range = 390.04 kms

Max. Endurance = 5.88 hours

1.2 Fixed Wing Aircraft Dynamics

We are using the Linearized Models in the **?** for Developing the Flight Dynamics for the Fixed Wing Aircraft. The Aircraft Dynamics is primarily divided into two different parts, The Longitudinal Dynamics and the Lateral Dynamics. The Linearized Equations are demonstrated in the State Space Models as given in the further subsections. The Longitudinal Dynamics has the elevator and the thrust as the inputs. The elevator controls the pitching and the thrust controls the Speed. The Lateral Dynamics has the aileron and the rudder as the inputs.

1.2.1 Longitudinal Flight Dynamics

The Longitudinal Dynamics has 6 states, u, w, q, θ, h. The Longitudinal Statespace can be seen as below. The Longitudinal The Dynamics Equations for all these states are linearized using Taylor's sequence at the Equilibrium Locations. We calculate the Equilibrium values using the non-linear Dynamic Equations.

Given a nonlinear system described by the differential equations

$$\dot{x} = f(x, u),$$

where $f : \mathbb{R}^n \times \mathbb{R}^m \to \mathbb{R}^n$, x is the state of the system, and u is the input, the system is said to be in equilibrium at the state x^* and input u^* if

$$f(x^*, u^*) = 0.$$

When an Aircraft is in constant-altitude, wings-level steady flight, a subset of its states are in equilibrium. In particular, the altitude h; the body frame velocities u, v, w; the Euler angles ϕ, θ, ψ; and the angular rates p, q, and r are all constant, Zero in this case.

The final state-Space Equation is as given below

$$\begin{pmatrix} \dot{\bar{u}} \\ \dot{\bar{w}} \\ \dot{\bar{q}} \\ \dot{\bar{\theta}} \\ \dot{\bar{h}} \end{pmatrix} = \begin{pmatrix} X_u & X_w & X_q & -g\cos\theta^* & 0 \\ Z_u & Z_w & Z_q & -g\sin\theta^* & 0 \\ M_u & M_w & M_q & 0 & 0 \\ 0 & 0 & 1 & 0 & 0 \\ \sin\theta^* - \cos\theta^* & 0 & u^*\cos\theta^* + w^*\sin\theta^* 0 & & \end{pmatrix} \begin{pmatrix} \bar{u} \\ \bar{w} \\ \bar{q} \\ \bar{\theta} \\ \bar{h} \end{pmatrix}$$
$$+ \begin{pmatrix} X_{\delta_e} & X_{\delta_t} \\ Z_{\delta_e} & 0 \\ M_{\delta_e} & 0 \\ 0 & 0 \\ 0 & 0 \end{pmatrix} \begin{pmatrix} \bar{\delta}_e \\ \bar{\delta}_t \end{pmatrix}$$

The Coefficients of the Matrix are as given below:

```
Cxo = -P.CDo*cos(alpha) + P.CLo*sin(alpha)
Cxa = -P.CDa*cos(alpha) + P.CLa*sin(alpha)
Cxq = -P.CDq*cos(alpha) + P.CLq*sin(alpha)
Cxdelta_e = -P.CDdelta_e*cos(alpha) + P.CLdelta_e*sin(alpha)

Xu  = (2*qs*Cxo - rho*Sprop*Ue*C_T)/m
Xw  = Cxa*qs/m
Xq  = Cxq*qs1*c_bar/(2*m)
Xdelta_e = Cxdelta_e*qs1_2/m
Xdelta_t = (rho*Sprop*C_T*k^2)/m

Czo = -P.CLo*cos(alpha) - P.CDo*sin(alpha)
Cza = -P.CLa*cos(alpha) - P.CDa*sin(alpha)
Czq = -P.CLq*cos(alpha) - P.CDq*sin(alpha)
Czdelta_e = -P.CLdelta_e*cos(alpha) - P.CDdelta_e*sin(alpha)
```

```
Zu    = 2*qs*Czo/m
Zw    = Cza*qs/m
Zq    = Ue + Czq*qs1*c_bar/(2*m)
Zdelta_e = Czdelta_e*qs1_2/m

Mu    = 2*qs*c_bar*Cmo/Iyy
Mw    = Cma * qs * c_bar/Iyy
Mq    = Cmq*qs1*c_bar*c_bar/(2*Iyy)
Mdelta_e = Cmdelta_e*qs1_2*c_bar/Iyy
```

1.2.2 Lateral Flight Dynamics

For the lateral dynamics, the variables of interest are the roll angle ϕ, the roll rate p, the heading angle ψ, and the yaw rate r. The control surfaces used to influence the lateral dynamics are the ailerons δ_a, and the rudder δ_r . The ailerons are primarily used to influence the roll rate p, while the rudder is primarily used to control the yaw ψ of the aircraft.

$$\begin{pmatrix} \dot{\bar{v}} \\ \dot{p} \\ \dot{\bar{r}} \\ \dot{\bar{\phi}} \\ \dot{\bar{\psi}} \end{pmatrix} = \begin{pmatrix} Y_v & Y_p & Y_r & g\cos\theta^*\cos\phi^* & 0 \\ L_v & L_p & L_r & 0 & 0 \\ N_v & N_p & N_r & 0 & 0 \\ 0 & 1 & \cos\phi^*\tan\theta^* & q^*\cos\phi^*\tan\theta^* - r^*\sin\phi^*\tan\theta^* & 0 \\ 0 & 0 & \cos\phi^*\sec\theta^* & p^*\cos\phi^*\sec\theta^* - r^*\sin\phi^*\sec\theta^* & 0 \end{pmatrix}$$
$$\times \begin{pmatrix} \bar{v} \\ \bar{p} \\ \bar{r} \\ \bar{\phi} \\ \bar{\psi} \end{pmatrix} + \begin{pmatrix} Y_{\delta_a} & Y_{\delta_r} \\ L_{\delta_a} & L_{\delta_r} \\ N_{\delta_a} & N_{\delta_r} \\ 0 & 0 \\ 0 & 0 \end{pmatrix} \begin{pmatrix} \bar{\delta}_a \\ \bar{\delta}_r \end{pmatrix}$$

The Coefficients of the Matrix are as given below:

```
Yv = 2*Cyo*qs/m + qs*Cyb/m
Yp = Cyp*qs1*b/(2*m)
Yr = -Ue + (Cyr*qs1*b/(2*m))
Ydelta_a = Cydelta_a*qs1_2/m
Ydelta_r = Cydelta_r*qs1_2/m

Lv = Cpb*qs*b
Lp = Cpp*qs1*b^2/2
Lr = Cpr*qs*b^2/2
```

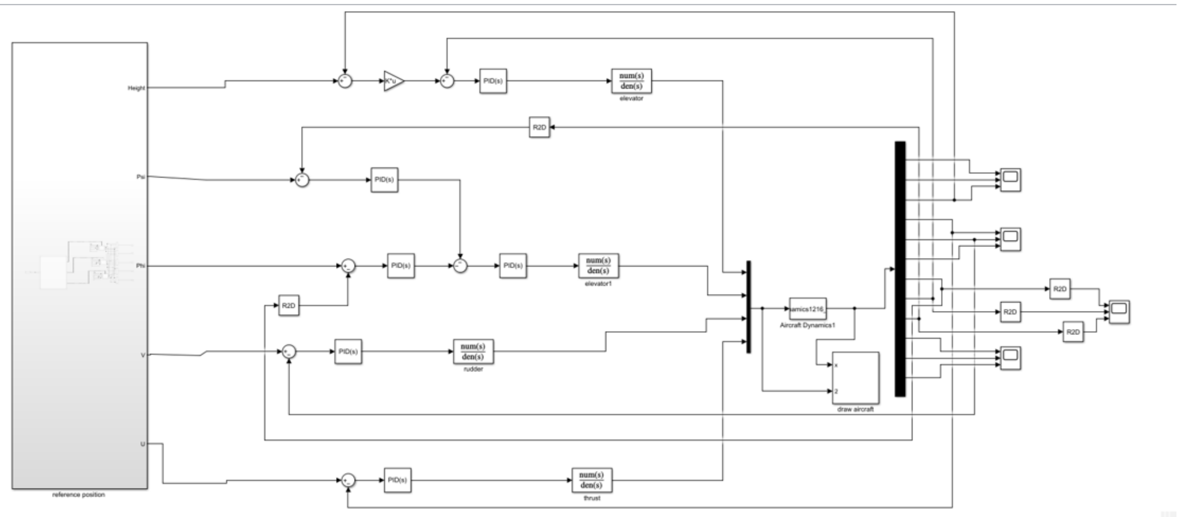

Figure 1.2: Simulink Model

```
Ldelta_a = Cpdelta_a * qs1_2 * b
Ldelta_r = Cpdelta_r * qs1_2 * b

Nv = Crb * qs * b
Np = Crp * qs1 * b^2 / 2
Nr = Crr * qs1 * b^2 / 2
Ndelta_a = Crdelta_a * qs1_2 * b
Ndelta_r = Crdelta_r * qs1_2 * b
```

1.3 Simulink Model

The Given State-Space setup was implemented in Matlab with a Level2 S Function Block. The Block takes 4 inputs in the following order : δ_e, δ_a, δ_r, δ_t. The Block further provides 12 outputs, giving the state for both the longitudinal and lateral Systems. The complete Simulink Model can be seen in the figure 1.2.

The Simulink model performs all the three manuevers in the order provided in the Queation. The input for the system is controlled from the refernece position subsystem, which gives out 5 outputs, height h, ψ, ϕ, V, U. The error are calculated using feedback loops and are further fed to PIDs. A Radian to Degree converter is used in ϕ and ψ as the outputs are in Radians. The reference position subsystem is time-based with a digital clock as an input and time based if-else block. We calculated the time for each manuever and used the digital clock to change the input for the positions.

Chapter 2

Multi-Rotor Development

2.1 Range & Endurance

For a multirotor, range can be estimated by how far it can be controlled by the controller before losing its connection. The range of a multirotor is dependent on certain factors like weight, motor power, weather conditions, etc. Based on the application requirement, they can be customized for longer ranges and better performances. In this project, the forward flight momentum theory was used to estimate the range and endurance of the aircraft for a given set of forward speeds. A velocity range of 0-20 m/s was chosen to perform this calculation and the estimated values of range, endurance, and forward speeds of the multi-rotor drone were calculated using the following steps as follows:

2.1.1 Forward Flight Momentum Theory

The forward flight expression can be written as:

$$v = \frac{T}{2} \tag{2.1}$$

At a given forward speed, V we can then solve for $_D$, v, T, P_{ind}, P_{tot} as follows:

1. Calculate the quadrotor drag $D = \frac{\rho C_D V^2}{2}$ at $_D = 0$ Assume $_D$ does not affect the drag, otherwise need to iterate to find solution
2. Solve $_D = tan^{-1}1(\frac{D}{W})$
3. Square both sides of 2.1 replace T_2 by W_2 + D_2 and re-arrange to get, $v_4 + (2V sin(_D))v_3 + V2v_2(W2 + D2)/(2A)2 = 0$

4. Positive real root of equation gives v, then $P_{ind} = T_v; P_{tot} = T(v + V sin(_D))$

5. Notice that this total power does not include profile drag, swirl, or additional losses due to non-uniform induced velocity.

6. Solve for a range of speeds and plot results versus V

The maximum range of the quadcopter is calculated using the formula:

Max. Range = $(\frac{E_b * m_e * esc_e}{MinimumTotalPower/Velocity})$

Where, E_b = Energy of the battery; M_e = Motor efficiency; Esc_e = ESC efficiency

The maximum endurance of the quadcopter is calculated using the formula:

Max. Endurance = $(E_b * m_e * esc_e)/(MinimumTotalPower)$
Where, E_b = Energy of the battery; M_e = Motor efficiency; Esc_e = ESC efficiency

Based on the above-mentioned procedures, the results of the multi-rotor drone (quadcopter) are as follows:

- Max. Range = 34 kms
- Max. Endurance = 68.1 minutes
- Forward Speed for range = 9.5 m/s
- Forward Speed for endurance = 7 m/s

2.2 Quadrotor Dynamics Model

The development of dynamics model of quadrotor was done by using the following state space matrices given for each control channel.

Given State-Space Matrices :

The output from position controller was fed into the dynamics model. The dynamics model consists of each control channel mentioned above put in a separate state-space block in Simulink. For roll control the input was the desired roll angle and the output from the state-space block was

- Roll

$$A = \begin{bmatrix} -4.2683 & -3.1716 \\ 4 & 0 \end{bmatrix} \qquad B = \begin{bmatrix} 2 \\ 0 \end{bmatrix}$$

$$C = \begin{bmatrix} 0.7417 & 0.4405 \end{bmatrix} \qquad D = \begin{bmatrix} 0 \end{bmatrix}$$

- Pitch

$$A = \begin{bmatrix} -3.9784 & -2.9796 \\ 4 & 0 \end{bmatrix} \qquad B = \begin{bmatrix} 2 \\ 0 \end{bmatrix}$$

$$C = \begin{bmatrix} 1.2569 & 0.6083 \end{bmatrix} \qquad D = \begin{bmatrix} 0 \end{bmatrix}$$

- Yaw

$$A = \begin{bmatrix} -0.0059 \end{bmatrix} \qquad B = \begin{bmatrix} 1 \end{bmatrix}$$

$$C = \begin{bmatrix} 1.2653 \end{bmatrix} \qquad D = \begin{bmatrix} 0 \end{bmatrix}$$

- Height

$$A = \begin{bmatrix} -5.8200 & -3.6046e^{-6} \\ 3.8147e^{-6} & 0 \end{bmatrix} \qquad B = \begin{bmatrix} 1024 \\ 0 \end{bmatrix}$$

$$C = \begin{bmatrix} 1.4907e^{-4} & 1.3191e^{3} \end{bmatrix} \qquad D = \begin{bmatrix} 0 \end{bmatrix}$$

- Pitch to u

$$A = \begin{bmatrix} -0.665 \end{bmatrix} \qquad B = \begin{bmatrix} 2 \end{bmatrix}$$

$$C = \begin{bmatrix} -3.0772 \end{bmatrix} \qquad D = \begin{bmatrix} 0 \end{bmatrix}$$

- Roll to v

$$A = \begin{bmatrix} -0.4596 \end{bmatrix} \qquad B = \begin{bmatrix} 2 \end{bmatrix}$$

$$C = \begin{bmatrix} 2.3868 \end{bmatrix} \qquad D = \begin{bmatrix} 0 \end{bmatrix}$$

Figure 2.1: State-Space Matrices

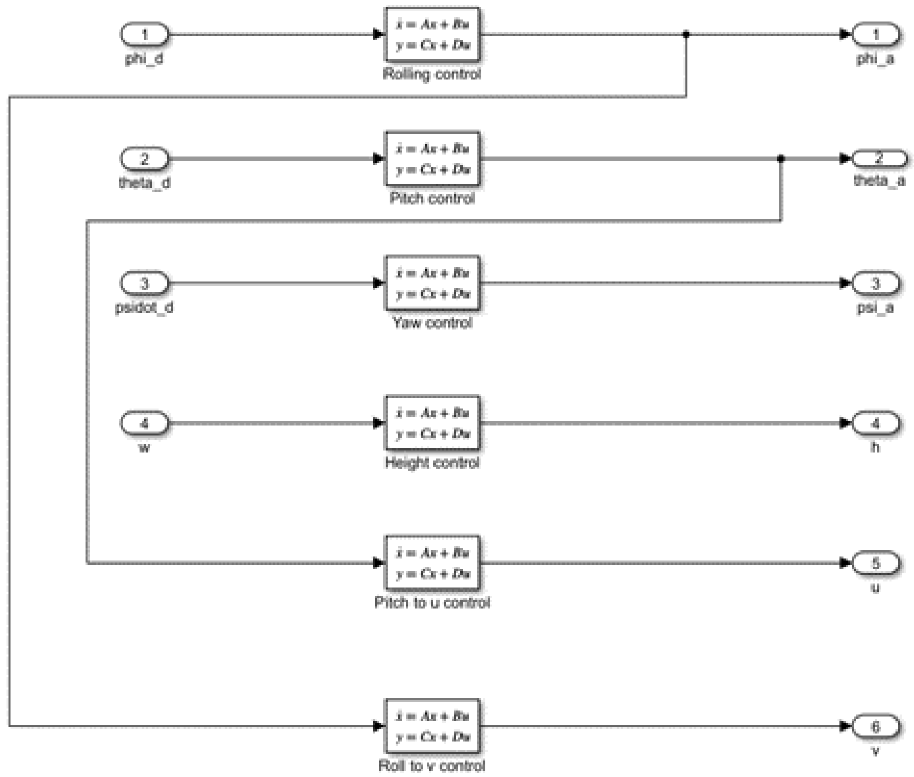

Figure 2.2: Simulink Design Dynamic Model

the actual roll angle. This actual roll angle was then fed into the roll to v control channel as input, to get the actual velocity v about y-axis.

For pitch control the input was desired pitch angle and the output was the actual pitch angle. This actual pitch angle was then fed into the pitch to u control channel which gives the actual velocity about x-axis.

Yaw control channel used desired yaw rate, also called as desired heading rate, for input while the output of this control channel is actual yaw angle. The last control channel used for the dynamic model of this quadrotor is altitude control, for this the input was desired vertical velocity w while the output was orientation along z- axis, which is also called altitude.

The Simulink design of Dynamic Model is shown in the figure below:

2.3 Development of Position/Orientation control system:

Position/Orientation Control System contains two types of state input, one is the target x, y, and z coordinate input and the other is actual x, y and z coordinates of the quadrotor. A reference yaw command is also given into the position controller to ensure that the quadrotor maintains the commanded yaw while operating for the target position.

The target x coordinate is given into a summation block while the actual x orientation of the quadrotor is given to the summation block as feedback. This command is fed into the PD controller, which gives corresponding derivative $\dot{x}$ (i.e., speed along x axis). This derived velocity u is given to the summation block whereas the actual velocity u_a is given as feedback in the same block. A PD controller is used again which finally gives the derivative $\ddot{x}$.

Similarly, target y position and actual y position of the quadrotor is also given into the summation block and then into the PD controller. The output from this PD controller is further given into a summation block where the actual velocity v is given as feedback. This output is then given into the PD controller to get acceleration along y-axis.

Actual heading angle psi, along with $\ddot{x}$ and $\ddot{y}$ is used to calculate the desired pitch angle and desired roll angle. The following equations [reference no?] are used for this:

$$\begin{bmatrix} \phi_d \\ \theta_d \end{bmatrix} = \begin{bmatrix} -\sin\psi & -\cos\psi \\ \cos\psi & -\sin\psi \end{bmatrix}^{-1} \tfrac{m}{U_1} \begin{bmatrix} \ddot{x}_d \\ \ddot{y}_d \end{bmatrix}$$

Since the derivation of these equations in (reference no) uses small angle approximation, we must ensure that the desired angles θ_d and ϕ_d are within the limit of –20° and 20°. Therefore, a saturation block is placed at the output of MATLAB function block.

We also need the desired vertical velocity w and the desired heading rate

2.4 Development of Position Estimation:

For estimating the position of the quadrotor, the output states from the dynamic model i.e., psi, psi, theta, h, u and v, are used. The dynamic model provides information about the actual value of states of the quadrotor. To acquire data regarding the actual x and y coordinated of the quadrotor we integrate the u and v velocities, respectively. The Simulink model of position estimation subsystem is shown in 2.3.

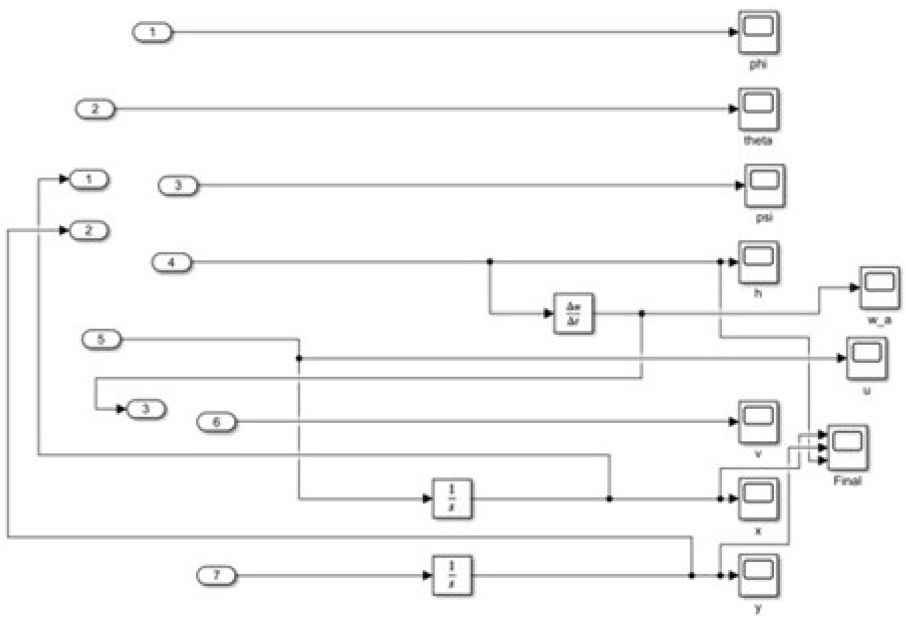

Figure 2.3: Position Estimation

2.5 Development of linear simulation model:

For the development of the linear simulation model, a reference subsystem was created which was used for giving the desired coordinate information to the position controller. The desired pitch and roll angles θ_d and ϕ_d along with the heading rate $\dot{psi}$ and desired vertical velocity wd goes into a summation block followed by a PID controller. To this summation block feedback is given of actual orientation of the respective states.

Further, these values are given into the dynamics model subsystem which is also explained in the previous section. The actual values obtained from the dynamic model are used in the position estimation subsystem to get information about Euler angles, position, and velocities of the quadrotor.

The reference position subsystem developed in for this project is an If-Else condition block which is used with Action block to activate a particular command for an If-Else condition. The reference command subsystem is shown in the 2.4.

The complete linear simulation system that was developed in Simulink is shown in the 2.5

PID Tuning methods used for this project were Inbuilt PID tuners provided by Simulink. For tuning PD controllers, the following steps were taken:

1. Proportional gain Kp was increased until steady oscillations were obtained

2. Derivate gain Kd was increased until the oscillations were critically damped

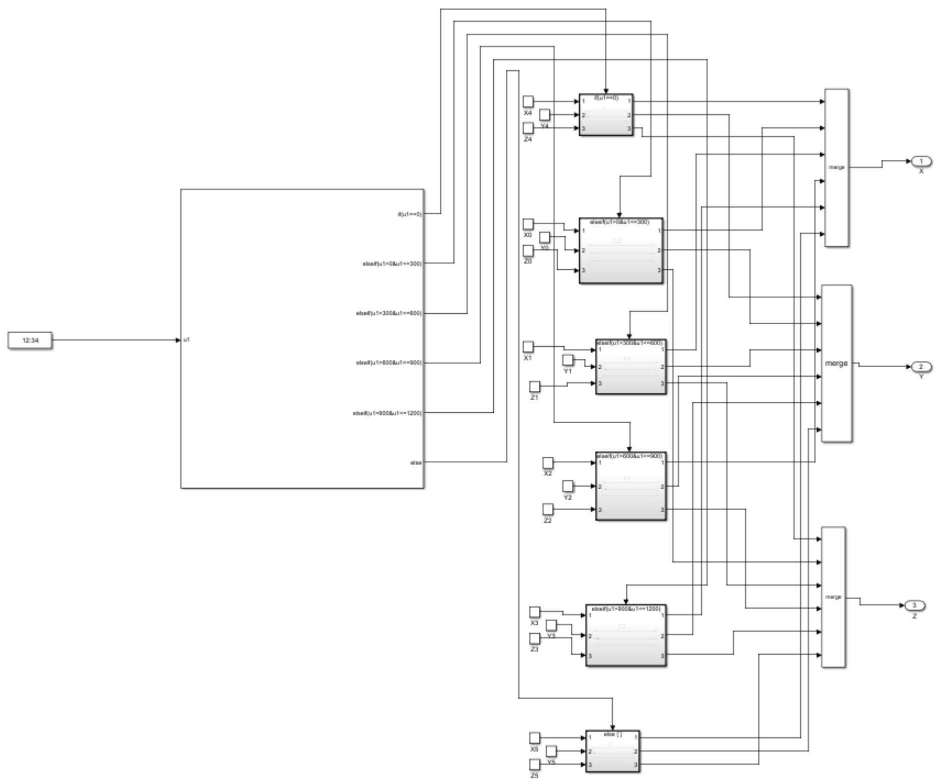

Figure 2.4: Reference Position

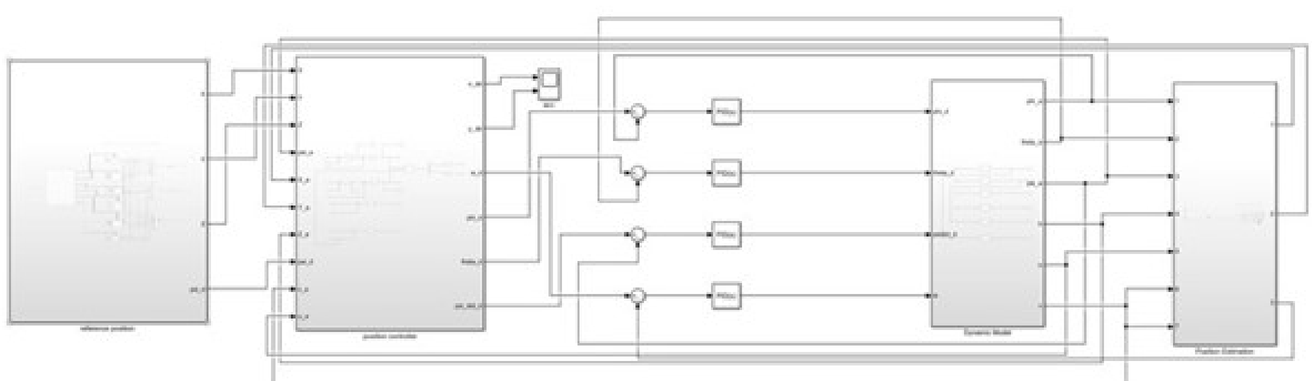

Figure 2.5: Complete Multirotor Dynamic System

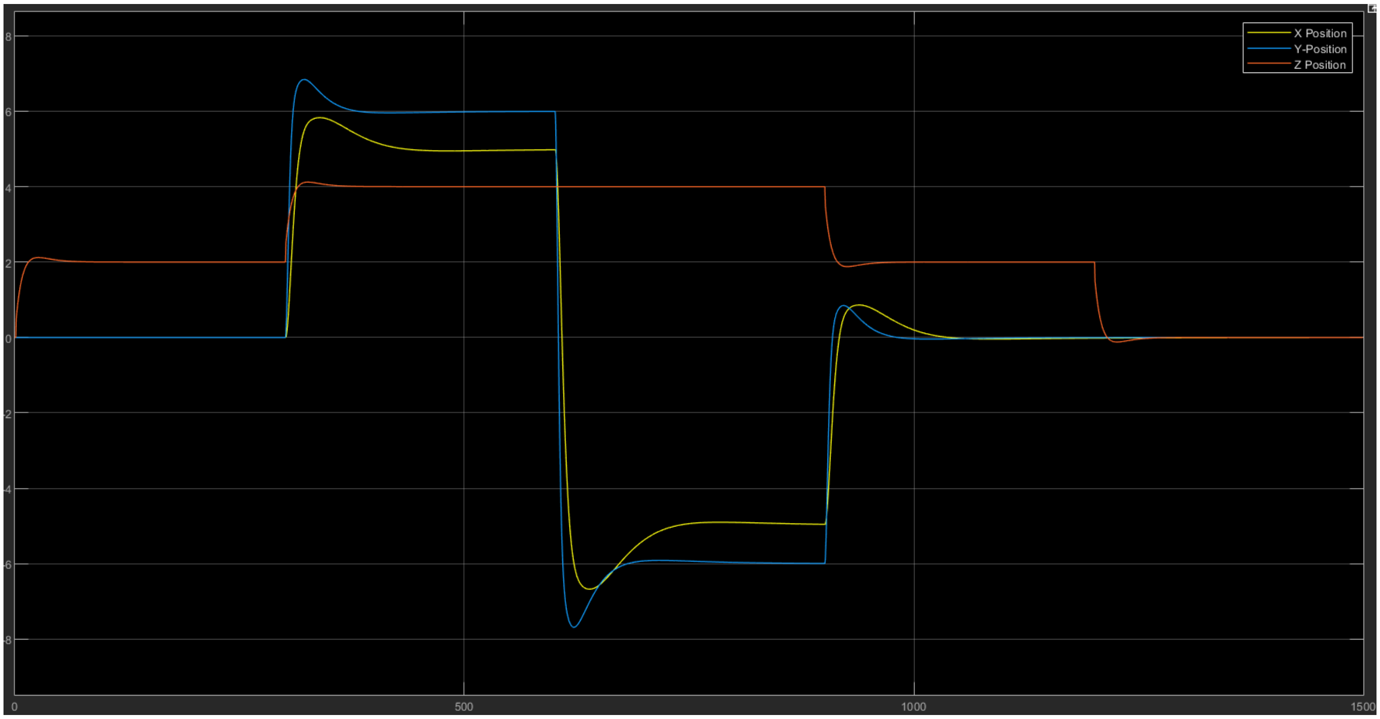

Figure 2.6: Quadrotor Simulation Solution

Total simulation time was taken to be 1500 seconds for the following four maneuvers:

1. Take off and hover at 2 meters above origin
2. Fly to the first target (x = 5 m, y = 6 m, h = 4 m) and hover
3. Fly to the second target (x = -5 m, y = -6 m, h = 4 m) and hover
4. Return to 2 meters above origin and land

2.6 Result

The quadcopter was successfully able to complete all the given maneuvers for this project. The simulation result, which shows the trajectory of x, y and z coordinates of the quadcopter for 1500 seconds is shown in .

In the figure the quadcopter is initially at x=0, y=0 and z=0 position, in the first 300 seconds it takes off and hover at x=0 y=0 and z=2 m. Next, the quadcopter reaches to x=5, y=6 and z=4.For the third maneuver it goes from its current position to the target position, x=-5, y=-6 and z=4 and then for the last maneuver is goes from current position to target position of x=0, y=0 and z= 2 and then back to the origin.

Chapter 3

Conclusions

The various lessons learned during the implementation of the fixed-wing and multi-rotor UAS configurations are listed below as follows:

- The theoretical values obtained using various theories like the Momentum Theory or Thin-Airfoil theory will vary largely from the obtained values of a flying system because of the assumptions and inconsistencies in data.
- Implementation of non-linear dynamics will resonate more with real-time data than with the implementation of a linearized system. Linearization can be done for learning about the properties of the system and simulation, but not for real-time flight dynamics.
- The obtained values of range and endurance in the fixed-wing and multirotor aircraft are large and unrealistic, mainly because of the inconsistencies in the data.
- About PID control
- About altitude hold control/position control
- About simulations and maneuvers

Bibliography