Available online at www.sciencedirect.com

ScienceDirect

IFAC PapersOnLine 50-2 (2017) 187–192

Design and Flight Testing of a Gain-Scheduled Autopilot Based on Reduced Model Constraints

Paulin Kantue *

* *Principal Flight Control Systems Engineer, Denel Dynamics, Centurion, 0046 South Africa (e-mail: paulin.kantu@deneldynamics.co.za).*

Abstract: The purpose of this paper focuses on the design, evaluation and flight testing of a gain-scheduled autopilot for a missile system based on reduced modelling constraints. The controller synthesis based on frequency response analysis theory formed the basis for the implementation of gain-scheduled controllers. During the controller synthesis process, the design constraints were based on a reduced missile model and analytically computed controller initial gains which improved the controller optimization phase while satisfying the robustness and performance requirements across the envisioned flight envelope. This process was shown to produce successful results in both a nonlinear desktop simulation and a flight testing campaign.

Keywords: Flight control, Missiles, Aerospace control, Robustness, Computer-aided control system design, Control system synthesis

1. INTRODUCTION

The design of a missile autopilot is a well-known problem with the challenging task of controlling a highly complex nonlinear system with non-minimum phase and wide parameter variation dynamics across the entire flight envelope. The main of objective of a homing missile flight control system or autopilot is to accurately track acceleration commands issued by the guidance algorithms as quickly as possible. For the modern missile, actuator bandwidth, structural modes, aerodynamic uncertainty, microcomputer processing power and sensor bandwidth are some of the constraints to consider during the design process. Given the nonlinear nature of the problem, the gain scheduling technique has been used as a widespread approach for associating a linear system with a nonlinear one using methods such as the series expansion linearisation theory Leith and Leithead (2000). Moreover, it has been demonstrated to be successful as a nonlinear control technique for missile autopilot design Yuan et al. (2016); Saussié et al. (2008); Buschek (1999).

The traditional gain scheduling technique is implemented with the design of flight controllers around several linearised models at a attained equilibrium point, then using a scheduling algorithm (such as linear interpolation) to develop a robust controller throughout the envisioned flight envelope. To deal with the issue of guaranteed closed-loop stability and performance, various modern control techniques have been employed for missile autopilot design Al-Sunni and Lewis (1993); Shamma and Cloutier (1993); Calise et al. (2000); Sharma et al. (2006); Rotondo (2016). However gain-scheduling remains a preferred method given it's simplicity and ease of implementation for practical problems Theodoulis and Duc (2009). Nesline and Zarchan (1984) also suggested that traditional robustness techniques, such frequency response analysis, should be used with modern control methods to ensure the system does not go unstable when implemented and tested. Combining the controller synthesis based on reduced aerodynamic model and analytically computed controller gains based on Zarchan (2002) as constraints, the gain-scheduling routine can be further simplified while meeting the robustness and performance requirements. This paper focuses on the implementation and evaluation of such a method.

2. MISSILE MODEL

A tail-controlled, agile air-to-ground missile with an axisymmetric configuration is used. This allowed the use of a skid-to-turn steering policy, which makes no clear distinction between longitudinal and lateral-directional control as is the case with bank-to-turn missiles. Given these decoupled dynamics, this allowed the use of one autopilot that is identical for the longitudinal and the lateral plane (lateral autopilot) and a separate roll autopilot that provides attitude control.

The airframe has four control fins (Fin_1 to Fin_4) which, through the use of a mixing algorithm can produce roll, pitch and yaw command on the airframe. Fin_4 is indexed as the fin on the top starboard side of the airframe (top-right fin when viewing the airframe from the rear) with fin numbering in a clockwise fashion. The control mixing algorithm to convert commands to individual fin deflection is described below:

* Manager of the Strategic Engineering Group. PhD candidate at the University of the Witwatersrand.

10.1016/j.ifacol.2017.12.034

$$\begin{aligned} Fin_1 &= -\delta_{roll} - \delta_{pitch} - \delta_{yaw} \\ Fin_2 &= -\delta_{roll} + \delta_{pitch} - \delta_{yaw} \\ Fin_3 &= -\delta_{roll} + \delta_{pitch} + \delta_{yaw} \\ Fin_4 &= -\delta_{roll} - \delta_{pitch} + \delta_{yaw} \end{aligned} \quad (1)$$

Where δ_{roll}, δ_{pitch} and δ_{yaw} corresponds to roll, pitch and yaw commands respectively coming from the flight control module.

2.1 Model Trim

A trim point is defined as a point in the parameter space of a dynamic system upon which such a system is at steady state. For most airframes, steady state is achieved when the total moment on the airframe is close to zero. Specifications for trim is done through configuring the model inputs, states and outputs. The Mach number $Mach$, altitude z_E and the control deflection δ_{pitch} are the only known non-zero specifications during the operating point search. Given that forces and moments aerodynamic coefficients are independent of air density but only a function of Mach number M, aerodynamic incidence angle σ and aerodynamic roll angle λ, at trim, the following equation holds true. See Nesline and Nesline (1984)

$$C_{m_\alpha}[\sigma, \lambda, M] + C_{m_\delta}[\sigma, \lambda, M] = 0 \quad (2)$$

C_{m_α} and C_{m_δ} are the pitching moment coefficient due angle of attack and control deflection respectively.

The total pitching moment below the value $1e-3$ is used to ensure the trim process is valid and the trim results can be stored and used for linearisation. Based on the trim process, the normal force coefficient at trim CN_{trim} can be calculated.

2.2 Model Linearisation

Linearisation involves creating a linear approximation of a non-linear system that is only valid in a small region around the computed trim point. The Linearisation process made use of the trim results described in the previous section. The computed trim points were used to develop linear-time-invariant (LTI) state space matrices. The certain parameters within these state space matrices were extracted which formed part of a reduced linearised airframe model used for the autopilot design process as described below. Their relationship to dimensionless aerodynamic coefficients is defined as follows:

$$M_\alpha = \frac{1}{2}\rho V_m^2 S_{ref} L_{ref}\left(\frac{1}{I_{yy}}\right) C_{m_\alpha} \quad (3)$$

$$M_\delta = \frac{1}{2}\rho V_m^2 S_{ref} L_{ref}\left(\frac{1}{I_{yy}}\right) C_{m_\delta} \quad (4)$$

$$M_q = \frac{1}{2}\rho V_m^2 S_{ref} L_{ref}\left(\frac{1}{I_{yy}}\right)\left(\frac{L_{ref}}{2V_m}\right) C_{m_q} \quad (5)$$

$$Z_\alpha = \frac{1}{2}\rho V_m^2 S_{ref}\left(\frac{1}{mV_m}\right) C_{N_\alpha} \quad (6)$$

$$Z_\delta = -\frac{1}{2}\rho V_m^2 S_{ref}\left(\frac{1}{mV_m}\right) C_{N_\delta} \quad (7)$$

where M_α, M_δ and M_q is the change in angular acceleration due to α, δ and q respectively. Z_α and Z_δ which are the change in normal acceleration due to α and δ. is the airframe reference area, is airframe reference length, ρ is the air density, V_m is the trim speed Air speed, C_{m_α} is pitching moment coefficient due to angle of attack, C_{m_δ} is Pitching moment coefficient due to control deflection, C_{m_q} is pitching moment coefficient due to pitch rate, q is Airframe pitch rate, I_{yy} is pitching moment of inertia, C_{N_α} is normal force coefficient due to angle of attack, C_{N_δ} is normal force coefficient due to control deflection. See Nesline and Nesline (1984); Zarchan (2002).

The roll dynamics parameters due to roll rate L_p and aileron control L_δ has also been determined. Their relationship to dimensionless aerodynamic coefficients is defined as follows:

$$L_\delta = \frac{1}{2}\rho V_m^2 S_{ref} L_{ref}\left(\frac{1}{I_{xx}}\right) C_{l_\delta} \quad (8)$$

$$L_p = \frac{1}{2}\rho V_m^2 S_{ref} L_{ref}\left(\frac{1}{I_{xx}}\right)\left(\frac{L_{ref}}{2V_m}\right) C_{l_p} \quad (9)$$

where C_{l_δ} is rolling moment coefficient due to control deflection, C_{l_q} is rolling moment coefficient due to roll rate. The airframe natural frequency ω_{AF}, damping ratio ζ_{AF} and transfer function zero ω_z can be computed based on the above parameters:

$$\omega_{AF} = \sqrt{-M_\alpha} \quad (10)$$

$$\zeta_{AF} = -\frac{(M_q + Z_\alpha)}{2\omega_{AF}} \quad (11)$$

$$\omega_z = \frac{M_\alpha Z_\delta - Z_\alpha M_\delta}{Z_\delta} \quad (12)$$

The above parameters extracted from the state space model, constitute the reduced aerodynamic transfer functions describing airframe normal acceleration due to control deflection $\frac{N_L}{\delta}$ and airframe pitch rate due control deflection $\frac{q}{\delta}$ can be derived. See Nesline and Nesline (1984):

$$\frac{N_L}{\delta} = K_1\left(1 - \frac{s^2}{\omega_z^2}\right) / \left(1 + \frac{2\zeta_{AF}}{\omega_{AF} s} + \frac{s^2}{\omega_{AF}^2}\right) \quad (13)$$

where

$$K_1 = \frac{-V_M\left[M_\alpha Z_\delta - Z_\alpha M_\delta\right]}{M_\alpha} \quad (14)$$

$$\frac{q}{\delta} = K_3\left(1 + T_\alpha s\right) / \left(1 + \frac{2\zeta_{AF}}{\omega_{AF} s} + \frac{s^2}{\omega_{AF}^2}\right) \quad (15)$$

where

$$K_3 = \frac{K_1}{V_M} \quad (16)$$

and

$$T_\alpha = \frac{M_\delta}{M_\alpha Z_\delta - Z_\alpha M_\delta} \quad (17)$$