Structured Adaptive Model Inversion Applied To Tracking Aggressive Aircraft Maneuvers

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A model reference adaptive control formulation is presented that uses the dynamical structure of the state space descriptions of a large class of systems. Of particular interest, the formulations enable the imposition of exact kinematic differential equation constraints upon the adaptation process that compensates for model errors and disturbances at the acceleration level. The resulting closed loop error dynamics are shown to be globally asymptotically stable and all signals within the closed loop system are shown to be bounded. The utility of the resulting structured adaptive control formulation is studied by considering the problem of tracking an aggressive aircraft maneuver in the presence of large unknown model errors. The reference maneuver is an aggressive turn in the horizontal plane with the altitude and velocity held constant. Parameter uncertainties are incorporated explicitly in both the aerodynamic stability derivatives and the control influence matrix. A simplified nonlinear model for the equations of motion is considered and the approach is validated against a reference trajectory generated using the same model and a more complex medium-fidelity simulation.

Introduction

The behavior of mechanical and aerospace systems is often well explained by the interaction between aerodynamics, propulsion, structural dynamics, and the characteristics of the sensors and actuators; all of which operate under the laws of kinematics and dynamics. The mathematical structure governing this behavior is highly structured, and is further affected greatly by the choice of coordinates [1,2,3]. Recognizing these facts and exploiting the underlying analytical structure of the differential equations, one can simplify the task of a control design process to come up with more elegant and desirable controllers to perform required tasks. Most mechanical systems that are encountered in practice can be represented as second order dynamics in vector form. Half of the dynamical equations consist of exact kinematic relationships between position and velocity coordinates, while the other half are momentum level relationships of acceleration coordinates to external and control forces. For aircraft dynamics, which have six degrees of freedom (momentum states), we usually construct the dynamical model in a second order vector form. The fact that enforcing exact kinematic relationships reduce the dimensionality of space of parameters being adapted in a given control problem has been recognized and one can find numerous applications of the same in the Structured Model Reference Adaptive Control (SMRAC) formulation related literature [4,5,6]. Furthermore, this approach makes use of the intuitively reasonable principle that one should not use an adaptive approximation of relationships that can be exactly enforced. This formulation explicitly imposes the analytical structure of the kinematic relationships on the mathematical model on a class of actual systems to design robust adaptive controllers. The resulting control systems are simpler and easy to implement because all the learning and adaptation due to uncertainty and model errors is restricted to occur only in a physically admissible subspace of the entire state space. Structured Adaptive Model Inversion (SAMI) is motivated through concepts well embedded in the control literature as SMRAC, Feedback Linearization, and Dynamic Inversion. Firstly the mathematical structure of the set of differential equations governing almost all-dynamical systems is recognized. Dynamic model inversion is then employed to solve for the ideal control law. Since this control law assumes perfect knowledge of the system properties, an adaptive formulation is proposed to account for the imprecise knowledge of the system properties, modeling errors and distur-
bances. It is also recognized that this approach is akin to the classical input-output feedback linearizing technique. It should be pointed out that the classical I/O feedback linearization requires the designer to guarantee a parameter-independent diffeo-morphism that renders the nonlinear system linear in the transformed coordinates. The existence of such a diffeomorphism for an arbitrary choice of outputs is an open question. However if one recognizes the inherent structure of any dynamical system, and chooses as outputs to be observable position coordinates, then a near-trivial parameter independent diffeo-morphism always exists. This is due to the nature of kinematic differential equations, which are typically exact. The diffeomorphism is non-singular so long as the kinematic differential equations themselves are non-singular. So, a judicious choice of position coordinates will always ensure a non-singular diffeo-morphism to guarantee the existence of a feedback linearization. In this paper, we present a complete development of the structured adaptive model inversion formulation. The paper is organized as follows. The control law development using the aforementioned methodology is presented for the aircraft trajectory-tracking problem in detail, which is then followed by a section with example simulations. In the last section, we summarize our conclusions and recommendations.

System Description

In accordance with our design methodology, we shall consider a partitioned class of systems described by

\[ \dot{\sigma} = f(\sigma, \omega) \]  
\[ \omega = g(\sigma, \omega, p) + h(\sigma, \omega, p)u + H(\sigma, \omega) \]  
\[ y = \sigma \]  

Where the state vector \([ \sigma \quad \omega ]^T \in M\), a compact subset of \( \mathbb{R}^{m+n} \); the control input \( u \in U \subset \mathbb{R}^m \); the output \( y \in \sigma \in \mathbb{R}^m \); \( p \in \Omega \subset \mathbb{R}^p \) is a vector of unknown parameters. Here the number of outputs is considered to be equal to the number of inputs, since \( m \) inputs can control \( m \) outputs independently. The function \( f(\sigma, \omega) \) is assumed to be continuously differentiable sufficient number of times on \( M \). In the present context \( \sigma \) is the vector of position coordinates and \( \omega \) is the vector of velocity coordinates. Observe that we do not require \( g(\sigma, \omega, p), h(\sigma, \omega, p) \) and \( H(\sigma, \omega) \) to be sufficiently differentiable, it only suffices if they are continuous. The function \( H(\sigma, \omega) \) is the vector of known non-linearities that would result from inertial and kinematic coupling. Though this depends on terms like the inertia parameters and gravity, we assume them to be known with sufficient accuracy. Further the functions that depend on the unknown parameters can be parameterized linearly in terms of the uncertain parameters. Note that Eq. 1 describes the exact kinematic relationships for a specific choice of position and velocity coordinates while Eq. 2 describes the momentum level dynamics that depend on uncertain force and moment influences.

Problem Formulation

For the system described as above, we seek to derive a control law: \( u = -(GL + M\Psi) \), where \( L, M \) are obtained from some adaptive law to be specified below and \( G, \Psi \) are nonlinear functions involving the states and some design constants that are to specified in later sections. The control law is designed so that the system described in Eq. 1 and Eq. 2 tracks a desired reference trajectory specified by \( \sigma_r(t) \). Furthermore, we prescribe the desired tracking error dynamics to be linear, satisfying the relationship

\[ \dot{e} + Ce + Ke = 0 \]  

Where, \( e = \sigma - \sigma_r \) and \( C > 0, \ K > 0 \) are user specified matrices which effectively shape the tracking error dynamics. The kinematic relationships in Eq. 1 can be differentiated once to obtain the following

\[ \dot{\sigma} = \frac{\partial f}{\partial \sigma} \sigma + \frac{\partial f}{\partial \omega} \omega \]  

Substituting for \( \dot{\omega} \) in Eq. 5, we obtain

\[ \dot{\sigma} = \frac{\partial f}{\partial \sigma} \sigma + \frac{\partial f}{\partial \omega} [g(\sigma, \omega, p) + H(\sigma, \omega)] + \frac{\partial f}{\partial \omega} h(\sigma, \omega, p)u \]  

Substituting for \( \dot{\sigma} \) from Eq. 6 in Eq. 4, we obtain the following

\[ \frac{\partial f}{\partial \sigma} \dot{\sigma} + \frac{\partial f}{\partial \omega} [g(\sigma, \omega, p) + H(\sigma, \omega)] + \frac{\partial f}{\partial \omega} h(\sigma, \omega, p)u - \dot{\sigma} + C(\dot{\sigma} - \dot{\sigma}_r) + K(\sigma - \sigma_r) = 0 \]  

From Eq. 7, if the control law is chosen such that

\[ u = -\left[ \frac{\partial f}{\partial \omega} h(\sigma, \omega, p) \right]^{-1} [\left\{ \frac{\partial f}{\partial \sigma} \dot{\sigma} + \frac{\partial f}{\partial \omega} [g(\sigma, \omega, p) \right. 

\left. + H(\sigma, \omega)] - \dot{\sigma} + C(\dot{\sigma} - \dot{\sigma}_r) + K(\sigma - \sigma_r) \right] \]  

Then one can explicitly enforce the tracking error dynamics to be of the form described in Eq. 4. Now, if \( h(\sigma, \omega, p) \) and \( g(\sigma, \omega, p) \) are parameterized so that the uncertain parameters appear linearly, the control law indeed takes the form \( u = -(GL + M\Psi) \), where the terms \( G \) and \( \Psi \) can be deduced from Eq. 8.

Aircraft Model

For studying the dynamics of the aircraft, its principal axes are chosen as body axes. A simplified model of the F/A 18 like aircraft is considered modeled as follows.

\[
\begin{pmatrix}
\phi \\
\theta \\
\psi
\end{pmatrix} =
\begin{pmatrix}
\frac{p + q \tan \theta \sin \phi + r \tan \theta \cos \phi}{q \cos \phi - r \sin \theta} \\
q \sin \phi \sec \theta + r \cos \phi \sec \theta
\end{pmatrix}
\]  

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\[ \dot{\p} = l_\beta + l_\theta q + l_\tau r + (l_\beta a + l_\tau r) \Delta a \\
+ l_p p - i z q r + l_\delta a \delta a + l_\delta q \delta q \\
\dot{q} = \tilde{m}_a \Delta a + \tilde{m}_q q + i z p r - m_\delta p \beta \\
+ m_\delta (g_0/V) (\cos \theta \cos \phi - \cos \theta_0) + l_\delta q \delta q \\
\dot{r} = n_\beta + n_r + n_p p + n_\delta p \Delta a \\
- i z p q + n_\delta q + l_\delta a \delta a + l_\delta q \delta q \\
\dot{\alpha} = q - p \beta + z_\alpha \Delta a + (g_0/V) (\cos \theta \cos \phi - \cos \theta_0) \\
\dot{\beta} = y_\beta + p (\sin a_0 + \Delta a) - r \cos a_0 \\
+ (g_0/V) \cos \theta \sin \phi \] (10)

Where, \( \sigma = [\phi \ \theta \ \psi]^T \), the Euler angles are the position coordinates, \( \Delta a = a - a_0 \), \( \omega = [p \ q \ r]^T \) are the angular velocity components in the body frame and the control vector \( u = [\delta_\alpha \ \delta_\beta \ \delta_\phi]^T \). Observe though the angle of attack \( \alpha \) and angle of sideslip \( \beta \) appear as part of the momentum level equations in Eq. 10, the controls do not affect these equations. For the controls to appear explicitly in the angle of attack and the angle of sideslip equations, we need to differentiate these equations one more time. Treating the angle of attack and the angle of sideslip as position coordinates helps, but then we would not obtain a parameter independent diffeomorphism which then adds further complexity. This was done in Ref.[7] but in a different way. We simply make the following observation. The angle of attack and the angle of sideslip can be controlled indirectly through the pitch rate \( q_\beta \) and the roll and yaw rates \( p_\beta \) and \( r_\beta \) respectively. Hence we shape the reference commands \( p_\beta \) and \( q_\beta \) respectively. Therefore we note that the open-loop linearized equations in the angle of attack and angle of sideslip channel are stable, which is essentially the zero-dynamics in this case. Observing the equations Eq. 9, Eq. 10 we see that the model can be cast into the structure of equations given in Eq. 1 and Eq. 2. Identifying the various quantities in relation to the system description, we note that \( h(\sigma, \omega, p) \) is a constant matrix. The remaining terms are identified as follows.

\[ f(\sigma, \omega) = \begin{pmatrix} p + q \tan \theta \sin \phi + r \tan \theta \cos \phi \\ q \cos \phi - r \sin \phi \\ q \sin \phi \sec \theta + r \cos \phi \sec \theta \end{pmatrix} \]

\[ \frac{\partial f}{\partial \sigma} = \begin{bmatrix} q \tan \theta \cos \phi & q \sec^2 \theta \sin \phi & 0 \\ -r \tan \theta \sin \phi & -r \sec^2 \theta \cos \phi & 0 \\ -q \sin \phi - r \cos \phi & 0 & 0 \\ q \sec \theta \cos \phi & q \sin \phi \sec \theta \tan \theta & 0 \\ -r \sec \theta \sin \phi & r \cos \phi \sec \theta \tan \theta & 0 \end{bmatrix} \]

\[ \frac{\partial f}{\partial \omega} = \begin{bmatrix} \tan \theta \sin \phi & \tan \theta \cos \phi & 0 \\ 0 & \cos \phi & -\sin \phi \\ 0 & \sin \phi \sec \theta & \cos \phi \sec \theta \end{bmatrix} \]

The vector function \( g(\sigma, \omega, p) \) is further parameterized linearly in terms of the uncertain parameters to give \( g(\sigma, \omega, p) = G(\sigma, \omega) L \), where \( L \) contains the vector of uncertain parameters or as \( AG(\sigma, \omega) \), in which case \( A \) now is a matrix formed using the uncertain parameters. It is evident that the latter representation is clearly an over-parameterization, which would lead to increased dimensionality when the adaptive laws are derived. Clearly, the former parameterization is the minimal parameter representation for the dynamics. Rearranging terms in the control law derived in Eq. 8 for the aircraft model we have the following.

\[ u = -\left[ \frac{\partial f}{\partial \omega} \right]^{-1} \left[ \frac{\partial f}{\partial \sigma} \frac{\partial G(\sigma, \omega) L + H(\sigma, \omega)}{\partial \sigma} \right] \\
- \frac{\partial G(\sigma, \omega) \tilde{L}}{\partial \sigma} + C(\sigma - \sigma_r) + K(\sigma - \sigma_r) \] (11)

Rearranging terms in Eq. 11 further we define,

\[ \tilde{\Psi} = \frac{\partial f}{\partial \sigma} \frac{\partial G(\sigma, \omega) L + \tilde{H}}{\partial \sigma} \]

\[ u = -M^{-1} \left[ \frac{\partial f}{\partial \omega} \right]^{-1} \left[ \frac{\partial f}{\partial \omega} G(\sigma, \omega) L + \tilde{\Psi} \right] \\
= -M^{-1} \left[ G(\sigma, \omega) L + \left[ \frac{\partial f}{\partial \omega} \right]^{-1} \tilde{\Psi} \right] \\
= -M^{-1} [G(\sigma, \omega) L + \tilde{\Psi}] \]

Where, \( \Psi = \left[ \frac{\partial f}{\partial \omega} \right]^{-1} \tilde{\Psi} \) has been used. We have finally derived the form of the control law as desired. The implementation of the control law however requires the calculation of two inverses, \( \left[ \frac{\partial f}{\partial \omega} \right]^{-1} \) and \( M^{-1} \). The inverse of the Jacobian almost always exists as long as the position and velocity coordinates remain in a singularity free volume. In the present case the singularity exists for \( \theta = 90^\circ \). Observing the control influence matrix, we note that the second inverse exists. Further the calculation of the time derivative of \( M^{-1} \) can be avoided at every instant of time and we could adapt for the inverse directly if we recognize the following.

\[ \frac{d}{dt} (MM^{-1}) = 0 \]
\[ M \frac{d}{dt} (M^{-1}) + \dot{M} M^{-1} = 0 \]
\[ \Rightarrow \frac{d}{dt} (M^{-1}) = -M^{-1} \dot{M} M^{-1} \]

Using the control law as derived above, we note the tracking error dynamics takes the form in Eq. 4. However, the implementation of the control law derived above requires the precise knowledge of the parameters in \( L \) and \( M \). We propose the following adaptive laws (derived in the next section) for the case when the parameters are uncertain.

**Structured Adaptive Model Inversion**

Let us assume that the true values of the parameters are represented by \( L^* \) and \( M^* \), that are constant. Further let \( L(t) \) and \( M(t) \) be the estimates of these parameters at every instant of time and the control law is calculated using the "Certainty Equivalence Principle". The tracking error dynamics with the true parameters can be written as follows, after substituting for various terms.

\[ \ddot{e} + C\dot{e} + Ke = G(\sigma, \omega)L + \Psi + Mu \quad (13) \]

The control law in Eq. 11 for the known parameter is obtained by setting the right hand side in Eq. 13 to zero. We will call this the control identity. Adding and subtracting the control identity for the known parameters to the RHS of Eq. 13, we get the following

\[ \ddot{e} + C\dot{e} + Ke = G(\sigma, \omega)\bar{L} + \bar{M}u \quad (14) \]

where \( G(\sigma, \omega)\bar{L} + \Psi + \bar{M}u = 0 \) has been used. Let us now propose a candidate Lyapunov function as follows,

\[ V(e, \dot{e}, \bar{L}, \bar{M}) = \frac{1}{2} e^T \dot{e} + \frac{1}{2} e^T C e + \frac{1}{2} Tr \left[ \bar{L}^T \bar{L}^{-1} \dot{L} \right] + \frac{1}{2} Tr \left[ \bar{M}^T \bar{M}^{-1} \dot{M} \right] \quad (15) \]

Where, \( \Gamma_1, \Gamma_2 \) are user specified symmetric positive definite adaptation gain matrices. Clearly, \( V > 0 \) \( \forall \ e \neq 0, \dot{e} \neq 0, \bar{L} \neq \bar{M} \neq 0 \) and \( V(0, 0, 0, 0) = 0 \).

\[ \dot{V} = \dot{e}^T \dot{e} + \dot{e}^T C e + Ke + \frac{1}{2} Tr \left[ \bar{L}^T \bar{L}^{-1} \dot{L} + \bar{M}^T \bar{M}^{-1} \dot{M} \right] \]

\[ = -\dot{e}^T C \dot{e} + \dot{e}^T G(\sigma, \omega)\bar{L} + e^T \bar{M} u + Tr \left[ \bar{L}^T \bar{L}^{-1} \dot{L} + \bar{M}^T \bar{M}^{-1} \dot{M} \right] \quad (16) \]

Using the matrix trace identities and rearranging terms in Eq. 16, we obtain the following.

\[ \dot{V} = -\dot{e}^T C \dot{e} \quad \leq 0 \]

\[ \dot{\bar{L}} = \bar{L} = \Gamma_1 G(\sigma, \omega)^T \dot{e} \quad (17) \]

\[ \dot{\bar{M}} = \bar{M} = \Gamma_2 \dot{e} u^T \quad (18) \]

Eqs. 17 & 18 summarize the adaptive laws to be used along with the control law to achieve the desired tracking objectives. The adaptive laws as chosen above ensure that the tracking errors and the parameter errors are bounded. To prove asymptotic stability of the tracking error dynamics in Eq. 14 with the control law in Eq. 11 and the adaptation laws in Eqs. 17 & 18, we follow the same approach as outlined in Ref. [8].

**Reference Trajectory and Results**

A reference trajectory is generated using the model described by Eqs. 9 & 10. The actual simulation assumes that all parameters are in error upto 30% of their nominal values. This error is injected "carefully", so that the net result is to destroy some of the open loop stability properties and there is a loss in control effectiveness. The reference maneuver is an aggressive turn maneuver in the horizontal plane with the velocity and altitude held constant. The maneuver starts with a roll-into-the-turn phase, where a desired bank angle is achieved which is then followed by a 180 degrees heading change at constant bank angle. The final segment corresponds to a roll-out-of-the-turn thereby achieving a steady, straight and level flight. Fig(1) shows the reference trajectory states.

**Medium Fidelity Simulation Results**

The aircraft simulation that was used to generate the following results is a high performance aircraft with 2 engines, 2 stabilators, 2 ailerons, 2 rudders, 2 leading edge flaps, and 2 trailing edge flaps. To maintain commonality with the previous point simulation, the flaps were scheduled with flight condition, and were not otherwise used. The other surfaces were ganged into three controls of collective stabilator and rudder and differential aileron. Velocity was controlled by a separated PID auto-throttle that attempts to maintain constant velocity. The simulation uses the standard equations of motion and kinematic relations found in a variety of standard references [9].

\[ \dot{u} = \frac{F_x + F_x}{m} - g \sin \Theta \]
\[ + ru - qw \]
\[ \dot{v} = \frac{F_y + F_y}{m} + g \cos \Theta \sin \Phi \]
\[ + pv - ru \]
\[ \dot{w} = \frac{F_z + F_z}{m} + g \cos \Theta \cos \Phi \]
\[ + qu - pv \]

\[ \dot{p}(I_{xx} I_{zz} - I_{xz}^2) = I_{zz}(l_a + l_T - qr(I_{zz} - I_{yy})) + pl_{xz} + I_{xz}(n_a + n_T) + pq(I_{xx} - I_{yy}) - qr I_{xz} \]

\[ \dot{q}(I_{yy}) = m_A + m_T + (r^2 - p^2)(I_{xz}) + pr(I_{zz} - I_{xx}) \]

\[ \dot{r}(I_{xx} I_{zz} - I_{xz}^2) = I_{xx}(n_a + n_T) + qr(I_{xx} - I_{yy}) - qr I_{xz} + I_{xz}(l_a + l_T) \]
The components of the aerodynamic forces $(F_{XA}, F_{YA}, F_{ZA})$ and moments $(l_A, m_A, n_A)$ are calculated from table look-ups. Gross thrust, $T$, is calculated from the following equation:

$$
T = \left[1 + a_1\alpha + a_2\alpha^2\right] F_T(h, M, P_{LT}) [kP_{LT} + c]
$$

where $a_1, a_2, c$, and $k$ are constants, $F_T$ is calculated from a table look-up, and $P_{LT}$ is lagged throttle position. The throttle model is a first order linear system with a variable time constant and variable rate limit based on the value of $P_{LT}$. The actuator models are 2nd order linear systems (except for stabilators, which are fourth order) with rate and position limits. There were no disturbances and sensors were considered to be perfect. It was necessary to make several modifications from the above design to get acceptable performance with this higher fidelity simulation. Note that there was minimal time spent tuning this controller so it may not represent the best that could be done with this approach. Furthermore, the controller was only flight tested at 1 flight condition for 1 maneuver, so the choice of parameters may not be very good for other flight conditions or classes of inputs. The modifications were:

1. The gains were reduced to $C = \text{diag}[7.5 5 5]$ and $K = \text{diag}[3.6 2 2]$. Despite the lack of noise in the simulation, it was important to ensure gains were in a practical range to use with realistic sensors.

2. $G$ was multiplied by the respective a priori estimated values of the stability parameters and updated during the simulation as a function of angle-of-attack and Mach number. The initial value of $L$ was then set to a vector of ones. This solved 2 problems. First, it acted to normalize the adaptation of the values in $L$. Secondly, it allowed the controller to take into account knowledge of how the stability parameters change as the flight condition changes. This was particularly important for the maneuver used in this paper since $C_{ma}$ and $C_{n,b}$ change considerably over the course of the maneuver.

3. The adaptation equation for $M$ was multiplied by a factor of 5 to get fast enough adaptation.

4. The reference values for $q$ and $r$ were modified over the maneuver to take into account angle-of-attack and sideslip feedback as well as some axis de-coupling terms. This was a brute force fix to ensure angle-of-attack and sideslip remained within acceptable bounds. Otherwise there were fairly large sideslip excursions, particularly at high roll angles when the system would otherwise attempt to control pitch angle with $r$. Similarly, angle-of-attack would otherwise exceed the controllable angle-of-attack range during periods of high pitch rate early in the maneuver.

5. The trim angle of attack was updated in Eq. 10 as a function of flight condition.

6. Adaptation was turned off during periods when the actuators were saturated.

Fig. (2) show the actual and desired values of the states and controls for the same aggressive high-g maneuver as the previous section starting at a flight condition of .9M and 10K ft. As can be seen, the controller tracks the desired pitch, roll, and yaw angles fairly well to the extent that is possible given the limitations of this more complex model with a significant change in velocity during the maneuver. There are some steady-state errors, but those could probably be dealt with using integrated error.

To show the beneficial effects of the adaptation laws, Fig. (3) show the same maneuver without any adaptation. The controller is now essentially a fairly simple dynamic inversion controller. As can be seen, there are some significant degradations of performance, particularly in longitudinal and directional control.

Fig. (4) show the same maneuver with a simultaneous lost right stabilator, aileron, and rudder. This was simulated by driving the surfaces to the position at which they have no effect on the simulation. The adaptation in this case is fast enough to keep the aircraft from departing, and the tracking results are quite reasonable given the severity of the failure. For this case, if the adaptation is turned off, the aircraft departs after only a few seconds when trying to perform this maneuver.

Conclusions

A structured model adaptive inversion based controller has been designed to track aggressive aircraft maneuvers. The control law together with the adaptive laws is shown to work excellently well in the presence of bounded disturbances and significant parameter errors. However, this was only demonstrated for a single maneuver at a single flight condition, so much more testing would be needed before making any broader conclusions about the utility of this control law.

References


Fig. (1): Reference Trajectory States and Control Histories.
Fig. (2): State and Control histories with SAMI on the Medium Fidelity Simulation
Fig. (3): State and Control histories without SAMI on the Medium Fidelity Simulation
Fig. (4): State and Control Histories with SAMI in the presence of actuator failure on the Medium Fidelity Simulation