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Transition flight mode control strategy for Quad-Tiltrotor aircraft

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ABSTRACT

Tiltrotor has been developed as a platform that combined the features of multicopter aircaft and fixed-wing aircraft to overcome both shortcomings. However, since the dynamics of the system continuously changes during the transition mode, the conversion corridor analysis and the controller design are required separately to the hovering and forward modes. In this paper, nonlinear model of quad-tiltrotor was derived for constructing simulator and dynamics analysis. From the dynamic model analysis, the conversion corridor line was calculated using aerodynamic coefficients, and nonlinear controller for transition mode was derived. The control strategy was tested in the simulation model using the nonlinear dynamic modelling.

Keywords : Quad-Tiltrotor aircraft, Transition Mode, Mathematical model, Nonlinear control

I. INTRODUCTION

Recently, Muticopter aircrafts are widely used in taking aerial shot, agriculture and military purposes since they do not need a runway for take-off and landing, and can hover in particular spot in the sky. However, the multicopter has a operation limit that it cannot move fast forward for structural reasons and suffer from low endurance since multicopter overcomes the weights only using the thrust of the motors. On the other hand, fixed-wing aircraft requires a runway and cannot be hovered in place, while endurance and range of which are long. The one good alternative platform is a tiltrotor aircraft, which combines the advantages of multicopter and fixed-wing aircraft, respectively.

The tiltrotor aircraft has 3 operational modes; hovering mode, forward mode, and transition mode. It can fly vetically for take-off and landing or hover when it's in hovering mode. It generates forward thrust and can fly at high speed by tilting motors in logitudinal directions in during forward mode. Transition mode is a process of transforming between a multicopter mode and a fixed wing mode. At this mode, tiltrotor aircraft has the ambiguous dynamic characteristics which is neither that of complete multicopter dynamics nor that of complete fixed-wing dynamics. Since it has ambiguous dynamic characteristics, controller for transition mode needs to be designed separately. Since the transition mode is a bridge between two modes, the controller for transition mode is the key point of tiltrotor controller design and has been studied in many papers.

The concept of conversion corridor is used to connect the two modes and lead the aircraft convert securely in the section with ambigous dynamic characteristics. Conversion corridor is limit range of velocity according to a specific tilt angle and to create tilt path in the transition process. The lower limit of the conversion corridor represents the stall speed at specific tilt angle, and the upper limit represents the performance limit of tiltrotor. (Maisal et al. 2000) Since conversion corridor leads safe transition between two modes, there have been various studies for the design of the conversion corridor.

The conversion corridor was constructed with finding trim condition in various thrusts and attitudes using a wind tunnel or cfd tool to form conversion corridors in (Muraoka et al. 2012)(Diaz et al. 2004). In the case of KARI TR-60, they constructed the optimal conversion corridor line through manual flight experiments and performance analysis, and set the appropriate velocity range for conversion corridor.(Kang et al. 2008)(Kang et al. 2014) Conversion corridor has been actively studied since it is necessary for tiltrotor operation and control. However, many studies use wind tunnel or cfd tools for analyze the large size aircraft or UAV.

Even if the conversion corridor is used for control, it is impossible to control the tiltrotor in transition mode using only one linear controller due to the system dynamics changing depending on the tilt angle and nonlinear aerodynamics. For this reason, the studies for controller design using gain scheduling is done in (Öznalbant et al. 2018)(Yuksek et al. 2016). The gain is scheduled by the speed or tilt angle using linear control method like PID. For overcoming these nonlinearity, intelligent controllers such as fuzzy and neural networks are used for controller design.(Chang et al. 2013)(Kim et al. 2010)(Ta et al. 2012) Recently, nonlinear control method is used for overcome the complicated aerodynamic model.(Kong et al. 2018)

In this paper, the controller of the quad-tiltrotor aircraft in transition mode is designed which uses four motors and tilts only two motors in front of the tiltrotor aircraft. In section II, mathematical modeling of quad-tiltrotor will be derived for simulation and analyze the dynamic characteristics. In section III, conversion corridor and control strategy will be derived during transition mode. Conversion corridor will be designed using a mathematical model analysis for small UAV and the transition flight mode controller uses nonlinear feedback controller for considering the nonlinear dynamics such as aerodynamics during the transition mode. In section IV, The simulation results for cases will be shown and end with conclusion and future works in section V

II. MATHMATECAL MODELING

In this section, the dynamics of the quad-tiltrotor aircraft in transition mode is derived. The appearance of the quad-tiltrotor is shown in Figure 1. As shown in Figure 1, the quad-tiltrotor is formed with wingbody aircraft, and the four motors are mounted on the boom which are symmetric mounted to the center of gravity of the aircraft. All the motors are tilted at same angle which represents the upward angle of the motors with respect to the body frame x-axis. The motor generates thrust in the same direction as the tilt angle of the motor and only positive thrust is generated which are limted the range. It is assumed that using the motor to control quad-tiltrotor during the transition mode without using flap behind the wing and the center of gravity and the center of the aerodynamics are at the same position.

Figure 1. Appearance of quad-tiltrotor

Frames

Prior to deriving the equations of motion of the quad-tiltrotor, it is necessary to define the frames and transformation between the frames to express the motion of the aircraft. Inertial frame I refers to the frame in which gravity acts on z-axis which represents the NED frame. The body frame B represents a frame aligned with the fuselage and the origin of the frame represents the center of mass. The wind frame represents the aerodynamic acting on the aircraft where the x-axis represents the direction of the aircraft's speed.

In the inertial frame, position vector and velocity vector are presented as $\mathbf{p}_I = \begin{bmatrix} x_I & y_I & z_I \end{bmatrix}^T$ and $\mathbf{v}_I = \begin{bmatrix} \dot{x}_I & \dot{y}_I & \dot{z}_I \end{bmatrix}^T$. In the body frame, velocity vector is represented as $\mathbf{v}_B = \begin{bmatrix} u & v & w \end{bmatrix}^T$. The transformation between the inertial frame and the body frame is obtained by the rotational transformation by the Euler angles. Euler angle vector $\Theta = \begin{bmatrix} \phi & \theta & \psi \end{bmatrix}^T$ represents roll, pitch and yaw angle. Transformation matrix between body frame and inertial frame is follow as

$$R_{B\to I} = T_1(\phi)T_2(\theta)T_3(\psi) = \begin{bmatrix} c_\theta c_\psi & c_\theta s_\psi & -s_\theta \\ -c_\phi s_\psi + s_\phi s_\theta c_\psi & c_\phi c_\psi + s_\phi s_\theta s_\psi & s_\phi c_\theta \\ s_\phi s_\psi + c_\phi s_\theta c_\psi & -s_\phi c_\psi + c_\phi s_\theta s_\psi & c_\phi c_\theta \end{bmatrix} \tag{1}$$

$$Where \;\; R_{I\to B} = (R_{I\to B})^{-1} = R_{I\to B}{}^T$$

$R_{1\to 2}$ is a transformation matrix for 1frame to 2 frame, and $T_i(\theta)$ is a rotational matrix in θ angle for i-axis. Similar as Eq.(1), angular rate transformation between inertial frame and body can be obtained.

$$\begin{bmatrix} \dot{\phi} \\ \dot{\theta} \\ \dot{\psi} \end{bmatrix} = \begin{bmatrix} P + [Q\sin\phi + R\cos\phi]\tan\theta \\ Q\cos\phi - R\sin\phi \\ [Q\sin\phi + R\cos\phi]\sec\theta \end{bmatrix} \tag{2}$$

Equation Of Motions

The equation of motion for quad-tiltrotor are derived using Newton's second law. Since the model of forces and moments are generated based on the body frame, Newton's second law need to be derived for the body frame.

$$\dot{\vec{\mathrm{v}}}_{\mathrm{B}} + \omega \times \vec{v}_B = \frac{1}{m}\vec{F} \tag{3}$$

$$\mathbf{I}\dot{\omega} + \omega \times \mathbf{I}\omega = \vec{M} \tag{4}$$

$\omega = \begin{bmatrix} p & q & r \end{bmatrix}^T$ denotes body angular rate and $\mathbf{I}$ denotes the moment of inertia matrix for the aircrafts where

$$\mathbf{I} = \begin{bmatrix} I_{xx} & 0 & I_{xz} \\ 0 & I_y & 0 \\ I_{xz} & 0 & I_z \end{bmatrix} \tag{5}$$

Eq.(3) and Eq.(4) can be reassembled for $\dot{\mathrm{v}}_{\mathrm{B}}$ and $\dot{\omega}$ as follows

$$\begin{bmatrix} \dot{U} \\ \dot{V} \\ \dot{W} \end{bmatrix} = \begin{bmatrix} -\left[QW - RV\right] + \frac{1}{m}X \\ -\left[RU - PW\right] + \frac{1}{m}Y \\ -\left[PV - QU\right] + \frac{1}{m}Z \end{bmatrix} \tag{6}$$

$$\begin{aligned} \dot{P} &= \frac{1}{I_x I_z - I_{xz}{}^2}\left\{ I_{xz}(I_x - I_y + I_z)PQ - \left[I_z(I_z - I_y) + I_{xz}{}^2 \right]QR + I_z L + I_{xz} N \right\} \\ \dot{Q} &= \frac{1}{I_y}\left\{ -(I_x - I_z)PR + I_{xz}(R^2 - P^2) + M \right\} \\ \dot{R} &= \frac{1}{I_x I_z - I_{xz}{}^2}\left\{ -I_{xz}(I_x - I_y + I_z)QR + \left[I_x(I_x - I_y) + I_{xz}{}^2 \right]PQ + I_{xz} L + I_x N \right\} \end{aligned} \tag{7}$$

Forces And Moments Model

$\vec{F}$ represents total force and $\vec{M}$ represents total moments actin on the quad-tiltrotor. Forces and Moments can divide in three dominant categories which are gravity, aerodynamics, motor.

$$\begin{aligned} \vec{F} &= \vec{F}_{Gravity} + \vec{F}_{Aero} + \vec{F}_{\mathrm{Motor}} \\ \vec{M} &= \vec{M}_{Gravity} + \vec{M}_{Aero} + \vec{M}_{\mathrm{Motor}} \end{aligned} \tag{8}$$

Gravity

Below the title, Among the forcesand moments acting on the quad-tiltrotor, the gravitational force acts on the center of mass and acts in the parallel to the z_I axis. In the previous section, the equation of motion is derived to the body frame, so the expression for the body frame for gravitational force is needed. The force acting on the body frame is obtained by rotational transformation in Eq.(9), and the moment does not act since gravity acts entirely on the center of mass.

$$\vec{F}_{Gravity} = T_1(\phi)T_2(\theta)T_3(\psi)\begin{bmatrix} 0 \\ 0 \\ mg \end{bmatrix} = mg\begin{bmatrix} -\sin\theta \\ \sin\phi\cos\theta \\ \cos\phi\cos\theta \end{bmatrix} \tag{9}$$

$$\overrightarrow{M}_{Gravity} = \begin{bmatrix} 0 \\ 0 \\ 0 \end{bmatrix} \tag{10}$$

Aerodynamics

Since quad-tiltrotor has wing body shaped, the aerodynamic forces and moments are dominant when the speed of the aircraft is large enough. In hovering mode, the aerodynamic forces and moments are not dominant, but in transition mode, it becomes dominant effects while mode conversion.

Aerodynamic forces are defined for the wind frame, but the forces should be derived in body frame axis. Therefore rotational transformation is required for it.

$$\overrightarrow{F}_{Aero} = \begin{bmatrix} F_{Ax} \\ F_{Ay} \\ F_{Az} \end{bmatrix} = R_{w\to B} \begin{bmatrix} D \\ Y \\ L \end{bmatrix} \tag{11}$$

D,Y, and L denote aerodynamic drag, lift and lateral force, respectively. Rotation matrix $R_{w\to B}$ represent the transformation matrix for wind frame to body frame.(Gryte et al. 2018)

$$R_{w\to B} = \begin{pmatrix} \cos\alpha\cos\beta & \cos\alpha\sin\beta & -\sin\alpha \\ -\sin\beta & \cos\beta & 0 \\ \cos\beta\sin\alpha & \sin\alpha\sin\beta & \cos\alpha \end{pmatrix} \tag{12}$$

From the matrix above, $\alpha = \tan^{-1}\left(\frac{W}{U}\right)$, and $\beta = \tan^{-1}\left(\frac{V}{V_T}\right)$ represent angle of attack, sideslip angle, and total velocity $V_T = \sqrt{U^2+V^2+W^2}$ can be derived from the body velocity of the aircraft. The aerodynamic forces and moments are determined by the current state of the aircraft and the control surfaces, and the aerodynamic model for the simulation is expressed in Eq.(13), and Eq.(14)

$$\begin{bmatrix} D \\ Y \\ L \end{bmatrix} = \frac{1}{2}\rho V_T{}^2 S \begin{bmatrix} C_D(\alpha,\beta,Q,\delta_e) \\ C_Y(\beta,P,R,\delta_a,\delta_r) \\ C_L(\alpha,Q,\delta_e) \end{bmatrix} \tag{13}$$

$$\begin{bmatrix} l \\ m \\ n \end{bmatrix} = \frac{1}{2}\rho V_T{}^2 S \begin{bmatrix} bC_l(\beta,P,R,\delta_a,\delta_r) \\ cC_m(\alpha,Q,\delta_e) \\ bC_n(\beta,P,R,\delta_a,\delta_r) \end{bmatrix} \tag{14}$$

In aerodynamic force and moment equations, ρ represents for air density, S for wing area, b for wing span length, c for cord length, and δ_a, δ_r for aileron and rudder angle. The aerodynamic coefficients C_D, C_L, C_Y, C_l, C_m, *and* C_n are determined by the state of the aircraft and control surface which are used to express the aerodynamic forces and moments.

$$\begin{bmatrix} C_D \\ C_Y \\ C_L \end{bmatrix} = \begin{bmatrix} C_{D0} + C_{D\alpha 1}\alpha + C_{D\alpha 2}\alpha^2 + C_{DQ}\dfrac{c}{2V_T}Q + C_{D\beta 1}\beta + C_{D\beta 2}\beta^2 + C_{D\delta_e}\delta_e^{\ 2} \\ C_{Y0} + C_{Y\beta}\beta + C_{YP}\dfrac{b}{2V_T}P + C_{YR}\dfrac{b}{2V_T}R + C_{Y\delta_a}\delta_a + C_{Y\delta_r}\delta_r \\ C_{L0} + C_{L\alpha}\alpha + C_{LQ}\dfrac{c}{2V_T}Q + C_{Y\delta_e}\delta_e \end{bmatrix} \tag{15}$$

$$\begin{bmatrix} C_l \\ C_m \\ C_n \end{bmatrix} = \begin{bmatrix} C_{l0} + C_{l\beta}\beta + C_{lP}\dfrac{b}{2V_T}P + C_{lR}\dfrac{b}{2V_T}R + C_{l\delta_a}\delta_a + C_{l\delta_r}\delta_r \\ C_{m0} + C_{m\alpha}\alpha + C_{mQ}\dfrac{b}{2V_T}Q + C_{m\delta_e}\delta_e \\ C_{n0} + C_{n\beta}\beta + C_{nP}\dfrac{b}{2V_T}P + C_{nR}\dfrac{b}{2V_T}R + C_{n\delta_a}\delta_a + C_{n\delta_r}\delta_r \end{bmatrix} \tag{16}$$

Motor

Since the forces and moments hue to the thruster are onof the controllable parts, it is necessary to analyze it to construct the allocator in the cotroller design. The motor numbers are numbered counterclockwise starting from the front right motor. c_T is a moment-thrust ratio when the motor generates the thrust, which axial moments from the motor can be obtained by $M_i = c_T T_i$. The forces and moments act in body frame due to thrust distribution are derived as Eq.(17) and Eq.(18)

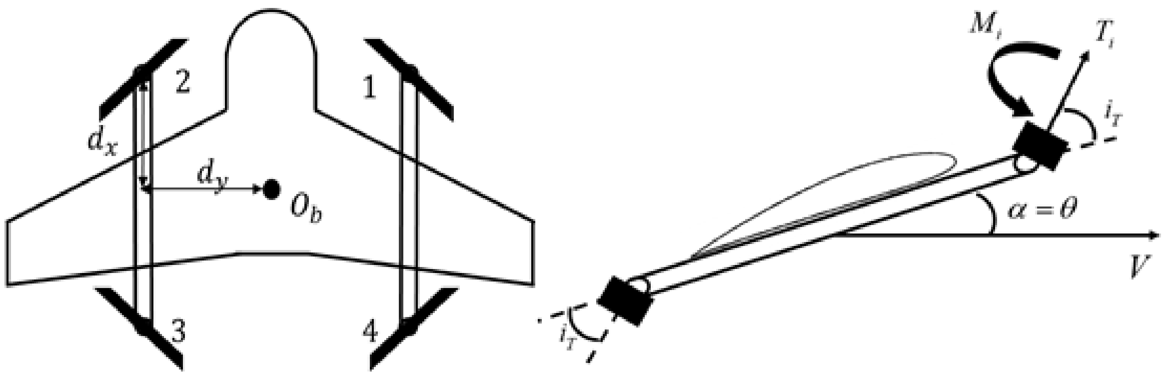

Figure 2. Thrust and moments act by motor

$$\vec{F}_{\text{Motor}} = \begin{bmatrix} F_{Tx} \\ F_{Ty} \\ F_{Tz} \end{bmatrix} = \begin{bmatrix} (T_1 + T_2 + T_3 + T_4)\cos i_T \\ 0 \\ -(T_1 + T_2 + T_3 + T_4)\sin i_T \end{bmatrix} \tag{17}$$

$$\vec{M}_{\text{Motor}} = \begin{bmatrix} M_{Tx} \\ M_{Ty} \\ M_{Tz} \end{bmatrix} = \begin{bmatrix} d_y[(T_2 - T_1 + T_3 - T_4)\sin i_T] + c_T[(T_2 - T_1 - T_3 + T_4)\cos i_T] \\ d_x[(T_1 + T_2 - T_3 - T_4)\sin i_T] \\ d_y[(T_2 - T_1 + T_3 - T_4)\cos i_T] + c_T[-(T_2 - T_1 - T_3 + T_4)\sin i_T] \end{bmatrix} \tag{18}$$

III. CONTROL STRATEGY

In this section, the controller of quad-tiltrotor in transition mode is designed. In the transition mode, only the motor is used to control,and control surface is not used. The controller is divided into 4 blocks; conversion corridor, outer loop, inner loop, and control allocation. Conversion corridor is designed to control the tilt angle according to the current state of the aircraft, outer loop generates body angular acceleration commands based on angle commands, inner loop generates force and moment commands based on body angular rate commands, and control allocation generates thrust commands of each motors using force and moment commands.

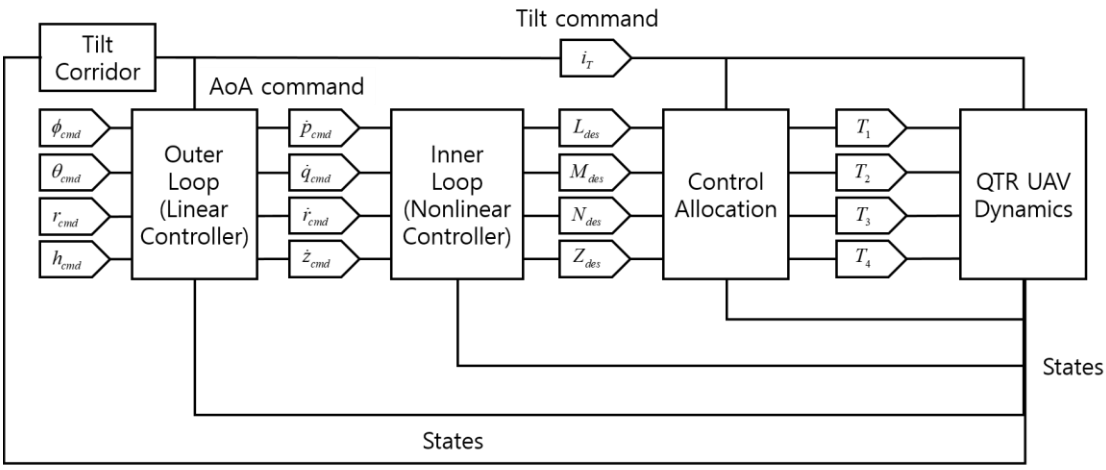

Figure 3. Control block diagram for quad-tiltrotor

Conversion Corridor

The conversion corridor is a tilt angle limitation of the motor according to the current state of the tiltrotor in the transition mode and it is an important role in operating a tiltrotor since it help tiltrotor can convert quickly from hovering mode to forward mode. Commonly, the converion corridor is plotted with tilt angle according to the total speed of the aircraft. In many cases, the conversion corridor of the tiltrotor is constructed the trim condition test using the wind tunnel.(Muraoka et al. 2012) In this paper, we obtained a conversion corridor using known aerodynamic coefficients without using a wind tunnel.

Since the altitude is controlled in transition mode, alttitude $h = h_o$ does not change and then inertial velocity to z axis $\dot{z}_I = 0$. Then, flight path angle, angle of attack, and pitch angle is derived as.

$$h = h_o \text{ , } V_z = 0 \rightarrow \gamma = 0 = \theta - \alpha \rightarrow \alpha = \theta \tag{19}$$

Using the assumptions above, the changeable factors in trim condition defined as pitch angle and the total force if only the longitudinal motion is considered. The limitations of the changeable factors are defined as

$$\begin{gathered} 0.2 * 4T_{\max} \le F \le 0.6 * 4T_{\max} \\ 0 \le \theta \le 10^{\circ} \end{gathered} \tag{20}$$

For maintaining the altitude, the force on the z-axis for inertial frame should be balanced and the trim velocity can be derived from the specific pitch angle and total thrust.

$$F\cos(i_T+\alpha)+L \;=\; mg \quad \left(L=\frac{1}{2}\rho S C_L(\alpha)V^2 \right)$$
$$\rightarrow V=\sqrt{\frac{mg-F\sin(i_T+\alpha)}{\frac{1}{2}\rho S C_L(\alpha)}} \tag{21}$$

From the trim velocity in specific pitch angle and total thrust, the excess power P_e is defined in Eq.(22) which is used for cost function in conversion corridor construction. For each trim condition, the case is excluded if the motor moment cannot be overcome aerodynamic moment by the distribution of the thrust.

$$P_e = TV - DV = V\left[T-D\right] = V[F\cos(\alpha+i_T)-\frac{1}{2}\rho S C_D(\alpha)V^2] \tag{22}$$

The conversion corridor is obtained from the maximum excess power velocity at specific tilt angle from various pitch angle and total thrust. The plot of excess power and conversion corridor is on Figure 4, Figure 5 and Figure 6.

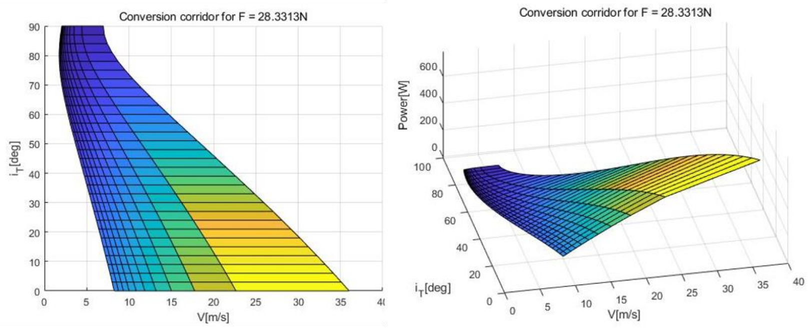

Figure 4. Excess power surface for specific total thrust

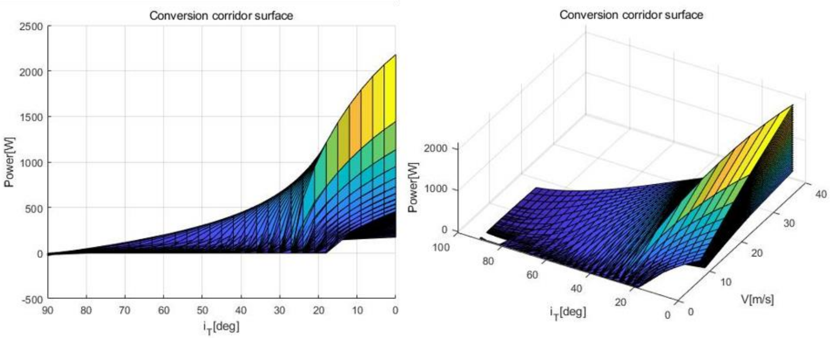

Figure 5. Excess power surface

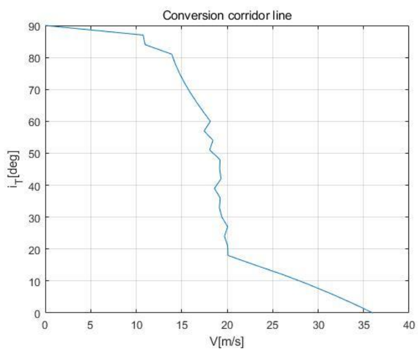

Figure 6. Conversion corridor line for quad-tiltrotor

Outer Loop

For ϕ *and* θ, PI controller is used to generate the p_{cmd} *and* q_{cmd}, and $\dot{p}_{cmd}$, $\dot{q}_{cmd}$ it generated using PID controller from p_{cmd} *and* q_{cmd}. For r *and* z_I, only PID controller is used to generate the $\dot{r}_{cmd}$ *and* $\ddot{z}_{cmd}$

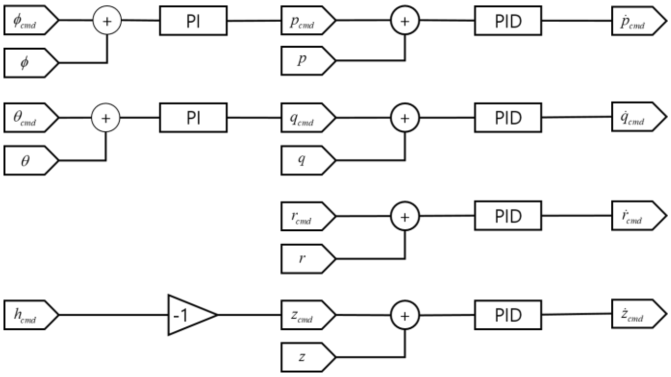

Figure 7. Outer loop Block

Inner Loop

In the transition mode, the speed is getting larger which makes the aerodynamic forces and moments becomes a dominant factor. Therefore, the force and moment commands need to be generated by considering the aerodynamics. Since these aerodynamic forces are nonlinear, the nonlinear model based controller is used for precise control.

For the moment commands, current states of the aircraft and body angular commands are used with reference to Eq.(4) and $\bar{q} = \frac{1}{2}\rho V^2$ represents a dynamic pressure.

$$\begin{bmatrix} L_{req} \\ M_{req} \\ N_{req} \end{bmatrix} = \begin{bmatrix} L_{prop} \\ M_{prop} \\ N_{prop} \end{bmatrix} = \begin{bmatrix} I_x \dot{p}_{req} - I_{xz}\dot{r} + (I_z - I_y)qr - I_{xz}pq - C_l S\bar{q}b \\ I_y \dot{q} + (I_x - I_z)pr + I_{xz}(p^2 - r^2) - C_m S\bar{q}\,\bar{c} \\ I_x \dot{r} - I_{xz}\dot{p} + (I_y - I_x)pq + I_{xz}qr - C_n S\bar{q}b \end{bmatrix} \tag{23}$$

As similar as moment commands, z-axis force command is derived by reference Eq.(1), and Eq.(6)

$$\begin{aligned} Z_{req} = Z_{prop} &= \frac{m(\dot{z}_I - g)}{\cos\theta\cos\phi} - L\cos\gamma \\ &= \frac{m(\dot{z}_I - g)}{\cos\theta\cos\phi} - C_L S\bar{q}\cos\gamma \end{aligned} \tag{24}$$

Control Allocation

The control allocation matrix is derived using a geometric parameters in Figure 2 and motor aerodynamic parameter c_T. The control allocation matrix convert force and moment commands to the thrust commands for each motor.

$$\begin{bmatrix} L_{prop} \\ M_{prop} \\ N_{prop} \\ Z_{prop} \end{bmatrix} = \mathbf{T}_{CA} \begin{bmatrix} T_1 \\ T_2 \\ T_3 \\ T_4 \end{bmatrix} \Rightarrow \begin{bmatrix} T_1 \\ T_2 \\ T_3 \\ T_4 \end{bmatrix} = \mathbf{T}_{CA}^{-1} \begin{bmatrix} L_{prop} \\ M_{prop} \\ N_{prop} \\ Z_{prop} \end{bmatrix} \tag{25}$$

$$\mathbf{T}_{CA} = \begin{bmatrix} -d_y \sin i_T - c_T \cos i_T & d_y \sin i_T + c_T \cos i_T & d_y \sin i_T - c_T \cos i_T & -d_y \sin i_T + c_T \cos i_T \\ d_x \sin i_T & d_x \sin i_T & -d_x \sin i_T & -d_x \sin i_T \\ -d_y \cos i_T + c_T \sin i_T & d_y \cos i_T - c_T \sin i_T & d_y \cos i_T + c_T \sin i_T & -d_y \cos i_T - c_T \sin i_T \\ -\sin i_T & -\sin i_T & -\sin i_T & -\sin i_T \end{bmatrix} \quad (26)$$

IV. SIMULATION RESULTS

In following section, simple simulation results will be presented by applying the controller presented at previous section. The simulation model is used dynamic model described in section. II. The 'skywalker x8' aerodynamic coefficients are used for simulation in (Gryte et al. 2018). Tilt angle is started from the 80 degree for accelerating at the start and end at the 10 degree since tilt angle is too small that tiltrotor cannot generate enough pitch moment using the thrust to overcome aerodynamic moments.

First Simulation – Regulation

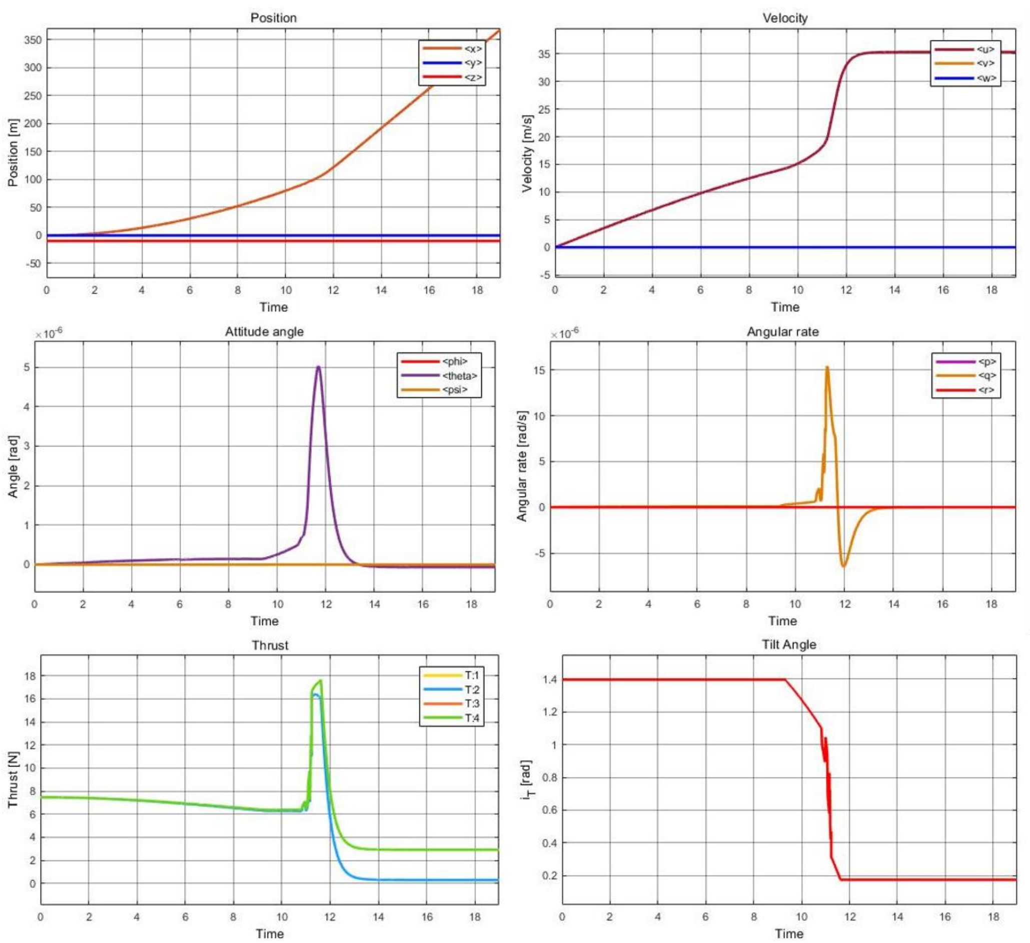

Figure 8. First simulation results

In the first simulation, a regulating is tested which the commands are all zero in the simulation. Figure 8 denotes the results of it. The attitude and altitude are regulated well during the most of the transition sequence. During the transition process, there is a region where the tilt angle changes abruptly for fast conversion. At this moment, the pitch angle perturbed a little, but it is compensated within the short time.

Second Simulation – Specific Tilt Angle

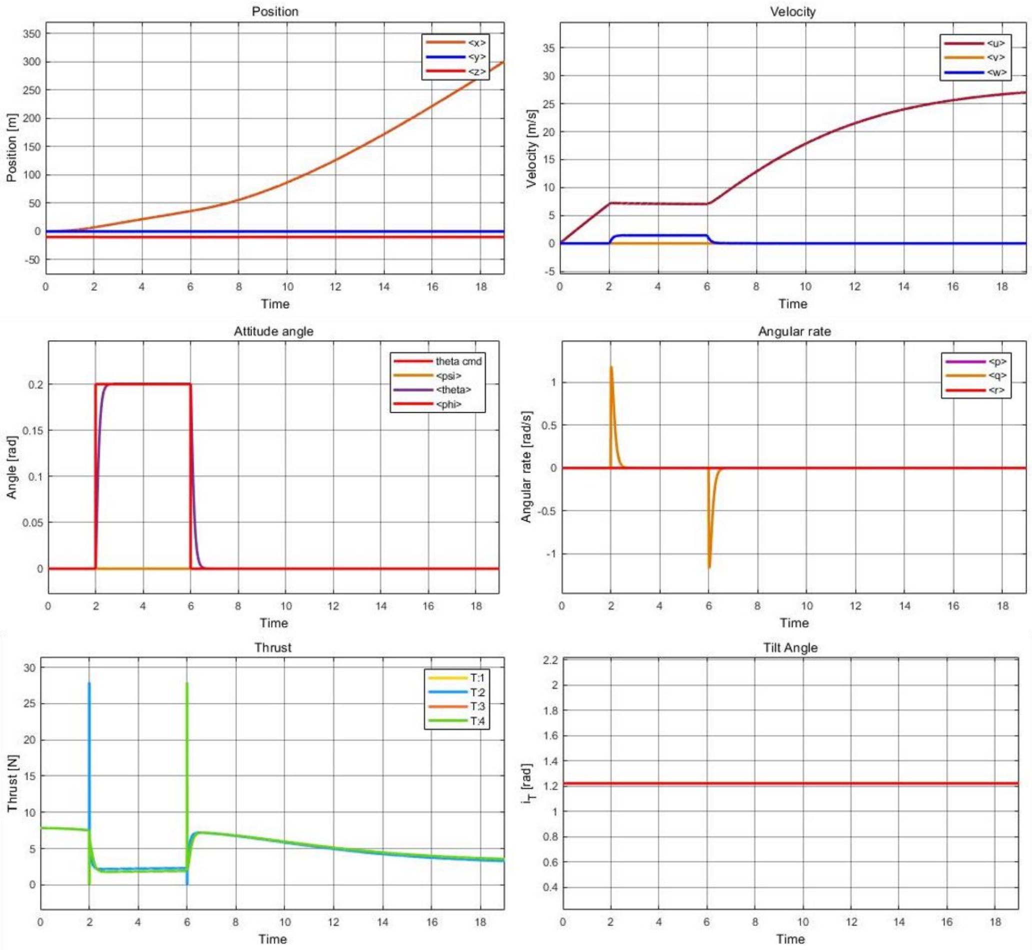

Figure 9. Second simulation results

Second simulation is the simulation result for the specific tilt angle. The command is given between 2 and 6 second along the pitch angle of 0.2rad, and the tilt angle was commanded to maintain 1.2rad. As a result of the pitch command tracking performance, the tracking was settled in 1 second. Also, the forward speed is stagnated while the command applied and it can delay the transition since the transition can be finished at satisfied velocity which the lift can overcome a weight of the aircraft.

V. CONCLUSION AND FUTURE WORKS

In this paper, mathematical modeling of quad-tiltrotor is derived which is used for a simulator built and a controller is constructed in transition mode. For the controller design, the conversion corridor line representing the tilt angle according to the current aircraft speed was obtained by using the maximum excess power based on the mathematical model without the wind tunnel test. The controller for transition mode is designed with the outer loop using the PID, the inner loop using nonlinear model of quad-tiltrotor, and a control allocation for thrust distribution. Simulation was performed with the designed controller for several cases on a simulator based on a dynamic model derived. Simulation for actual transition process which the command is not given, and the tracking ability according to the attitude command at the specific tilt angle is performed.

For the next step, we will design a model-based adaptive controller that can update the controller by real-time estimation of the aircraft aerodynamic parameters to generalize the aerodynamic coefficient-dependent controller. In addition, controller design for hovering and forward mode will be progressed. After the controller designed for quad-tiltrotor operation in parts, the flight test for the prototype of quad-tiltrotor will be carried out.

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