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PI-Based Fault Tolerant Control For Fixed-Wing UAVs Using Control Allocation

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Abstract: The fault-tolerant control of fixed-wing unmanned aerial vehicle in the presence of actuator failure is investigated. A longitudinal and lateral proportional + integral + derivative control structure is used in combination with sequential least squares to develop the virtual control for the unmanned aerial vehicle with redundant actuators as well as dynamic uncertainties. The proportional + integral + derivative control structure ensures that the unmanned aerial vehicle has good stability and tracking performance under normal operation. The control reallocation makes sure that, once a failure occurs and has been detected and isolated, the virtual control may be redistributed among the remaining relevant actuators with the inclusion of rate and amplitude constraints. The effectiveness of the proposed control methodology is illustrated by simulation.

Keywords: Fault-tolerant control, control allocation, PID control, sequential least squares, UAV

1. INTRODUCTION

Performance and robustness are two highly-valued characteristics that are essential in any method to be applied in the design of control systems. In the aviation industry, the safety of passenger aircraft and environmental effects have always been an important issue. In history, a number of aircraft disasters were due to damaged or faulty aircraft resulting in the aircraft becoming unstable during flight. Occurrence of these types of accidents leads to increasing interests in the emerging field of fault-tolerant control systems (FTCS) and fault detection and diagnosis (FDD). In the presence of a FDD system, the degree of danger of the faults or failures could be decreased, the safety of aircraft could be increased, and therefore the lives of passengers could be protected [Khan, (2016)].

Existing FTCS can be divided into two groups: Passive FTCS (PFTCS) that does not include controller reconfiguration, and Active FTCS (AFTCS) that integrates controller reconfiguration based on the outputs of a FDD module. The difference between PFTCS and AFTCS is that the controllers in PFTCS are designed to be robust against a class of presumed faults, while the controllers in AFTCS can be reconstructed on-line in case of faults occurring in the system [Ma, (2011)]. FTC is implemented by means of redundancy in the software such as in [Coopmans, Podhradsky, & Hoffer, (2015)] or hardware (sensors, control actuators and processors) as illustrated in [Wang, et al., (2016)], [Yu, Liu, & Zhang, (2016)] and [Hansen & Blanke, (2014)]. This is also achieved by the use of reconfigurable control as in multiple model switching [Guo, Zhang, & Jiang, (2010)] and optimal (i.e., dynamic control allocation and bisecting edge searching) [Peni, et al., (2014)], [Qian, Jiang, & Xu, (2012)] and non-optimal (i.e., daisy chaining, direct allocation, generalized inverse and pseudo inverse weighted inverse, and linear and quadratic programming) control allocation [Girish, et al., (2015)], [Valavanis, Oh, & Piegl, (2008)]. Other methods include Fault Detection, Diagnosis and Isolation (FDDI) methods such as direct measurement [Yoon, et al., (2015)], system identification [Hoffer, et al., (2015)], and model-based approaches (Kalman filtering and fault model comparison) [Chamseddine, Amoozgar, & Zhang, (2015)], [Freeman & Balas, (2014)]. As well as adaptive control methods like Gain Scheduling [Sadeghzadeh, (2015)], Auto-Tuning, Fuzzy logic (FL) Controllers [Hafez & Kamel, (2016)], Model Reference Adaptive Control (MRAC) [Zhou, Chen, & Jiang, (2015)], or Model Predictive Control [Khan, (2016)].

The control structure proposed in this paper includes PID controller associated with control allocation, aimed at accommodating multi-plicative faults in the plant and faults in the actuators. The hybrid controller presents a PID controller which, in the fault-free case, compensates the nonlinear modelling errors. For a faulty case, the control re-allocator provides to the system a compensation which the other controller is not capable to by redistributing the required control effort among the remaining control surfaces.

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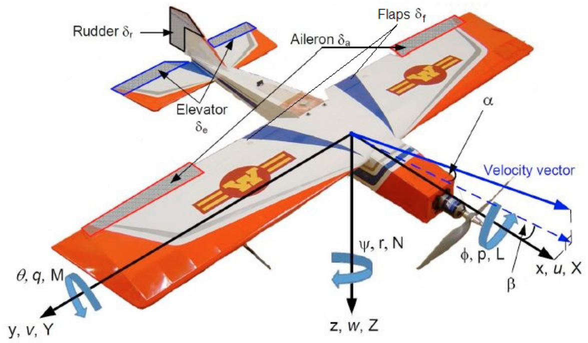

Fig. 1. Aircraft body-axis forces and moments [Paw, (2009)]

2. UAV MATHEMATICAL MODELLING

The fixed-wing UAV utilised in this research is a commercial of-the-shelf (COTS) Radio-Controlled (RC) aircraft, the Ultra Stick [Dorbantu, (2011)]. The airframe consists of a fuselage, with a forward, propellor joined to a rectangular wing. The wing has a symmetrical airfoil and both inboard flaps and ailerons. The tail of the UAV has elevators and a rudder. The UAV system references are shown in Figure 1. The important parameters and geometric specifications of the UAV are presented in Table 1 [Paw, (2009)].

Table 1. Table of the Ultra Stick Parameters

Parameter	Value	Symbol
Mass (kg)	7.411	$MTOW$
CG from firewall (m)	0.315	CG
Aero Ref from firewall (m)	0.320	A_{Ref}
Moment of Iner. about x-axis ($kg.m^2$)	0.8568	I_x
Moment of Iner. about y-axis ($kg.m^2$)	1.0095	I_y
Moment of Iner. about z-axis ($kg.m^2$)	1.7005	I_z
Prod. of Iner. about x-y axes ($kg.m^2$)	0	I_{xy}
Prod. of Iner. about x-z axes ($kg.m^2$)	-0.1898	I_{xz}
Prod. of Iner. about y-z axes ($kg.m^2$)	0	I_{yz}
Mean Aerodynamic Chord (m)	0.433	$barc$
Wing Span (m)	1.917	b_w
Wing Area (m^2)	0.769	S_{REF}

The aircraft dynamic equations of motion are derived by applying the Boltzmann-Hamel equations for mechanical systems with nonholonomic constraints [Pedro, (1992)]:

$$Q_\mu = \frac{d}{dt}\left(\frac{\partial T^*}{\partial \omega_\mu}\right) - \frac{\partial T^*}{\partial \pi_\mu} + \sum_{r=1}^{k}\sum_{\alpha=1}^{k} \gamma_{\alpha r}^{r} \frac{\partial T*}{\partial \omega_r}\omega_\alpha \tag{1}$$

where Q_μ is the generalized force; T^* is kinetic energy of the system expressed in quasi-coordinate and quasi-velocities; $\gamma_{\alpha r}^{r}$ are the three index Boltzmann multipliers; ω_μ are the Quasi-velocities; π_μ are the Quasi-coordinates; $\mu = 1, 2 \ldots k$ - Degrees of freedom.

Application of Equation 1 to the UAV gives its equations of motion in the body-fixed reference system as:

$$\text{m } \dot{u} = m\,(rv - qw) - mg\sin\theta + T\cos\psi_T\cos\theta_T - \tfrac{1}{2}\rho V_A^2 S$$
$$(C_D\cos\beta\cos\alpha + C_y\sin\beta\cos\alpha - C_L\sin\alpha) \tag{2}$$

$$\text{m } \dot{v} = m\,(pw - ru) + T\sin\psi_T + mg\cos\theta\sin\phi + \tfrac{1}{2}\rho V_A^2 S$$
$$(-C_D\sin\beta + C_y\cos\beta) \tag{3}$$

$$\text{m } \dot{w} = m\,(qu - pv) + mg\cos\theta\cos\phi + T\cos\psi_T\sin_T$$
$$-\frac{1}{2}\rho V_A^2 S\,(C_D\cos\beta\sin\alpha + C_y\sin\beta\sin\alpha + C_L\cos\alpha) \tag{4}$$

and the equations for rotational accelerations are:

$$\text{I}_x\dot{p} = I_{xz}\dot{r} + (I_y - I_z)\,qr + I_{xz}pq - \tfrac{1}{2}\rho V_A^2 Sb(C_l\cos\beta\cos\alpha$$
$$+\ \text{C}_m\sin\beta\cos\alpha - C_n\sin\alpha)$$
$$+I_T\omega_P\,(q\cos\psi_P\sin\theta_T + r\sin\phi_T) \tag{5}$$

$$\text{I}_y\dot{q} = (I_z - I_x)\,rp - I_{xz}\left(r^2 - p^2\right) + \tfrac{1}{2}\rho V_A^2 S\bar{c}(-C_l\sin\beta$$
$$+C_m\cos\beta) - I_P\omega_P\cos\psi_T\,(r\cos\theta_T + p\sin\theta_T) \tag{6}$$

$$\text{I}_z\dot{r} = I_{xz}\dot{p} + (I_x - I_y)\,pq - I_{xz}qr + I_P\omega_P(q\cos\psi_T\cos\theta_T$$
$$\text{- p }\sin\psi_T) - \tfrac{1}{2}\rho V_A^2 Sb(C_l\cos\beta\sin\alpha$$
$$+C_m\sin\beta\sin\alpha + C_n\cos\alpha) \tag{7}$$

where $u, v,$ and w are the aircraft velocity components along $x, y,$ and z-body axes, respectively; $p, q,$ and r are the roll, pitch, and yaw rates respectively; ϕ and θ are the roll and pitch angles respectively; m is the aircraft mass; and g is the acceleration due to gravity. I_x, I_y, and I_z are the vehicle moments of inertia about $x, y,$ and z axes respectively; I_{xz} the vehicle products of inertia, respectively; b is the wing span; c is the mean aerodynamic chord; I_T is the engine moment of inertia; S is the aircraft reference area; and T is the engine thrust. V_A is the velocity vector, α is the angle of attack, β is the angle of sideslip; and ρ is the air density. ψ_T and θ_T are the engine thrust yaw and pitch angles respectively. The aircraft model includes correction terms for the gyroscopic effects of the spinning propeller. The rotational kinematics are:

$$\dot{\phi} = p + \tan\theta\,(q\sin\phi + r\cos\phi) \tag{8}$$
$$\dot{\theta} = q\cos\phi - r\sin\phi \tag{9}$$
$$\dot{\psi} = (q\sin\phi + r\cos\phi)\sec\theta \tag{10}$$

where ψ is the yaw angle. The navigation equations are:

$$\dot{x}_E = u\cos\theta\cos\psi + v(\sin\phi\sin\theta\cos\psi - \cos\phi$$
$$\sin\psi) + w(\cos\phi\sin\theta\cos\psi + \sin\phi\sin\psi) \tag{11}$$

$$\dot{y}_E = u\cos\theta\sin\psi + v(\cos\phi\sin\theta\cos\psi - \cos\phi$$
$$\cos\psi) + w(\cos\phi\sin\theta\sin\psi - \sin\phi\cos\psi) \tag{12}$$
$$\dot{h} = -\dot{z}_E = u\sin\theta - v\sin\phi\cos\theta - w\cos\phi\cos\theta \tag{13}$$

where x_E, y_E, and h are the aircraft position components in the Earth-fixed frame.

The lift (C_L) and drag (C_D) coefficients are functions of the non-dimensional coefficients (whereby the small perturbation assumption is made and only linear terms are considered). The aerodynamic forces and moments coefficients are given by the following relationships:

$$C_D = C_{D_0} + C_{D_{\delta_e}}\delta_e + C_{D_{\delta_r}}\delta_r + \frac{\left(C_L - C_{L_{min}}\right)^2}{\pi e AR} \tag{14}$$

$$\text{C}_y = C_{y_0} + C_{y_\beta}\beta + C_{y_{\delta r}}\delta_r + C_{y_{\delta_a}}\delta_a + C_{y_{\delta_f}}\delta_f + \tfrac{b}{2V_A}$$
$$(C_{y_P}p + C_{y_r}r) \tag{15}$$

$$\text{C}_L = C_{L_0} + C_{L_\alpha}\alpha + C_{L_{\delta_e}}\delta_e + C_{L_{\delta_f}}\delta_f \tag{15}$$