Delft University of Technology

Practical assignment AE4-301P: Exercise Automatic Flight Control System Design

Design of autopilot systems for the General Dynamics F-16 model

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Chapter 1

Introduction

1.1 Introduction

The practical Automatic Flight Control System Design, code AE4-301P, is an additional practical which follows up the course Automatic Flight Control System Design, code AE4-301. The objective of this practical is to become familiar with classical flight controllers and their design, and to gain insight in handling qualities of open-loop and controlled aircraft. This exercise is a homework assignment to be solved using Matlab and Simulink in teams of up to three students.

1.2 Goal of the practical

The goal of this practical is to design part of a flight control system using classical control theory as is covered in the lectures on Automatic Flight Control System Design (AE4301).

1.3 Background information

The practical assignment is set up in such a way that it must be possible to solve the complete assignment by means of the material presented in the assignment and by means of the lecture slides and the additional information on Brightspace. However, if you would like to collect some additional background information, this can be found in ref. [1], [2], [3], [4], [5], [6] and [7].

1.4 Reporting

A written report must be delivered after completion of the practical assignment. Please make sure to explain the chosen procedure as well as the numerical results, but don't make your report too detailed. Matlab code used for answering the questions must be included in a separate zip-file. Make sure that all figures have clear and readable labels and are preferably *vector graphics*. For scientific reporting, the use of LaTeX is preferred, but you are allowed to use other programs, as long as you convert everything to a PDF file in the end. **The deadline for handing in the practical report is: 17:00, Sunday February 9th, 2025.** All reports and code must be submitted through Brightspace. Submission instructions can be found under the Assignments section of the AE4301P Brightspace page.

Chapter 2

F-16 model

The aircraft model to be used for this practical is the General Dynamics F-16, of which a picture and a threeview can be found in fig. 2.1 and 2.2.

Figure 2.1: A pair of General Dynamics F-16's in the Wild Weasel role, armed with pairs of AMRAAMs, Sidewinders and HARM anti-radar missiles together with a pair of additional underwing fuel tanks and a jamming pod on the centerline.

This non-linear model is obtained from University of Minnesota and it simulates the dynamics of the real aircraft. It consists of two models as described by ref. [7] and ref. [2]. From this point forward the model described by ref. [7] will be referred to as the low fidelity model and the model described in ref. [2] will be referred to as the high fidelity model. The main difference between both models is the use of different aerodynamic data and in the low fidelity model, a complete decoupling between longitudinal and lateral equations of motions is applied. Finally, the aerodynamic properties of the slats control surfaces at the wing leading edges are included in the high fidelity model only. More detailed information concerning the model structure and an introduction in the use of this airplane model can be found in ref. [6]. It is highly recommended to study the manual in ref. [6] first before proceeding with the practical assignment, in order to understand the working principle of the program and to be able to use the model properly. However, note that this manual is several years old and may not reflect the current MATLAB code accurately, as the scripts have been adapted to newer versions of MATLAB, overall making it more intuitive to work with.

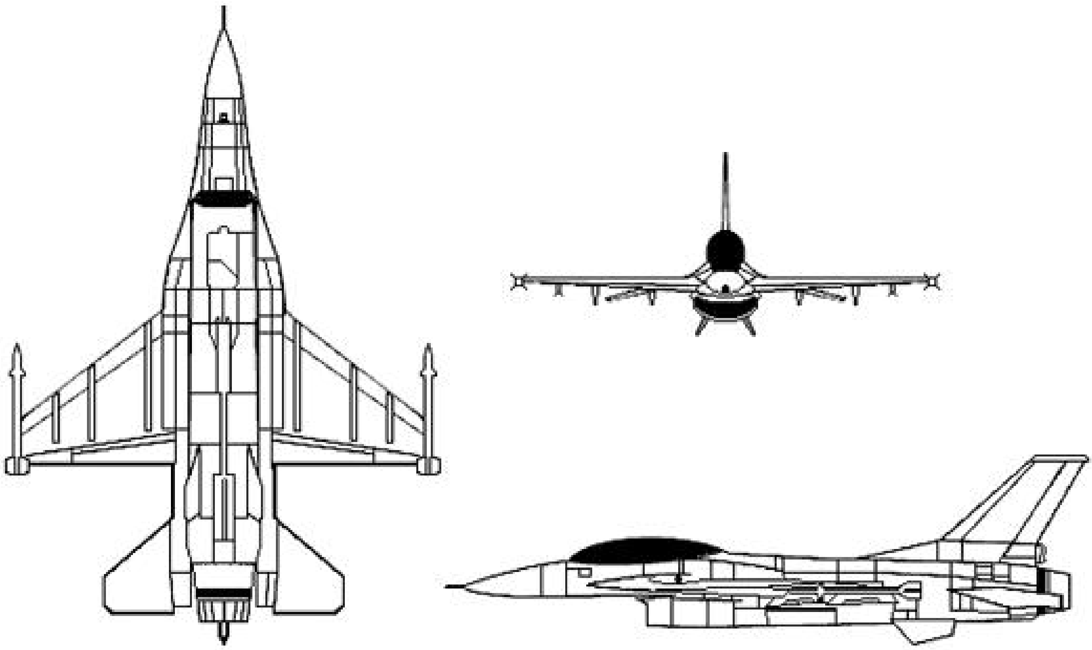


Figure 2.2: Threeview of the General Dynamics F-16 Fighting Falcon

Chapter 3

Assignment set up

Roughly speaking, the set up of the assignment is as follows:

1. choice of flight condition
2. trim and linearization
3. open loop analysis
4. design of a pitch rate command system satisfying CAP/Gibson Mil-specs
5. design of an integrated flight path/speed command system
6. reporting

The first three topics serve as a preparatory part of the assignment. During the elaboration of these topics, the material is prepared and some practice is built up which both serve as a basis for the subsequent topics. After trimming and linearizing the non-linear model for the chosen flight condition, one obtains the state space model which will serve as the basis to design some flight control system components, to be discussed later. The open loop analysis investigates the inherent behavior of the aircraft. In a real world flight control system, this analysis has to be performed for **all** flight conditions in the flight envelope and the results obtained by this analysis have to be compared with the flying and handling quality requirements, in this case MIL-F-8587C, and if any requirement is violated a stability augmentation system (SAS) has to be designed such that the modified behavior complies with all requirements. However, this part of the job has been skipped here. Instead, the second part consists of the remaining two topics where some components of the flight control system have to be designed. The requirements the closed loop system has to comply with (considerably reduced with reference to the MIL-F-8587C requirements) are mentioned in each topic. First a pitch rate command system needs to be designed. This type of command system is used for tracking tasks at low velocities. (For high velocity tracking tasks, a g-command is commonly used instead.) Finally, an automatic VOR controller is designed.

In the following pages, a detailed elaboration of the assignment set up will be given.

Chapter 4

Choice of flight condition

In this practical, you will make use of one individual flight condition. **Select your flight condition based on the rules in table** 4.1 and use it throughout the report, unless is stated otherwise. Mind that in real life, this design needs to be performed in each flight condition and thereafter gain scheduling needs to be performed over all the results, such that the complete flight envelope region is covered in which the flight control system is operational.

- altitude levels: 10000ft, 20000ft, 30000ft, 40000ft
- velocity levels: 300ft/s, 600ft/s, 900ft/s

4.1 Individual flight condition for accelerometer position analysis

For the task of the accelerometer position analysis in section 5.1, the following flight condition is defined for everybody:

- altitude level: 15000ft
- velocity level: 500ft/s

Last digit student number	First letter surname	Altitude	Velocity
0-2	a-h	$10000ft$	$\mathbf{350}ft/s$
	i-p	$10000ft$	$600ft/s$
	q-z	$10000ft$	$900ft/s$
3-5	a-h	$20000ft$	$300ft/s$
	i-p	$20000ft$	$600ft/s$
	q-z	$20000ft$	$900ft/s$
6-7	a-h	$30000ft$	$300ft/s$
	i-p	$30000ft$	$600ft/s$
	q-z	$30000ft$	$900ft/s$
8-9	a-h	$40000ft$	$300ft/s$
	i-p	$40000ft$	$600ft/s$
	q-z	$40000ft$	$900ft/s$

Table 4.1: Rules for selecting your flight condition. If you work together with somebody else, the rules apply to the person whose name appears first on the cover (the order should be alphabetical w.r.t. surname).

Chapter 5

Trim and linearization

First the F-16 model needs to be trimmed and linearized in the two flight conditions from chapter 4. The trimmed LTI model for the individual flight condition serves for the analysis concerning the influence of an accelerometer position in section 5.1. The other trimmed LTI-model for the self-chosen flight condition will serve as the basis in the remainder of the practical where the analysis and command system will be based upon.

Explain why a trim procedure is absolutely necessary before a linearization procedure can be started, in order to obtain a meaningful result.

Trim and linearize the nonlinear F-16 model in the two flight conditions using the m-file `findf16dynamics`. After completing the linearization procedure for the flight conditions (using high and low fidelity models), give the orders of the results obtained after the iterations of the cost function. These results serve as a measure of the achieved accuracy. Give comments on the reliability of the results, and choose another flight condition if necessary, i.e. if the accuracy is insufficient.

From this point onward, only the low fidelity model needs to be taken into account.

5.1 Influence of an accelerometer position

In a fighter aircraft, if an accelerometer is placed close to the pilot's station, aligned along the body Z_b-axis, and used as the feedback sensor for control of the elevator, the pilot has precise control over their Z_b-axis g-load during high-g maneuvers. If $1g$ is subtracted from the accelerometer output, the control system will hold the aircraft approximately in level flight with no control input from the pilot. If the pilot blacks out from the g-load, and relaxes any force on the control stick, the aircraft will return to $1g$ flight. Other useful features of this system are that the accelerometer output contains a component proportional to angle of attack α and can be used to stabilize an unstable short-period mode, and the accelerometer is an internal sensor that is less noisy and more reliable than an angle of attack sensor. The normal acceleration a_n at a point P, fixed in the aircraft body, is defined to be the component of acceleration at P in the negative Z_b direction of the body axes. The contributing component of the gravity acceleration g can be expressed in terms of pitch and roll angles (ignoring the oblateness of the earth). The accelerometer output f_n is then proportional to the specific force:

$$f_n = a_n + |g| \cdot \cos\theta \cdot \cos\phi. \tag{5.1}$$

If the measurement is expressed in g units, the ratio $|g|/g_D$ (g_D is the standard gravity $9.80665 m/s^2$ or the local value) is very close to unity near to the earth's surface, so that:

$$f_n \approx a_n + \cos\theta \cdot \cos\phi \quad [g \text{ units}]\,. \tag{5.2}$$

In level flight, at small pitch attitude angles, the feedback signal for the control system is:

$$(f_n - 1) \approx a_n \quad [g \text{ units}]\,. \tag{5.3}$$

This normal acceleration is approximately zero in steady level flight; it is often called the "incremental" normal acceleration, and given the symbol n_z when in g units. Note that the component of acceleration along the lift axis, in steady-state flight with α, β and ϕ small, can be written in terms of load factor as $(n - \cos\theta)$ g-units. At small angle of attack α the lift direction is nearly coincident with the body negative Z_b-axis, and,

$$a_n \approx n - \cos\theta \quad [g \text{ units}]\,, \tag{5.4}$$

or

$$a_n \approx n - 1 \quad [g \text{ units}]\,. \tag{5.5}$$

Therefore, the accelerometer measurement is an approximate measurement of load factor, under the above conditions.

If the accelerometer is on the body X_b-axis, at a distance x_a forward of the aircraft cg, and the aircraft is not rolling and yawing, the transport acceleration at that point is,

$$a_n = \frac{-\left(a_z - \dot{q}x_a\right)}{g_D} = n_z + \frac{\dot{q}x_a}{g_D} \quad [g \text{ units}]\,, \tag{5.6}$$

where $a_z[m/s^2]$ is the vertical acceleration in the center of gravity and $n_z = -\frac{a_z}{g_D}$. If this equation is included in the nonlinear aircraft model then, in steady level-flight, the normal acceleration is close to zero and numerical linearization will yield a linear equation for a_n. A linear equation can also be obtained algebraically by finding the increment in the aerodynamic and thrust forces due to perturbations in the state and control variables, and this involves the Z-derivatives. However, this is beyond the scope of this practical.

The following steps are required for the individual flight condition trimmed LTI model (h = 15000 ft and V = 500 ft/s):

1. Include equation 5.6 in the nonlinear aircraft model in the Simulink file `lin_f16block` , by generating an additional output making use of the states coming out of the c-code block. The Simulink file `lin_f16block` is linearized in `findf16dynamics` by means of the `linmod` operation. *Hint 1: mind that the Matlab command 'linmod' cannot handle the simulink block* $\frac{dy}{dt}$. *Hint 2: mind that* n_z *is one of the available states.*
2. Trim and linearize this model numerically in the selected flight condition and you will notice the consequence of the additional output on the state space matrices.
3. Give the linearized output equation ($y = Cx + Du$) for normal acceleration a_n at the cg ($x_a = 0$).
4. List the states the normal acceleration a_n depends on.
5. Determine the elevator-to-normal-acceleration transfer function.
6. Draw the normal acceleration response to a negative step elevator command (aircraft nose up), focus on the initial response during the first few seconds and adapt the time step in order to obtain a smooth time response.
7. Which transfer function component forces the step response to be initially in the opposite direction of the reference signal?
8. Give a physical explanation of this non-minimum-phase behaviour.
9. Calculate and analyse the transfer function zero values for different accelerometer positions: $x_a = 0ft$, $x_a = 5ft$, $x_a = 5.9ft$, $x_a = 6ft$, $x_a = 7ft$ and $x_a = 15ft$. Draw and compare the initial normal acceleration responses to a negative step elevator command (aircraft nose up) for the different accelerometer positions.
10. At what position is the "instantaneous center of rotation" located? Explain your answer.
11. Where should the pilot's station preferably be located: before, in or after the instantaneous center of rotation? Explain your answer.
12. Explain why it is important to place the accelerometer close to a node of the most important fuselage bending mode.

Chapter 6

Open loop analysis

The open loop analysis consists of two major parts. First, the obtained LTI state space model from the previous chapter must be reduced to the two familiar 4-state decoupled systems. Thereafter, the motion characteristics of the open loop model must be calculated. These characteristics show the inherent flying qualities of the aircraft.

Concretely, the matrices can be reduced to the following states of interest:

- longitudinal states of interest: V_t, α, θ, q;
- lateral states of interest: β, ϕ, p, r

Mind that that an ordinary reduction procedure (by selecting the appropriate rows and columns in the state space model matrices) results in input matrix $B = 0$. This is because the state space system includes actuator dynamics, and there is no direct connection between inputs and states. Therefore, an alternative approach is necessary, which will be shown here for the longitudinal model. The procedure for the lateral model is analogous.

First of all, reduce the linearized state space model for the flight condition chosen in chapter 4, but with the actuator dynamics included. This means that you have to find a 6×6 matrix A_{OL} and a 6×2 matrix B_{OL}, for the LTI model containing the following states: velocity v_T, angle of attack α, pitch attitude angle θ, pitch rate q, elevator deflection δ_e and throttle setting δ_t and as inputs the electrical signals for the elevator actuator u_e and for the throttle u_t. So far, this can be done by the ordinary reduction procedure, which is the selection of the appropriate rows and columns in the complete state space model matrices.

Subsequently, remove the actuator and engine dynamics out of this reduced model. This can be done according to the following principle, also described on pages 289-293 of ref. [7]. First consider that the servo transfer function, $H_{servo} = \frac{\delta_e(s)}{u_e(s)} = \frac{a}{s+a}$, can be written as a differential equation in the time domain as: $\dot{\delta}_e = -a\delta_e + au_e$. This equation can be recognized in the state space matrices:

$$\dot{x} = \left[\begin{array}{cccc|c} & & & & \\ & A_{a/c} & & & B_{a/c} \\ & & & & \\ \hline 0 & 0 & 0 & 0 & -a \end{array}\right] \cdot \left[\begin{array}{c} v_T \\ \alpha \\ \theta \\ q \\ \hline \delta_e \end{array}\right] + \left[\begin{array}{c} 0 \\ 0 \\ 0 \\ 0 \\ \hline a \end{array}\right] \cdot u_e, \tag{6.1}$$

where the additional state δ_e is the actual elevator deflection due to the command u_e.

By means of this principle, the LTI model between the elevator deflection δ_e and the states can be deduced from the LTI model between the actuator input u_{el} and those same states. It is fairly straightforward to verify the applicability of this principle by recalculating the model with actuator dynamics by putting the model without actuator dynamics, obtained here, and the actuator dynamics themselves, with $a = 20.2$, in series. Mind that there exist more non-unique state space models to represent the same dynamic system, but its transfer functions as well as the eigenvalues of its state matrix A are unique.

The state space matrices $A_{a/c}$, $B_{a/c}$, $C_{a/c}$ and $D_{a/c}$ obtained above will be used in section 6.1 and chapter 7.

6.1 Calculation of the inherent motion characteristics

In order to analyze the inherent behavior of the F-16 model, the following motion characteristics need to be calculated for the flight condition concerned.

- For periodic eigenmotions, namely short period, phugoid and Dutch roll:
  natural frequency ω_n, damping ratio ζ, period P and time to damp to half amplitude $T_{1/2}$.
- For aperiodic eigenmotions, namely aperiodic roll and spiral:
  natural frequency ω_n, time constant τ and time to damp to half amplitude $T_{1/2}$.

Plot the different eigenmotion time responses to support the numbers calculated above.

Chapter 7

Design of a pitch rate command system satisfying CAP/Gibson Mil-specs

In this section a pitch rate command system has to be designed. A pitch rate command system is very effective for tracking tasks at low velocities.

The pitch rate controller has to fulfill the following requirements, applied on the military specifications:

- the CAP (Control Anticipation Parameter) criterion
- the Gibson criterion

Both requirements will be elaborated in depth below, in order to define accurate requirements.

7.1 CAP criterion

For highly augmented airplanes, like the General Dynamics F-16, a requirement involving the Control Anticipation Parameter (CAP) has been defined and has in fact replaced the conventional short period undamped natural frequency and damping ratio requirements. This new requirement defines that the airplane must stay within a minimum and maximum range of values of the so-called CAP over a range of allowable short period damping ratios.

Physically, the Control Anticipation Parameter is a measure of maneuverability. If the CAP requirement is satisfied, then one can say that the aircraft's "nose follows the stick". This parameter can be defined physically as the ratio of the initial pitch acceleration over the steady state load factor, alternatively expressed in an equation:

$$CAP = \frac{\dot{q}\,(t=0)}{n_z\,(t=\infty)}. \tag{7.1}$$

The equation to estimate the control anticipation parameter (CAP), based upon the available data, is defined as follows:

$$CAP = \frac{\omega_{n_{sp}}^2}{n_\alpha} \tag{7.2}$$

where:

$\omega_{n_{sp}}$ is the undamped natural frequency of the short period mode

$n_\alpha = \dfrac{\partial n}{\partial \alpha}$ which is also referred to as the gust- or load-factor-sensitivity of an airplane

Because of the available data, another more preferable way to calculate the CAP is as follows:

$$CAP = \frac{g\omega_{n_{sp}}^2 T_{\theta_2}}{V} \tag{7.3}$$

where the parameter T_{θ_2} is right away available from the short period reduced pitch rate transfer function:

$$\frac{q\left(s\right)}{\delta_{el}\left(s\right)} = \frac{k_q\left(1 + T_{\theta_2}s\right)}{s^2 + 2\zeta_{sp}\omega_{n_{sp}}s + \omega_{n_{sp}}^2}. \tag{7.4}$$

The requirements concerning Control Anticipation Parameter and short period damping ratio are illustrated by the allowable regions shown in fig. 7.1.

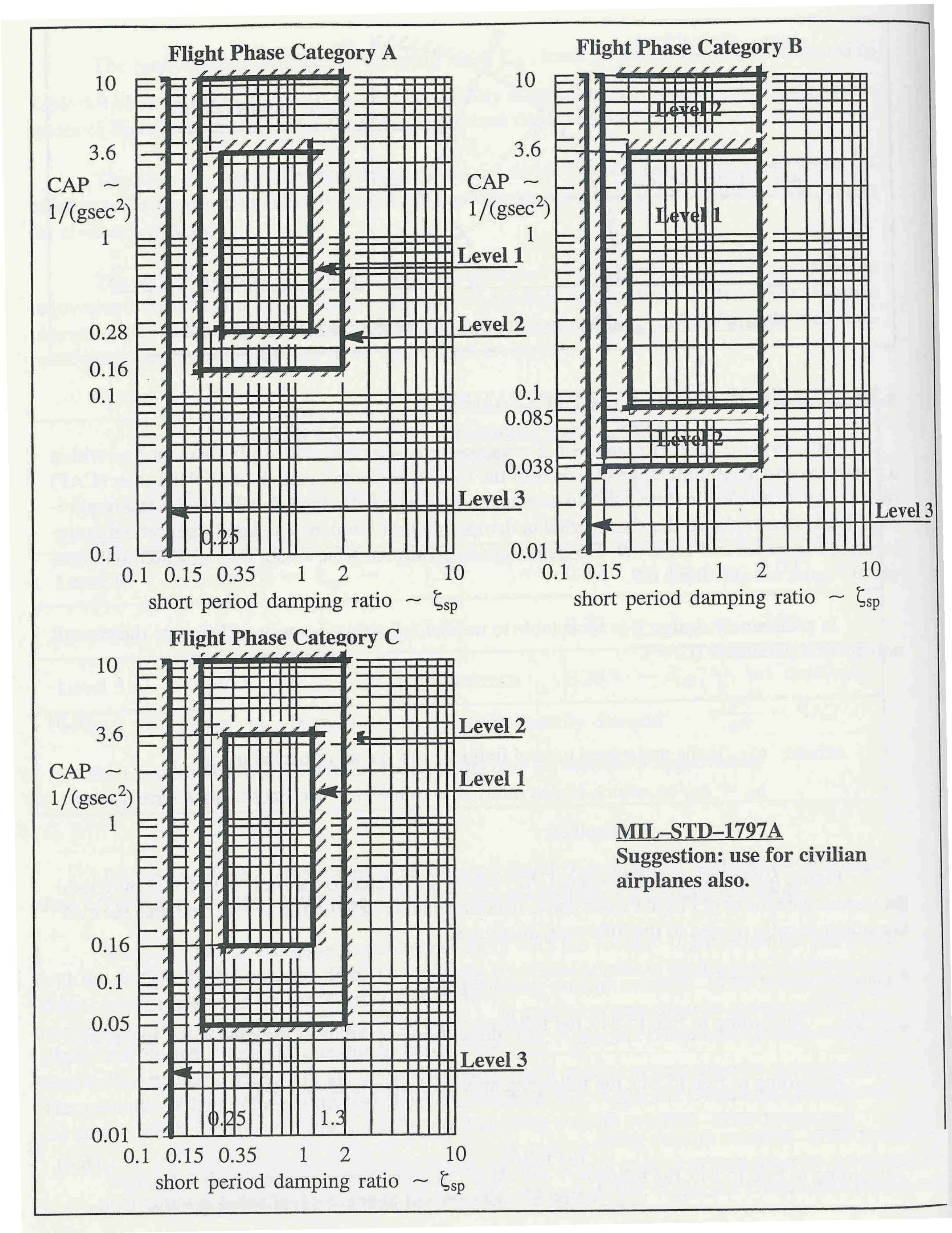


Figure 7.1: Control Anticipation Parameter and short period damping ratio requirements, source: ref. [4]

7.2 Gibson criterion

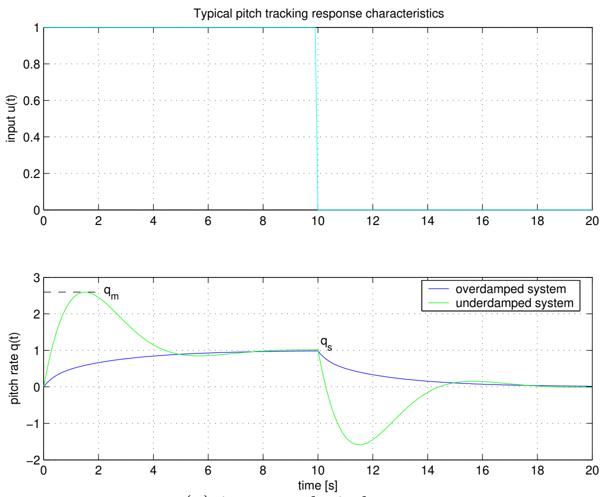
The Gibson Criterion grew out of the need to know how to design command and stability augmentation systems that would result in an aircraft with acceptable flying and handling qualities, a problem that is becoming increasingly urgent as fly-by-wire control systems become more and more common. This criterion consists of two separate criteria, namely the dropback criterion and the phase rate criterion. The phase rate criterion considers the robustness properties of the aircraft. Both will be given below.

7.2.1 Dropback criterion

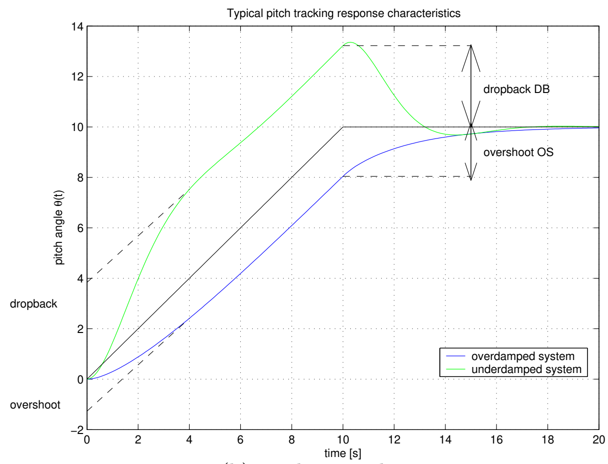
The criterion was originally defined in terms of limiting values on pitch rate overshoot ratio $\frac{q_m}{q_s}$ and on the ratio of attitude dropback (or overshoot, dependent on the direction of the transition when the step input is removed, see below) to steady state pitch rate. These parameters will be explained first.

All the parameters enumerated and explained below are illustrated in fig. 7.2.

- q_m: maximum pitch rate
- q_s: steady state value of pitch rate
- $\frac{q_m}{q_s}$: pitch rate overshoot ratio
- DB: dropback, amount of negative transition towards final value after the step input has been removed
- OS: overshoot, amount of positive transition towards final value after the step input has been removed



(a) input and pitch rate



(b) pitch attitude

Figure 7.2: Typical pitch tracking response characteristics

The requirements imposed on these parameters are shown in fig. 7.3 below.
Note that:

1. if the pitch rate overshoot ratio $\frac{q_m}{q_s} \leq 1$ then dropback is not possible and the lower part of the "satisfactory" region cannot be attained.
2. subsequent events have led Gibson to redefine the criterion such that zero dropback only is acceptable. The "satisfactory" region then collapses to the $\frac{q_m}{q_s}$-axis and in the event that this cannot be achieved precisely then it is better to transgress on the side of attitude dropback rather than overshoot.
3. the acceptable values of pitch rate overshoot lies in the range $1.0 \leq \frac{q_m}{q_s} \leq 3.0$

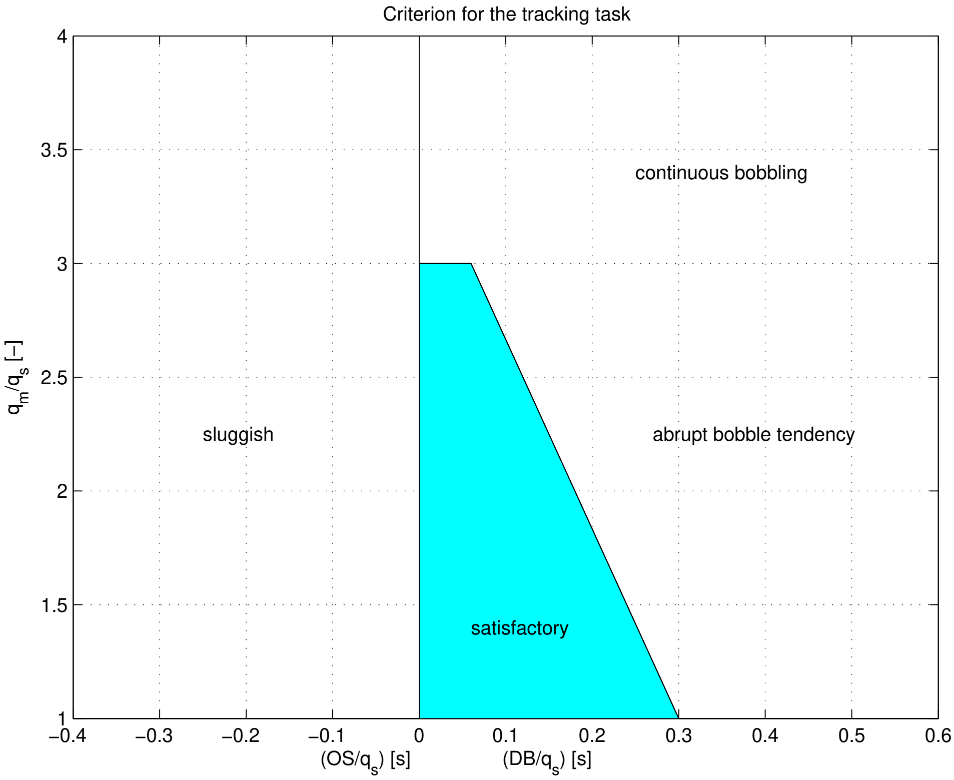


Figure 7.3: The criterion for the tracking task

7.2.2 Phase rate criterion

Even when the flying and handling qualities of a high order aircraft are acceptable it is possible that the closed loop gain and phase characteristics may be such that the addition of the pilot leads to the propagation of pilot induced oscillations (PIO) at certain conditions. It is now known that the likelihood of PIO is determined by the degrees of gain and phase compensation instinctively introduced by the pilot in flying the aircraft. The required compensation is in turn determined by the closed loop gain and phase characteristics of the aircraft at frequencies close to the "resonant" frequency of the human pilot. More recently, Gibson has turned his attention to the problem of PIO in otherwise satisfactory aircraft and has identified the desirable gain and phase characteristics for the closed loop high order aircraft if PIO is to be avoided. The findings define a useful adjunct to the criterion described above.
The phase rate criterion is concerned with the closed loop attitude frequency response in the region of $-180°$ phase and is evaluated from a plot of the closed loop attitude frequency response on a Nichols chart as shown in fig. 7.4.

Referring to fig. 7.4 the point of intersect is the "cross over point" where the phase first passes through $-180°$, the frequency corresponding with this point $\omega_{\phi=-180°}$ and the rate of change of phase with frequency at cross over $(\frac{\partial\phi}{\partial\omega})_{\phi=-180°}$ are relevant for the criterion.
Ideally, Gibson has established that:

$$\text{phase rate} = \left(\frac{\partial\phi}{\partial\omega}\right)_{\phi=-180°} \leq 85\frac{°}{Hz}, \qquad (7.5)$$

if PIO is to be avoided with a reasonable margin of certainty. The phase rate criterion is summarized in fig. 7.5 and generally requires that the cross over point should occur at a frequency of 0.5 Hz or above and that the phase rate should be less than 85° per Hz.

When an aircraft fails to meet the criterion suitable gain and phase compensation can be introduced into the command path using a lead-lag or lag-lead filter.

Note that when command path compensation is required it is unlikely that very large phase shift will be required. It may also be necessary to include some high frequency gain compensation in order to maintain the slope of the closed loop attitude frequency response plot to reasonable values at cross over.

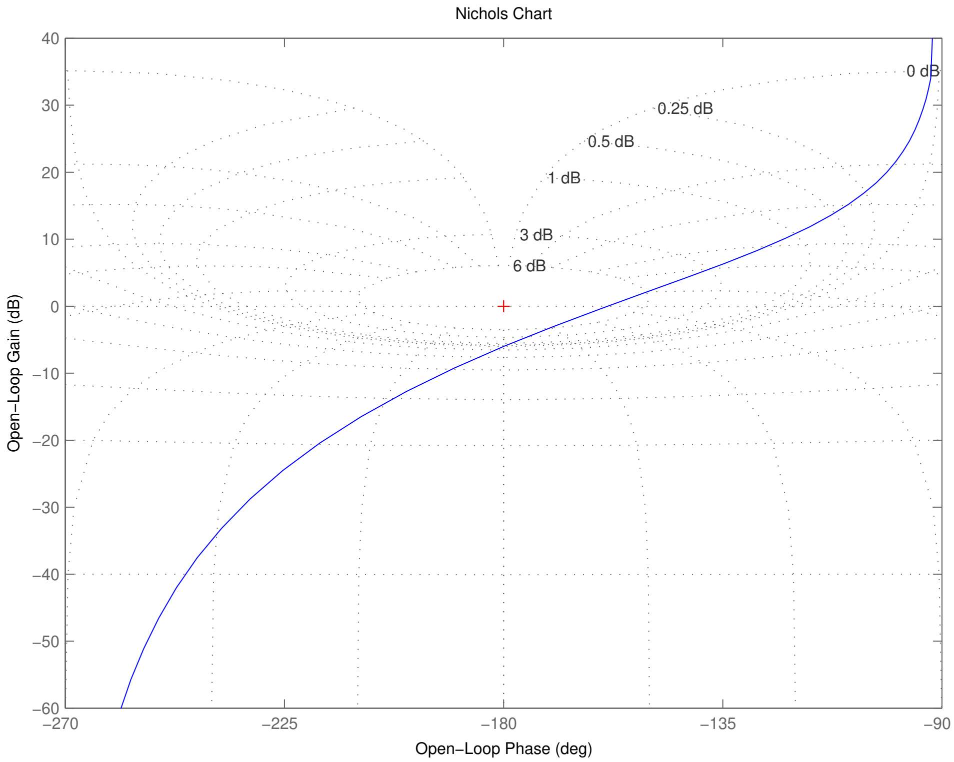


Figure 7.4: Closed loop attitude frequency response

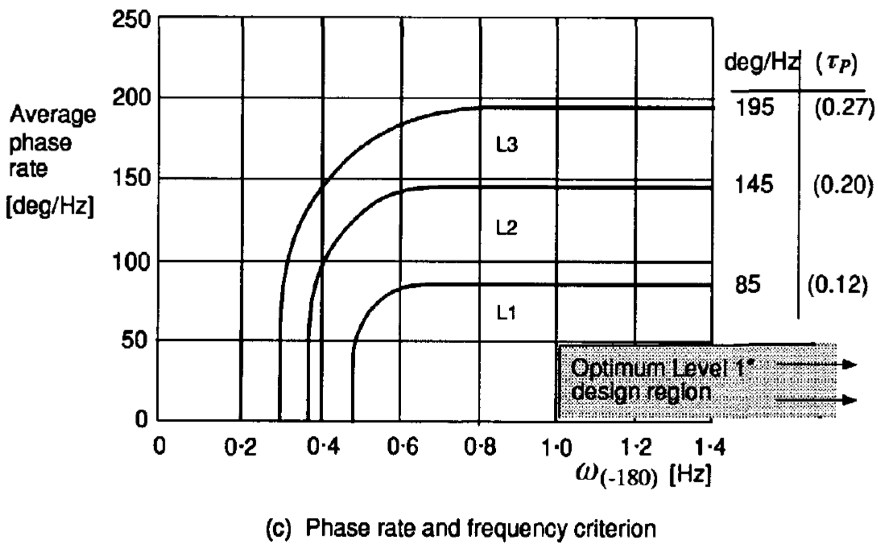


Figure 7.5: The phase rate criterion

7.3 Controller design task

Summarizing, the to-be-designed controller has to fulfill the requirements mentioned above, namely:

- the CAP (Control Anticipation Parameter) criterion
- the Gibson dropback criterion only

The following steps should be included in the design:

1. Construct the short period reduced model from the model without actuator dynamics, calculated in chapter 6. Mind that this short period reduced model involves only two states, namely angle of attack α and pitch rate q.

2. Compare time responses of the pitch rate q on a step input for the 4 state one without actuator dynamics and the reduced 2 state model without actuator dynamics. Analyze these time responses of q and determine which is the dominant mode. As you know that in this case only short term movements are considered, does this cause any problem? Why?

3. The requirements imposed by the CAP and Gibson criteria are not expressed in the usual quantities for controller design... Therefore, these requirements must be converted to the familiar frequency domain (requirements imposed on natural frequency $\omega_{n_{sp}}$, damping ratio ζ and time constant T_{θ_2}). These rewritten criteria can be expressed by the following relationships:

   - $\omega_{n_{sp}}(V,h) = 0.03V(V,h)$ with V in $[m/s]$
   - $1/T_{\theta_2}(V,h) = 0.75\omega_{n_{sp}}(V,h)$
   - $\zeta_{sp}(V,h) = 0.5$

4. What procedure can be used to obtain the required short period frequency $\omega_{n_{sp}}$ and damping ratio ζ?[1] Make a pitch rate command system using such a controller for the selected flight condition such that the short period frequency $\omega_{n_{sp}}$ and damping ratio ζ have the required value. Are the obtained levels of feedback gains acceptable with reference to possible gust? Mind the units in which K_α and K_q are expressed. Consider severe gust (design vertical gust: 4.572m/s) according to MIL-F-8785C (see lecture 11 about static stability augmentation systems).

5. The T_{θ_2} time constant cannot be modified by pole placement or another control loop structure. This needs to be done by means of pole-zero cancellation and by zero placement by means of a lead-lag prefilter. Explain why this prefilter must be located outside the loop.

6. Include a feed forward gain such that the steady state pitch rate tracks the input. Provide a control diagram of the pitch rate command system so far.

7. Give drawings of CAP and Gibson criteria allowable regions together with positions of the design point and the current parameter value. Verify if the requirements are met by implementing the obtained values into the following definitions:

   - $CAP = \frac{\omega_{n_{sp}}^2}{\frac{V}{g}\frac{1}{T_{\theta_2}}}$
   - $\frac{DB}{q_{ss}} = T_{\theta_2} - \frac{2\zeta_{sp}}{\omega_{n_{sp}}}$

8. Also verify if your controller satisfies the Gibson dropback criterion using time responses of pitch attitude angle and pitch rate.

[1]Mind that a known value of desired closed loop short period frequency $\omega_{n_{sp}}$ and damping ratio ζ actually mean that the location of the desired closed loop poles are known!

Chapter 8

Design of an automatic glideslope following and flare controller

The last chapter of the practical assignment deals with the design of an automatic glideslope following and flare control system for the F-16.

For this design, the following steps need to be performed:

1. Trim and linearize the F-16 model for the flight condition altitude $h = 5000ft$ and velocity $V = 300ft/s$. This flight condition is the lowest and slowest possible where the simulation results of this particular model are still valid.

2. Construct the reduced model for this flight condition, containing the **five** states **altitude** h, true airspeed V_t, angle of attack α, pitch attitude angle θ, and pitch rate q.

3. From level trimmed flight intercept a glideslope of 3 degrees that originates from an airfield located at an altitude of 3000ft. This means you intercept at a height of 2000ft above the runway. Place the runway in such a way that you intercept the glideslope approximately 10 seconds after starting your simulation.

4. Using a state space model block, define the aircraft model in Simulink. Subsequently, make a series connection at the input with a subsystem block containing the representation of the dynamics (using a transfer function block) and saturation limits (using a saturation block) for engine and actuator as follows:

   - engine: dynamics $H_{engine} = \frac{1}{s+1}$, saturation limits: [1000lb ; 19000lb]
   - elevator actuator: dynamics $H_{elevator} = \frac{20.2}{s+20.2}$, saturation limits: $[-25°; 25°]$

   Mind that when you trimmed the nonlinear model, the engine and the elevator were already set to a constant setting. The saturation blocks should be defined taking these trimmed values into account. Don't forget that δ_{el} and δ_{th} are deviations from the trimmed values and not the total values of elevator deflection and engine power.

5. Design a glideslope controller for this system, including any required inner loops, and incorporate this controller in the Simulink block diagram. Tune and modify the controller until you get a satisfactory[1] tracking response while not exceeding actuator limits. Explain in your report which inner loops you include and why, and what procedure you follow to find the control gains. Since the pilot will not fly manually, your controller does not need to satisfy the flying quality requirements.

6. Design a flare controller for this system and incorporate this controller in the Simulink block diagram. You are free to choose the geometry of the flare as long as you ensure that the vertical speed at touchdown is between 2 and 3 ft/s. Your simulation can stop as soon as there is touchdown. Analyse the touchdown location and vertical speed and compare it to your design values.

7. During glideslope and flare, keep the airspeed around 300ft/s (even though this is in reality too high for an F-16).

[1]This is your own interpretation of the controller performance. There is some design freedom here, but keep it realistic!

Some hints:

- Don't forget that the states of the system are deviations from the trimmed values. Therefore, the total altitude and the total velocity are given by:
  - $h_{total} = h_{trimmed} + h$
  - $V_{total} = V_{trimmed} + V_T$
- To calculate the position you can use the frequency domain approximation:
  position$(s) \approx \frac{V_T(s)}{s}$

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