AE4-301 Automatic Flight Control System Design

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Course structure

- Introduction automated flight control and recap flight dynamics.
- Control theory.
- **Performance and handling qualities.**
- Static/dynamic stability augmentation.
- Longitudinal autopilot modes.
- Lateral autopilot modes.

Lecture 9 structure

- Introduction.
- Longitudinal flying qualities requirements.
- Lateral flying qualities requirements.
- Pilot opinion rating.
- Flight Control System characteristics.
- Gibson dropback criterion.

Introduction

Flying and handling qualities of an airplane:

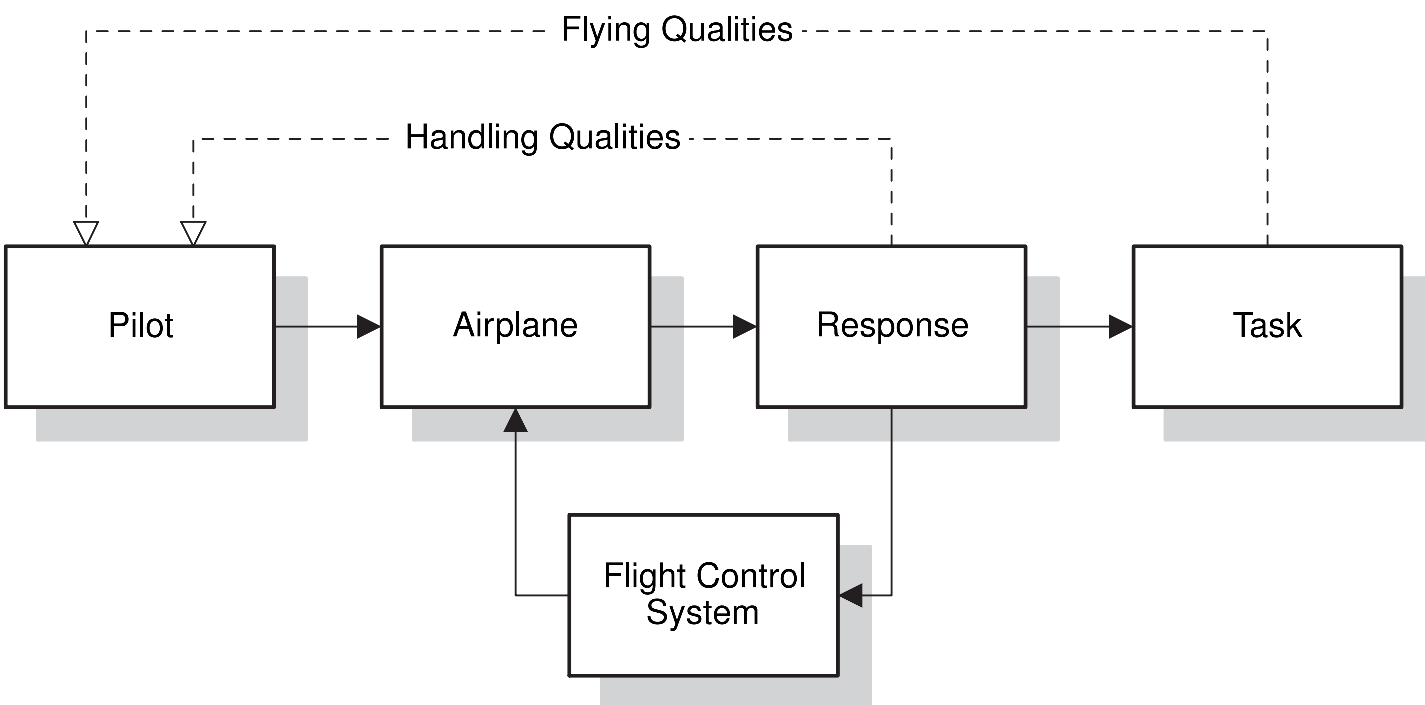
> *Those properties which describe the ease and effectiveness with which an airplane responds to pilot commands in the execution of some flight task.*

This is based upon a pilot opinion: subjective?

PIO Movies

Introduction

- Flying qualities: task related.
- Handling qualities: short term response related.



Introduction

Flying quality is influenced by:

- Airframe stability, control & dynamics.
- FCS dynamics.
- Response to atmospheric turbulence.
- Cockpit design.
- ...

Mathematical models are required to investigate the effect of these elements individually, but this is not always possible.

Introduction

Stability = ease of establishment of equilibrium flight condition without divergent tendency.

Trade-off stability vs. manoeuvrability:

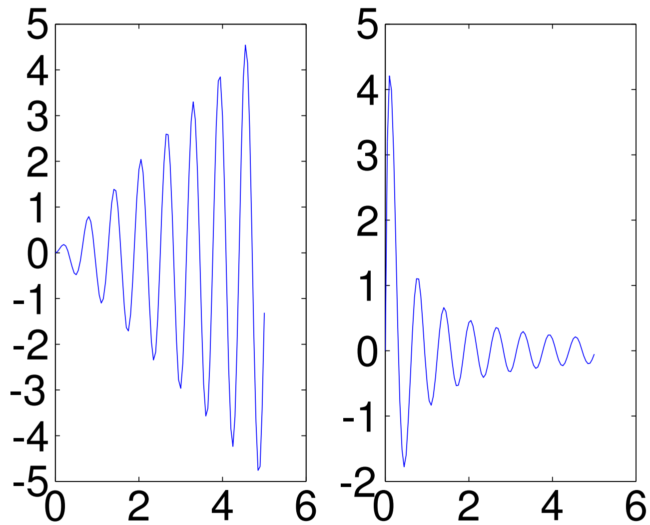
- High stability: large control forces.
- High manoeuvrability: small control forces.

CSAS: Control and Stability Augmentation System. Assistance in achieving well-harmonized control characteristics over the entire flight envelope by artificial means.

Introduction

Definition stability:

- Static stability: Deviation from equilibrium causes an opposing force/moment.
- Dynamic stability: Deviation is eliminated over time.



Introduction

Organisations and regulations.

- USA
  - Federal Aviation Administration(FAA)
  - Federal Aviation Regulations(FAR)
  - Department of Defence (DoD): Military (MIL-F-8785c)
- Europe
  - European Aviation Safety Agency (EASA)
  - NL: Inspectie Leefomgeving en Transport (ILT)
- UK
  - Civil Aviation Authority(CAA)
  - MoD: Military

Introduction

These organisations have posed a set of flying qualities requirements, i.e. the minimum acceptable standard of flying qualities.

Demonstration of compliance with requirements is performed with flight tests.

Aircraft type	EASA	FAA
Small airplanes	CS-23	FAR Part 23
Transport airplanes	CS-25	FAR Part 25
Small rotorcraft	CS-27	FAR Part 27
Large rotorcraft	CS-29	FAR Part 29
Very light airplanes	CS-VLA	
...		

Introduction

In this course, we will use the US military standard MIL-F-8785c for most examples. Its requirements are specified based on:

- Class of aircraft.
- Flight phase.

Introduction

Aircraft classification

Class I: small light airplanes.

Cessna 210
Rockwell OV-10A
Eclipse 500
Extra 400

Introduction

Aircraft classification

Class II: medium weight, low/medium manoeuvrability.

Grumman E-2C
Lockheed C-130
Boeing 737
Airbus A320

Introduction

Aircraft classification

Class III: large, heavy, low/medium manoeuvrability.

McDD C-17
Boeing B52
Boeing 747
Airbus A340

Introduction

Aircraft classification

Class IV: high manoeuvrability airplanes.

Lockheed F22
Lockheed Sr71
Extra 300
SU-26M

Introduction

Flight phase categories.

Category A: Non-terminal flight phases that require rapid manoeuvring, precision tracking or precise flight path control.

- Air-to-air combat
- Terrain following
- In-flight refueling
- Airshow demo

Introduction

Flight phase categories.

Category B: Non-terminal flight phases that require gradual manoeuvring, less precise tracking and flight path control.

- Climb
- Cruise
- Descent

Introduction

Flight phase categories.

Category C: Terminal flight phases that require gradual manoeuvring and precision flight path control.

- (catapult) takeoff
- landing
- (aborted) approach

Introduction

Levels of flying qualities

- **Level 1:** Flying qualities clearly adequate for the mission flight phase.
- **Level 2:** Flying qualities adequate to accomplish the mission flight phase, but with an increase in pilot workload and/or degradation in mission effectiveness.
- **Level 3:** Degraded flying qualities, but such that the airplane can be controlled, inadequate mission effectiveness and high, or limiting, pilot workload.

Introduction

Airplanes must be designed to satisfy level 1 flying quality requirements with all systems in their normal operating state.

The level of degradation probability is related to safety-crucial system failure probability.

Occurrence probability	MIL-F-8785c operational flight envelope	MIL-F-8785C service flight envelope
Level 2	$< 10^{-2}$ per flight	
Level 3	$< 10^{-4}$ per flight	$< 10^{-2}$ per flight

Longitudinal requirements

- Longitudinal control forces.
  - Manoeuvring flight
  - Steady state flight
  - takeoff and landing
  - Dives
- Phugoid damping.
- Flight path stability.
- Short period frequency and damping.
- Control Anticipation Parameter.

Longitudinal requirements

Longitudinal control forces in flight (don't memorize these!):

- Control force versus load factor gradients $\frac{\partial F_s}{\partial n}$ must fall within limits.
- No significant non-linearities in gradients.
- Center stick / wheel controllers.

Longitudinal requirements

MIL-F-87875C centre stick controller	$\min \frac{\partial F_s}{\partial n} \left[\frac{lbs}{g}\right]$	$\max \frac{\partial F_s}{\partial n} \left[\frac{lbs}{g}\right]$
Level 1	$\max\left\{\frac{21}{n_{\lim}-1}, 3.0\right\}$	$\frac{56}{(n_{\lim}-1)} \leq \frac{240}{\left({}^{n}\!/\!_{\alpha}\right)} \leq 28.0$
Level 2	$\max\left\{\frac{18}{n_{\lim}-1}, 3.0\right\}$	$\frac{85}{(n_{\lim}-1)} \leq \frac{360}{\left({}^{n}\!/\!_{\alpha}\right)} \leq 42.5$
Level 3	$\max\left\{\frac{12}{n_{\lim}-1}, 2.0\right\}$	56.0

Longitudinal requirements

MIL-F-87875C wheel controller	$\min \frac{\partial F_s}{\partial n} \left[\frac{lbs}{g}\right]$	$\max \frac{\partial F_s}{\partial n} \left[\frac{lbs}{g}\right]$
Level 1	$\max\left\{\frac{35}{n_{\lim}-1}, 6.0\right\}$	$\frac{120.0}{(n_{\lim}-1)} \leq \frac{500.0}{\left({}^{n}/_{\alpha}\right)} \leq 120.0$
Level 2	$\max\left\{\frac{30}{n_{\lim}-1}, 6.0\right\}$	$\frac{182.0}{(n_{\lim}-1)} \leq \frac{775.0}{\left({}^{n}/_{\alpha}\right)} \leq 182.0$
Level 3	5.0	240.0

Longitudinal requirements

Civilian requirements		
CS-VLA	FAR-23	FAR-25
$\frac{\partial F_s}{\partial n} > \frac{15.7}{n_{\text{lim}}}$	Wheel control: $\max\left\{\frac{(W_{TO}/140)}{n_{\text{lim}}}, \frac{15.0}{n_{\text{lim}}}\right\} < \frac{\partial F_s}{\partial n} < \frac{15.7}{n_{\text{lim}}}$ Stick control: $\frac{\partial F_s}{\partial n} > \frac{W}{140}$	No requirement

Longitudinal requirements

Longitudinal forces in steady flight. With airplane configuration changes (gear, flaps, etc.), control force changes must be within certain limits.

Max force [lbs]	CS-VLA	FAR-23	FAR-25
Temporary force: Center stick: Wheel controller:	 45.0 56.2	 60.0 75.0	 No req. 75.0
Prolonged force: Any controller:	 4.5	 10.0	 10.0
MIL-F-8785C: No requirement			

Longitudinal requirements

Longitudinal control forces in takeoff and landing:

- With fixed trim controls, control force must be within limits.
- In practice:
  - Trim is adjusted such that the control forces do not change.
  - Trim compensates for weight changes.

Longitudinal requirements

Longitudinal control forces in takeoff should be within:

MIL-F-8785C Airplane class	Takeoff control force [lbs] pull	push
Nosewheel:		
Classes I, IV-C	20.0	10.0
Classes II-C, IV-L	30.0	10.0
Classes II-L, III	50.0	20.0
Tail wheel:		
Classes I, II-C, IV	20.0	10.0
Classes II-L, III	35.0	15.0

Longitudinal requirements

Longitudinal control forces in landing:

MIL-F-8785C Airplane class	Landing control force [lbs] pull only
Classes I, II-C	35.0
Classes II-L	50.0
CS-VLA FAR-23 FAR-25	See steady-state requirements

Longitudinal requirements

Dynamic response requirements: Phugoid damping.

Oscillations must meet requirement:

	MIL-F-8785C	CS-VLA, FAR-23/25
Level 1	$\zeta_{ph} \geq 0.04$	No req.
Level 2	$\zeta_{ph} \geq 0$	No req.
Level 3	$T_{2_{ph}} \geq 55s$	No req.

Longitudinal requirements

Example F16 altitude 12km, airspeed 330 km/h.

- Trim and Linearize nonlinear Simulink model.
- Extract state space matrices.
- Convert to transfer function.
- Elevator deflection to airspeed:

```
Transfer function:
  -0.07016 s^3 + 0.04262 s^2 + 0.7298 s + 0.04113
---------------------------------------------------
s^4 + 0.4284 s^3 + 0.6968 s^2 + 0.05799 s + 0.01103
```

Longitudinal requirements

Example F16 altitude 12km, airspeed 330 km/h.
Poles:

$$
\begin{aligned}
-0.1742 + 0.7884i \\
-0.1742 - 0.7884i \\
-0.0399 + 0.1238i \\
-0.0399 - 0.1238i
\end{aligned}
$$

- Which eigenmotions are these?
- Which poles belong to which eigenmotion?
- **Exercise:** Is the level 1 requirement for phugoid damping satisfied?

Longitudinal requirements

$$-0.1742 + 0.7884i$$
$$-0.1742 - 0.7884i$$
$$\mathbf{-0.0399 + 0.1238i}$$
$$\mathbf{-0.0399 - 0.1238i}$$

The equation for the poles is:

$$s = -\zeta\omega_n \pm i\omega_n\sqrt{1-\zeta^2}$$

$$\omega_n = \sqrt{(\zeta\omega_n)^2 + \left(\omega_n\sqrt{1-\zeta^2}\right)^2}$$

$$\zeta = \frac{\zeta\omega_n}{\omega_n} = \frac{0.0399}{\sqrt{0.0399^2 + 0.1238^2}} = 0.307$$

Longitudinal requirements

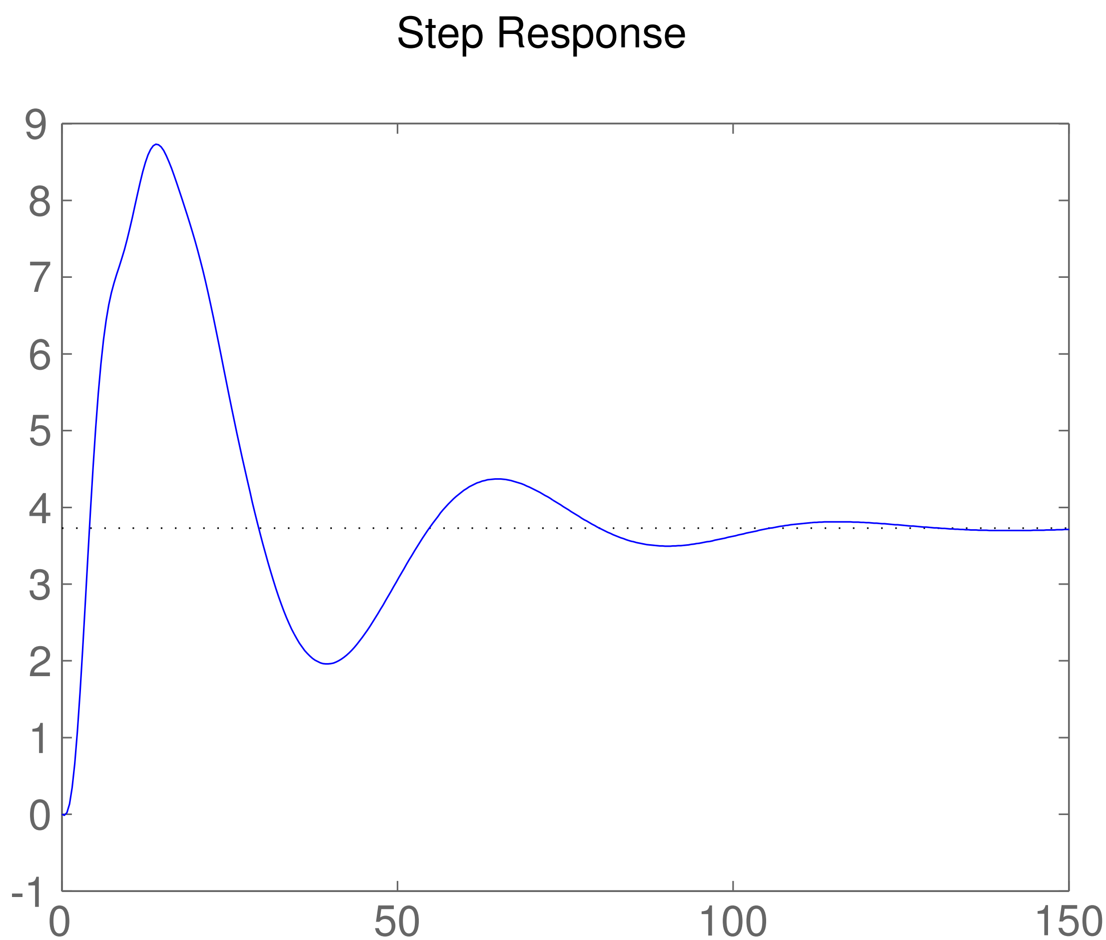


Figure 1: Airspeed response for step on elevator.

Longitudinal requirements

Phugoid damping.

Oscillations must meet requirement:

	MIL-F-8785C	CS-VLA, FAR-23/25
Level 1	$\zeta_{ph} \geq 0.04$	No req.
Level 2	$\zeta_{ph} \geq 0$	No req.
Level 3	$T_{2_{ph}} \geq 55s$	No req.

Longitudinal requirements

Phugoid oscillation expression:

$$A(t) = A_{ph} e^{-\zeta_{ph}\omega_{n_{ph}} t} \sin\left(\omega_{n_{ph}} t + \phi_{ph}\right)$$

Exercise:
An aircraft can still satisfy Level 3 phugoid requirements with a negative(=unstable) phugoid damping ratio. If $\zeta_{ph} = -0.1$ and $\omega_{n_{ph}} = 0.12$, compute the time to double amplitude $T_{2_{ph}}$.

Longitudinal requirements

Phugoid oscillation expression:

$$A\left(t\right) = A_{ph} e^{-\zeta_{ph}\omega_{n_{ph}}t} \sin\left(\omega_{n_{ph}}t + \phi_{ph}\right)$$

Exercise:
An aircraft can still satisfy Level 3 phugoid requirements with a negative(=unstable) phugoid damping ratio. If $\zeta_{ph} = -0.1$ and $\omega_{n_{ph}} = 0.12$, compute the time to double amplitude $T_{2_{ph}}$.

$$A(T_{2ph}) = 2A_{ph} = A_{ph} e^{-\zeta_{ph}\omega_{n_{ph}}T_{2ph}}$$

$$T_{2_{ph}} = -\frac{\ln 2}{\zeta_{ph}\omega_{n_{ph}}} \approx -\frac{0.693}{-0.1 * 0.12} = 57.76s$$

Longitudinal requirements

Flight path stability. Derivative $\frac{\partial \gamma}{\partial V_P}$ in approach (flight phase category C), with changes in pitch control only (no throttle changes), must meet requirements:

	MIL-F-8785C	CS-VLA, FAR-23/25
Level 1	$\frac{\partial \gamma}{\partial V_P} \leq 0.06° / knot$	No req.
Level 2	$\frac{\partial \gamma}{\partial V_P} \leq 0.15° / knot$	No req.
Level 3	$\frac{\partial \gamma}{\partial V_P} \leq 0.24° / knot$	No req.

Longitudinal requirements

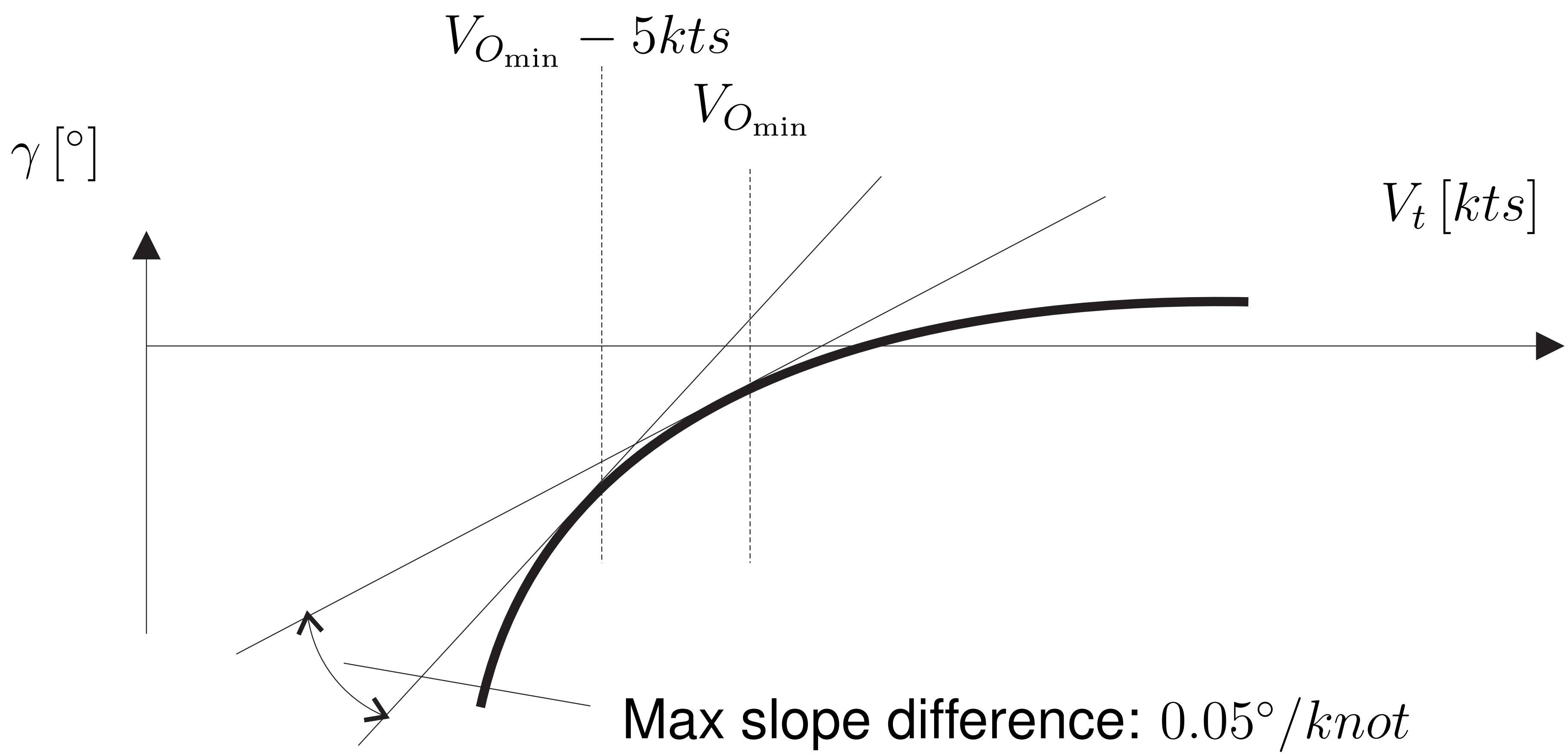
Flight path stability.

Additionally, derivative $\frac{\partial \gamma}{\partial V_P}$ in approach (flight phase category C), with changes in pitch control only (no throttle changes), must meet the requirements at $V_{O_{min}}$:

$$\frac{\partial \gamma}{\partial V_P}(V = V_{O_{\min}} - 5kts) \leq \frac{\partial \gamma}{\partial V_P}(V = V_{O_{\min}}) + 0.05^\circ/knot$$

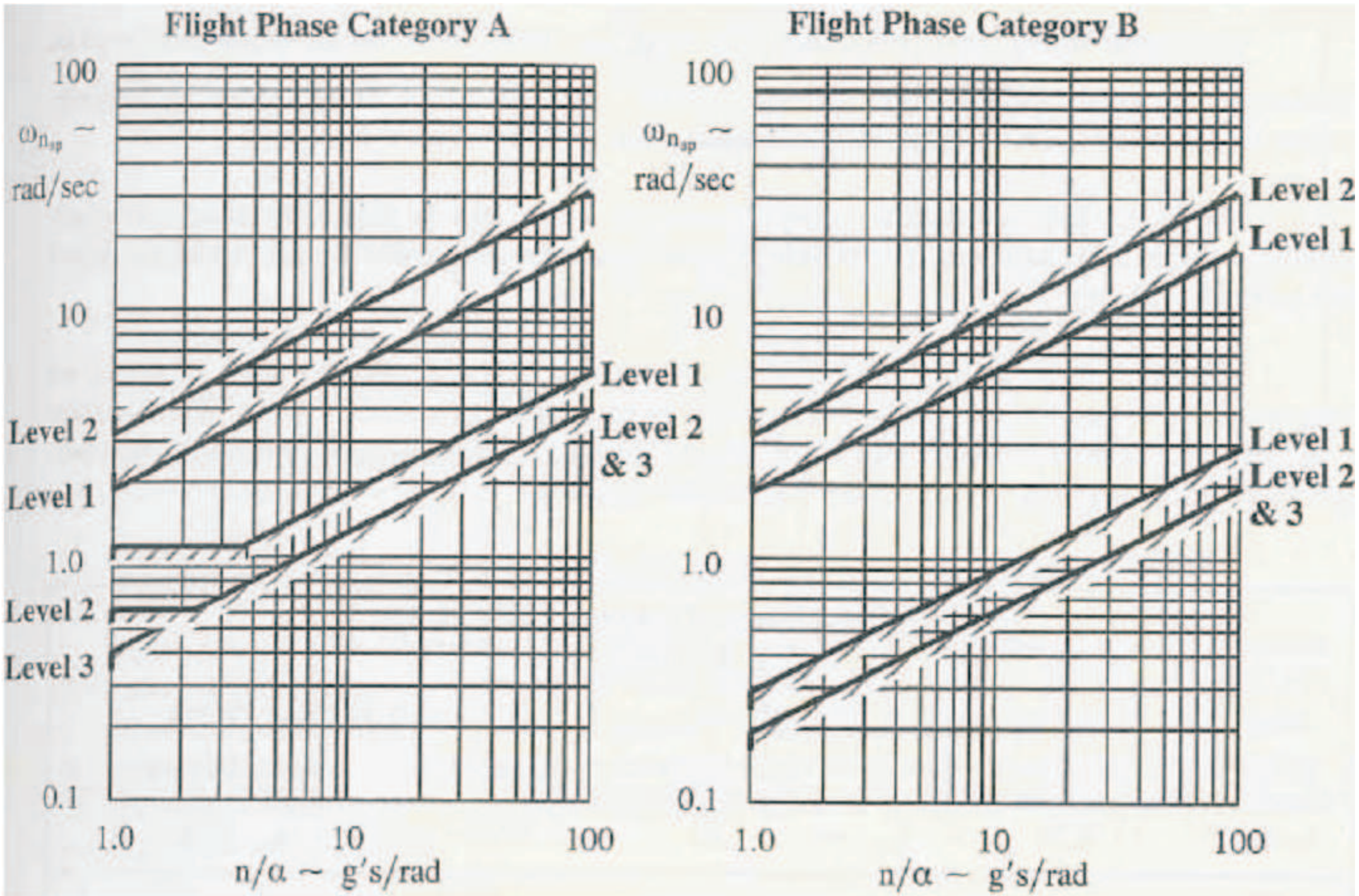
Longitudinal requirements

Flight path stability.



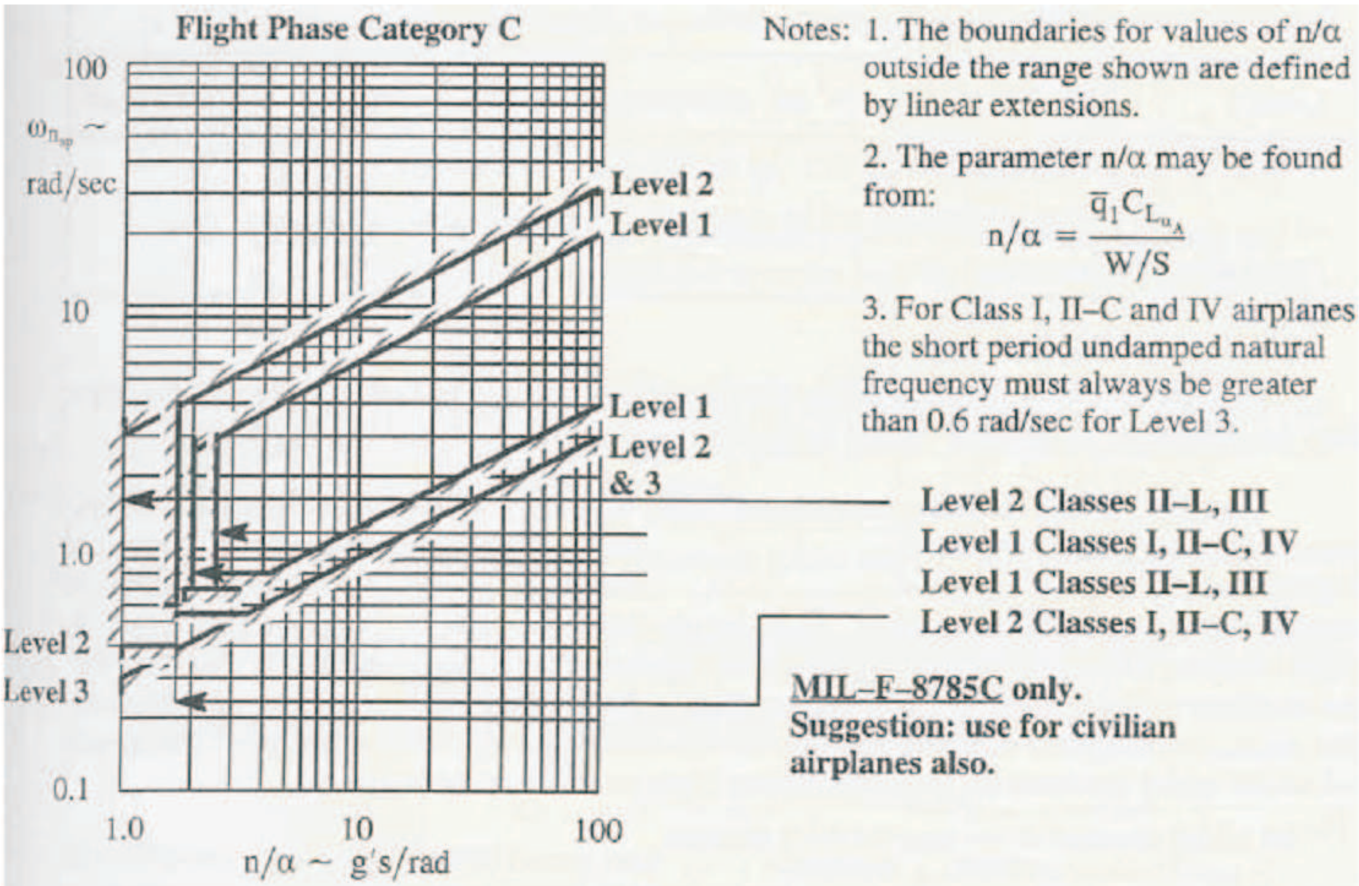
Longitudinal requirements

Short period frequency and damping (MIL-F-8785C).



Longitudinal requirements

Short period frequency and damping (MIL-F-8785C).



Longitudinal requirements

Short period damping ratio limits (MIL-F-8785C).

ζ_{sp}	cat. A & C phase		cat. B phase	
Level	min	max	min	max
Level 1	0.35	1.30	0.30	2.00
Level 2	0.25	2.00	0.20	2.00
Level 3	0.15	$-$	0.15	$-$

Longitudinal requirements

Example F16 altitude 12km, airspeed 330 km/h.
Poles:

$$\begin{aligned} &-0.1742 + 0.7884i \\ &-0.1742 - 0.7884i \\ &-0.0399 + 0.1238i \\ &-0.0399 - 0.1238i \end{aligned}$$

- (Which eigenmotions are these?)
- (Which poles belong to which eigenmotion?)
- Is the requirement for short period damping satisfied?

Longitudinal requirements

$$\begin{gathered}\mathbf{-0.1742 + 0.7884i} \\ \mathbf{-0.1742 - 0.7884i} \\ -0.0399 + 0.1238i \\ -0.0399 - 0.1238i\end{gathered}$$

The equation for the poles is:

$$s = -\zeta\omega_n \pm i\omega_n\sqrt{1-\zeta^2}$$

$$\omega_n = \sqrt{(\zeta\omega_n)^2 + \left(\omega_n\sqrt{1-\zeta^2}\right)^2}$$

$$\zeta = \frac{\zeta\omega_n}{\omega_n} = \frac{0.1742}{\sqrt{0.1742^2 + 0.7884^2}} = 0.216$$

Longitudinal requirements

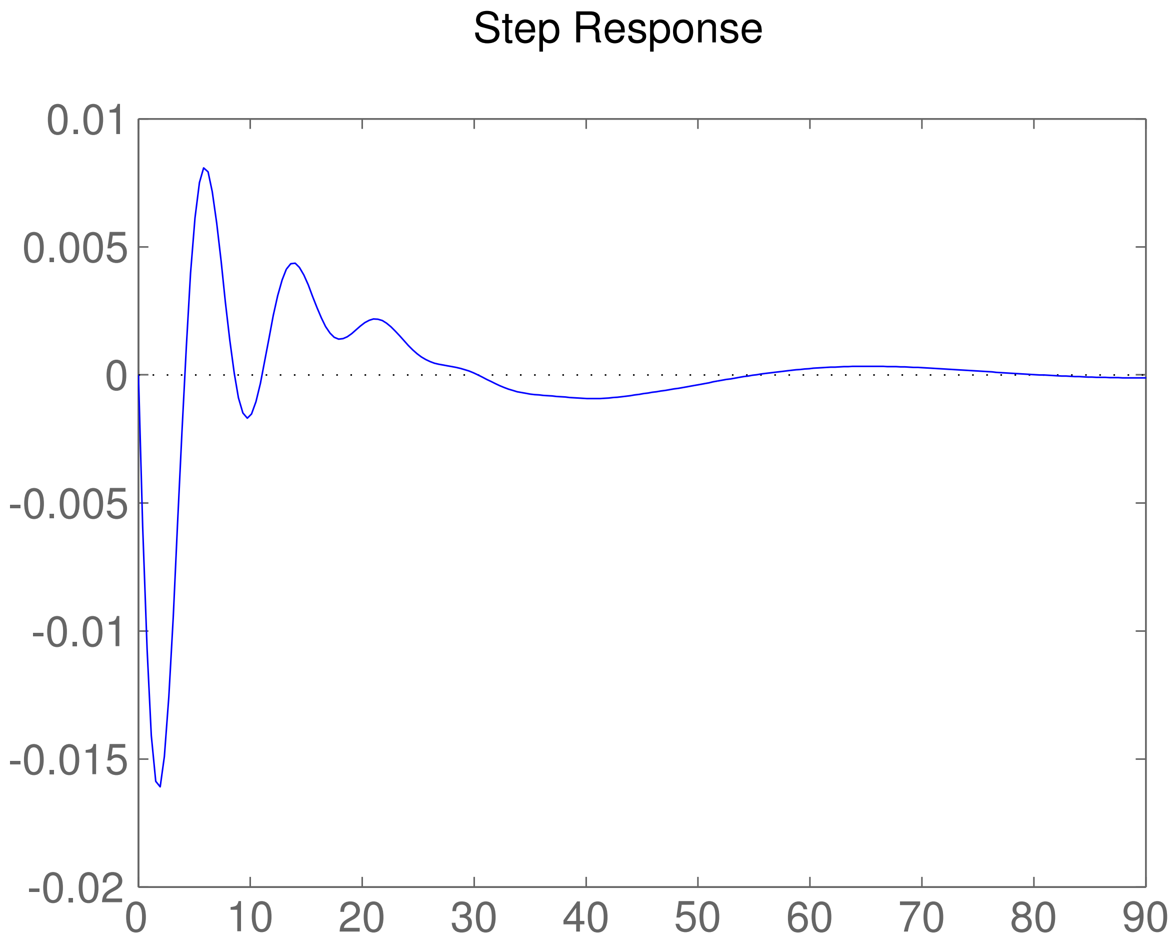
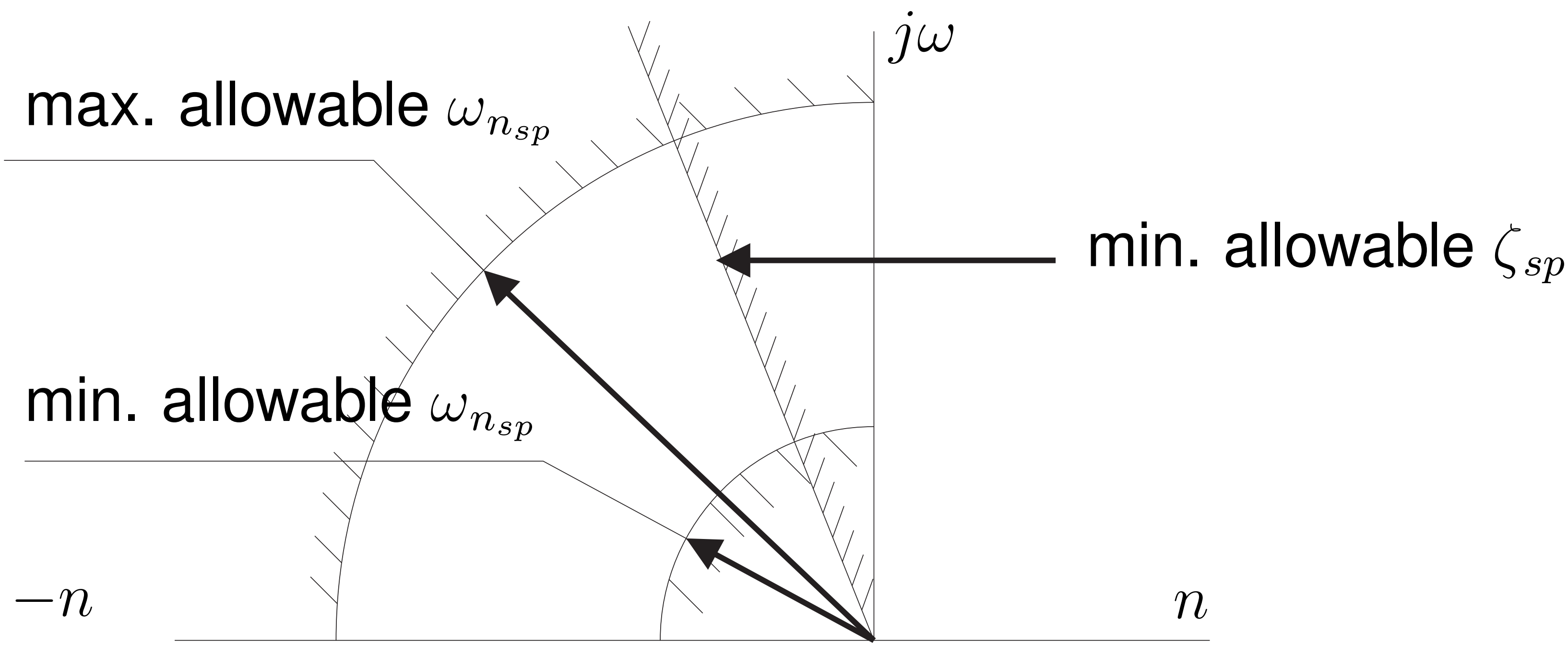


Figure 2: Pitch rate response for step on elevator.

Longitudinal requirements

Short period frequency and damping, s-plane representation.



Longitudinal requirements

Control Anticipation Parameter(CAP) replaces short period frequency and damping requirements for highly augmented airplanes.

$$CAP = \frac{\dot{q}\left(t=0\right)}{n_z\left(t=\infty\right)} \quad \rightarrow \quad CAP = \frac{\omega_{n_{sp}}^2}{n_\alpha}$$

where

- $\omega_{n_{sp}}$ is the undamped natural frequency of the short period mode
- $n_\alpha = \frac{\partial n}{\partial \alpha}$ is the gust or load-factor sensitivity.

Longitudinal requirements

Load factor sensitivity:

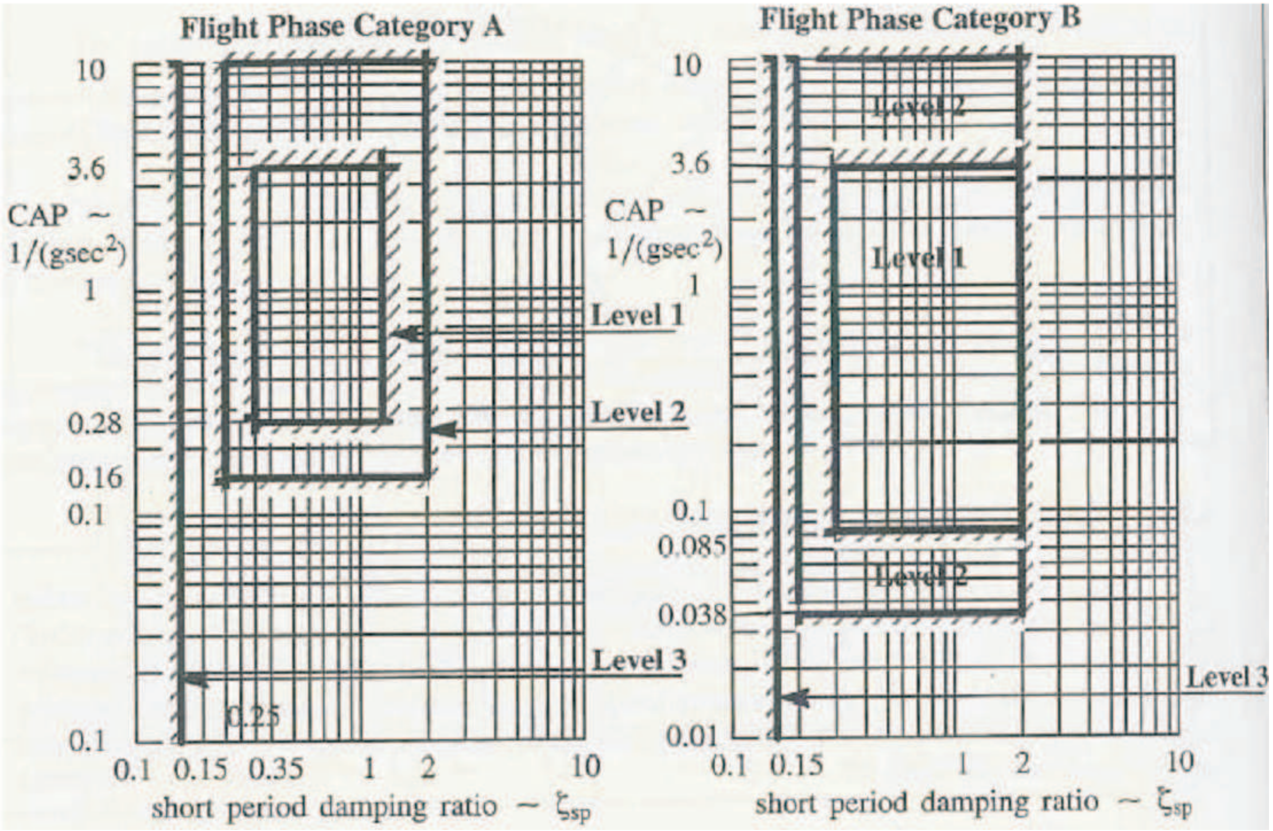
$$n = \frac{L}{W} = \frac{C_L \bar{q} S}{W} \quad \rightarrow \quad n_\alpha = \frac{C_{L_\alpha} \bar{q} S}{W}$$

Alternative CAP expression:

$$CAP = \frac{W\bar{c}}{I_{YY}} \left(-\frac{g\rho S \bar{c} C_{m_q}}{4W} + \bar{x}_{ac} - \bar{x}_{cg} \right)$$

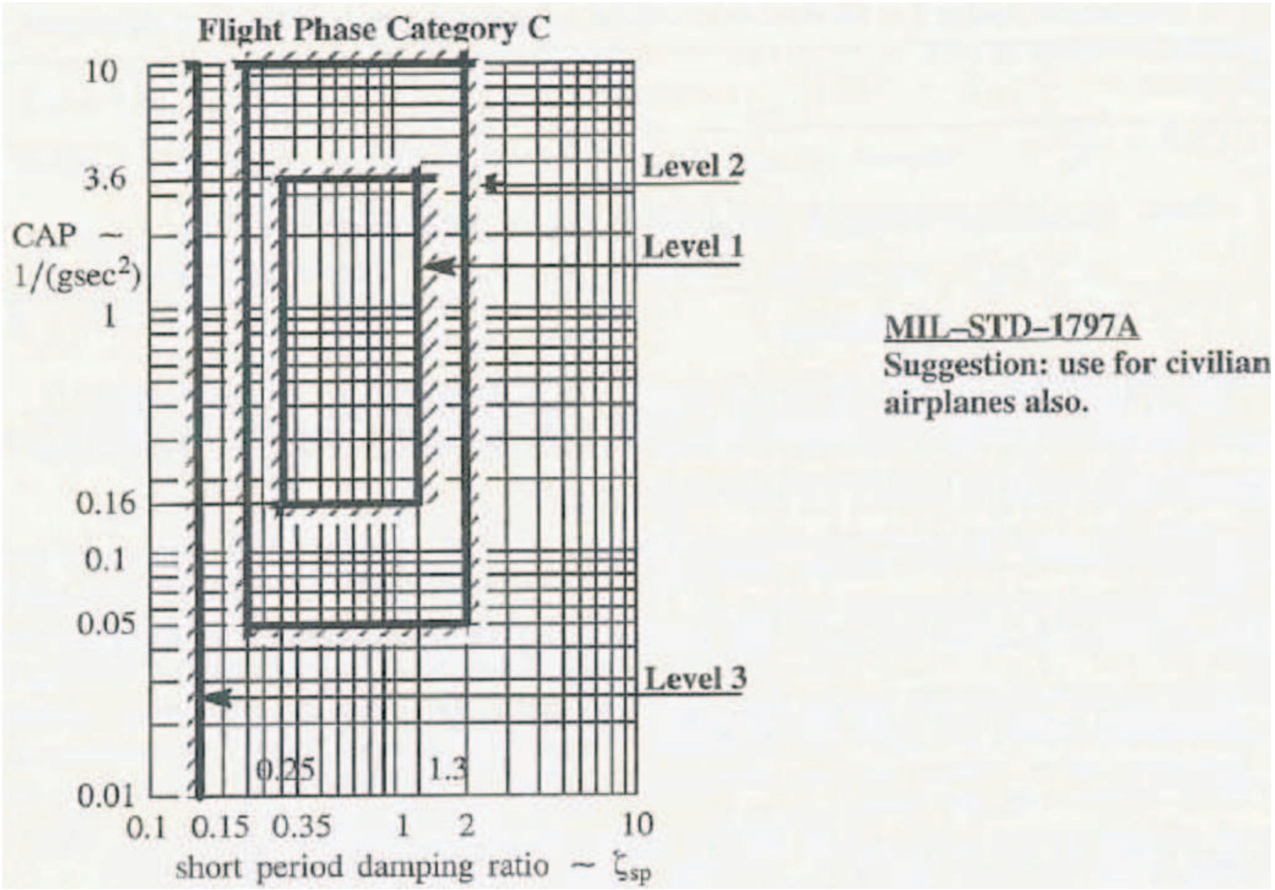
Longitudinal requirements

Control Anticipation Parameter(CAP):



Longitudinal requirements

Control Anticipation Parameter(CAP):



Lateral requirements

- Lateral control forces
- Dutch roll frequency and damping
- Spiral stability
- Coupled roll-spiral stability
- Roll mode time constant
- Roll control effectiveness
- Steady sideslips

Lateral requirements

Lateral control forces:

- Roll control forces.
- Directional control forces with asymmetric loadings.
- Directional and roll control forces with one engine inoperative.

Lateral requirements

Max roll control forces [lbs], MIL-F-8785C

Level	Class	Category	Stick force	Wheel force
Level 1	I, II-C, IV	A, B	20.0	40.0
		C	20.0	20.0
	II-L, III	A, B	25.0	50.0
		C	25.0	25.0
Level 2	I, II-C, IV	A, B	30.0	60.0
		C	20.0	20.0
	II-L, III	A, B	30.0	60.0
		C	30.0	30.0
Level 3	all	all	35.0	70.0

Lateral requirements

Directional control forces with asymmetric loadings.

Trimmed for symmetrical loading in straight line path, rudder pedal forces due to asymmetric loading must be less then 100 lbs for level 1&2 and less then 180 lbs for level 3.

Directional and roll control forces with one engine inoperative

Trim controls fixed and one engine inoperative: maximum allowable directional and roll control forces may not exceed certain limits.

Lateral requirements

Max force [lbs]	CS-VLA	FAR-23	FAR-25
Temporary force:			
Wheel controller	45.0	60.0	60.0
Rudder pedal	90.0	150.0	150.0
Prolonged force:			
Wheel controller	3.4	5.0	5.0
Rudder pedal	22.5	20.0	20.0
MIL-F-8785C: no req. pedal force < 180 lbs			

Lateral requirements

Dutch roll frequency and damping, MIL-F-8785C

Level	Category	Class	min ζ_d	min $\zeta_d \omega_{n_d}$	min ω_{n_d}
Level 1	A(C,GA)	IV	0.4	–	1.0
	A	I and IV	0.19	0.35	1.0
		II and III	0.19	0.35	0.4
	B	all	0.08	0.15	0.4
	C	I,II-C,IV	0.08	0.15	1.0
		II-L,III	0.08	0.10	0.4
Level 2	all	all	0.02	0.05	0.4
Level 3	all	all	0	–	0.4

Consider: $A(t) = A_d e^{-\zeta_d \omega_{n_d} t} \sin\left(\omega_{n_d} t + \phi_d\right)$

Lateral requirements

Dutch roll frequency and damping, civilian requirements.

- CS-VLA: $\zeta_d > 0.052$ with controls free and fixed.
- FAR-23: $\zeta_d > 0$.
- FAR-25: $\zeta_d > 0$ with controls free and controllable without exceptional pilot skills.

Spiral stability, MIL-F-8785C Allowable divergence, cockpit controls free

Category	Level 1	Level 2	Level 3
A and C	$T_{2_s} > 12s$	$T_{2_s} > 8s$	$T_{2_s} > 4s$
B	$T_{2_s} > 20s$	$T_{2_s} > 8s$	$T_{2_s} > 4s$
Civilian		none	

Lateral requirements

Roll mode time constant MIL-F-8785C Measure of rapidity of the roll response: smaller=faster.

Category	Class	Level 1	Level 2	Level 3
A	I and IV	$T_r < 1.0s$	$T_r < 1.4s$	$T_r < 10.0s$
	II and III	$T_r < 1.4s$	$T_r < 3.0s$	$-$
B	all	$T_r < 1.4s$	$T_r < 3.0s$	$T_r < 10.0s$
C	I,II-C,IV	$T_r < 1.0s$	$T_r < 1.4s$	$T_r < 10.0s$
	II-L, III	$T_r < 1.4s$	$T_r < 3.0s$	$-$
Civilian requirements:		None		

Lateral requirements

Roll control effectiveness
Minimum bank angle response within specified time after full lateral control deflection. This is determined by:

- Aircraft and aileron geometery.
- Flight control system lag.

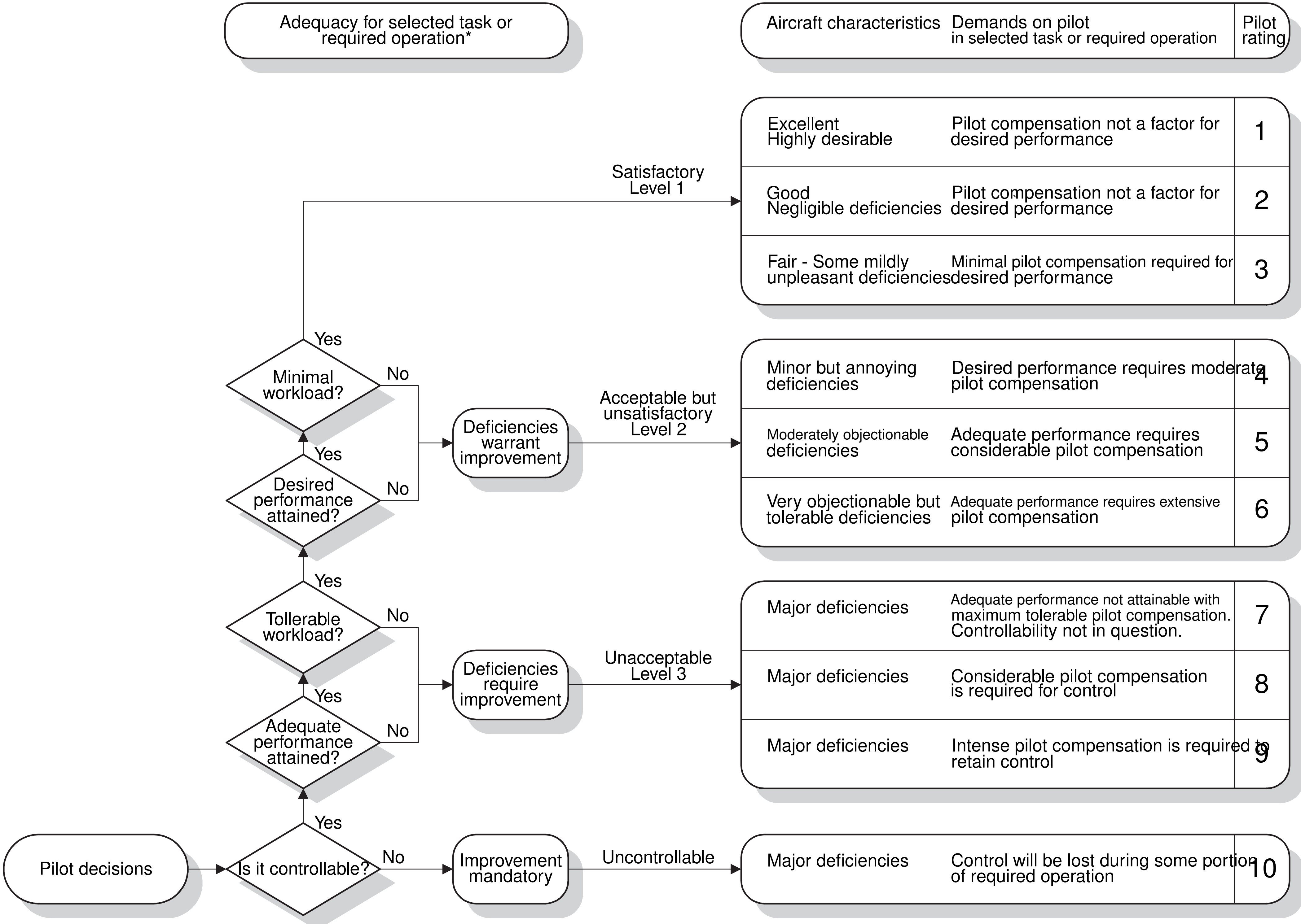
Steady sideslips Positive sideslip = wind component from the right (nose pointing to the left of the flying direction).

- $C_{n_\beta} > 0$ (directional stability)
- $C_{y_\beta} < 0$
- $C_{l_\beta} < 0$ (Effective dihedral)

Pilot opinion rating

Cooper-Harper rating scale.

- Formal procedure for qualitative assessment of aircraft flying qualities.
- Pilot rating between 1 and 10:
  - 1: Excellent handling qualities, low workload.
  - 10: Many handling qualities deficiencies.



* Definition of required operation involves designation of flight phase and/or subphases with accompanying conditions

Flight Control System characteristics

3 FCS factors with detrimental effect on flying qualities:

- In signal paths between cockpit and control surfaces:
  - Lags in mechanical signal paths.
  - Mechanical signal path compliance (deflection under load).
  - Actuator blow-down under load.
  - Electrical or FBW signal distortions.
- Lags in actuator response to inputs.
- Lags in displays.

Gibson criterion

CAS & SAS design requirements resulting in acceptable flying & handling qualities. Especially important for fly-by-wire systems with inherent instability.

2 separate criteria:

- Dropback criterion: transient behaviour.
- Phase rate criterion: avoid PIO's (e.g. Lockheed F-22).

Gibson criterion

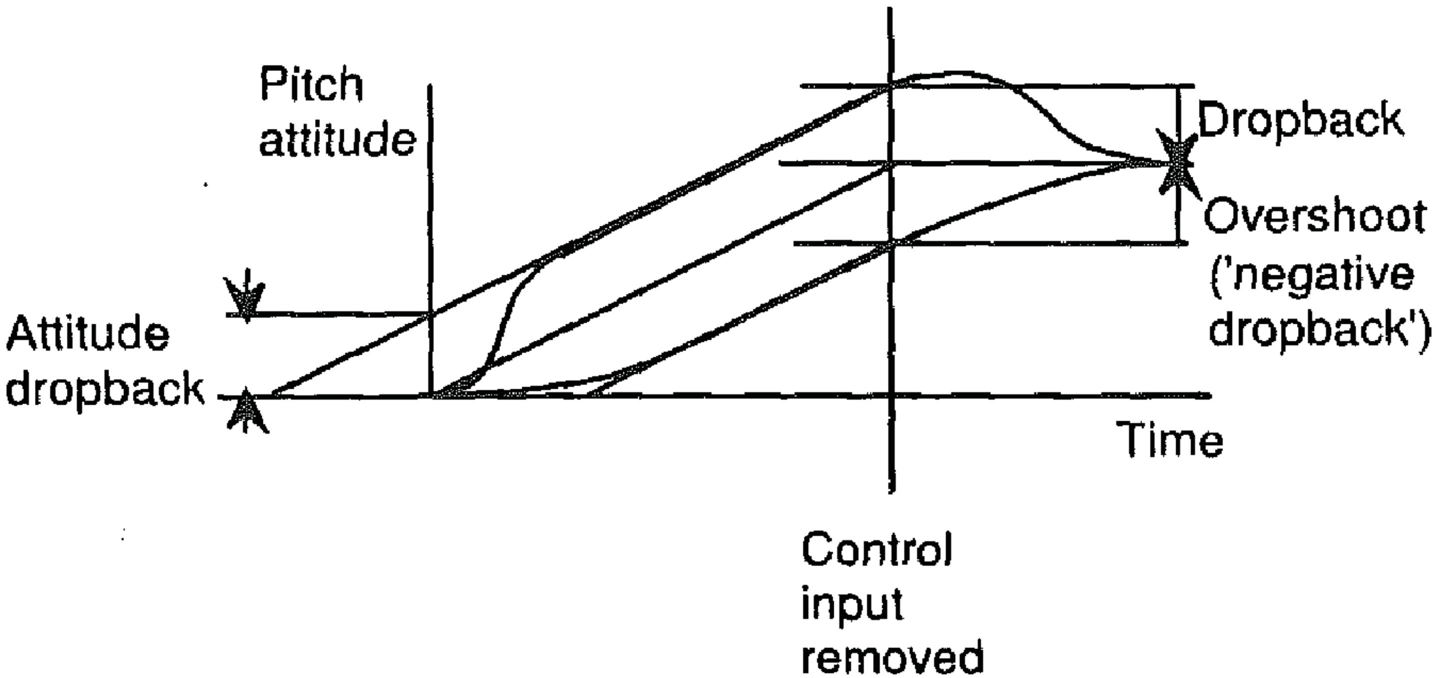


Figure 3: Gibson's Dropback

Gibson criterion

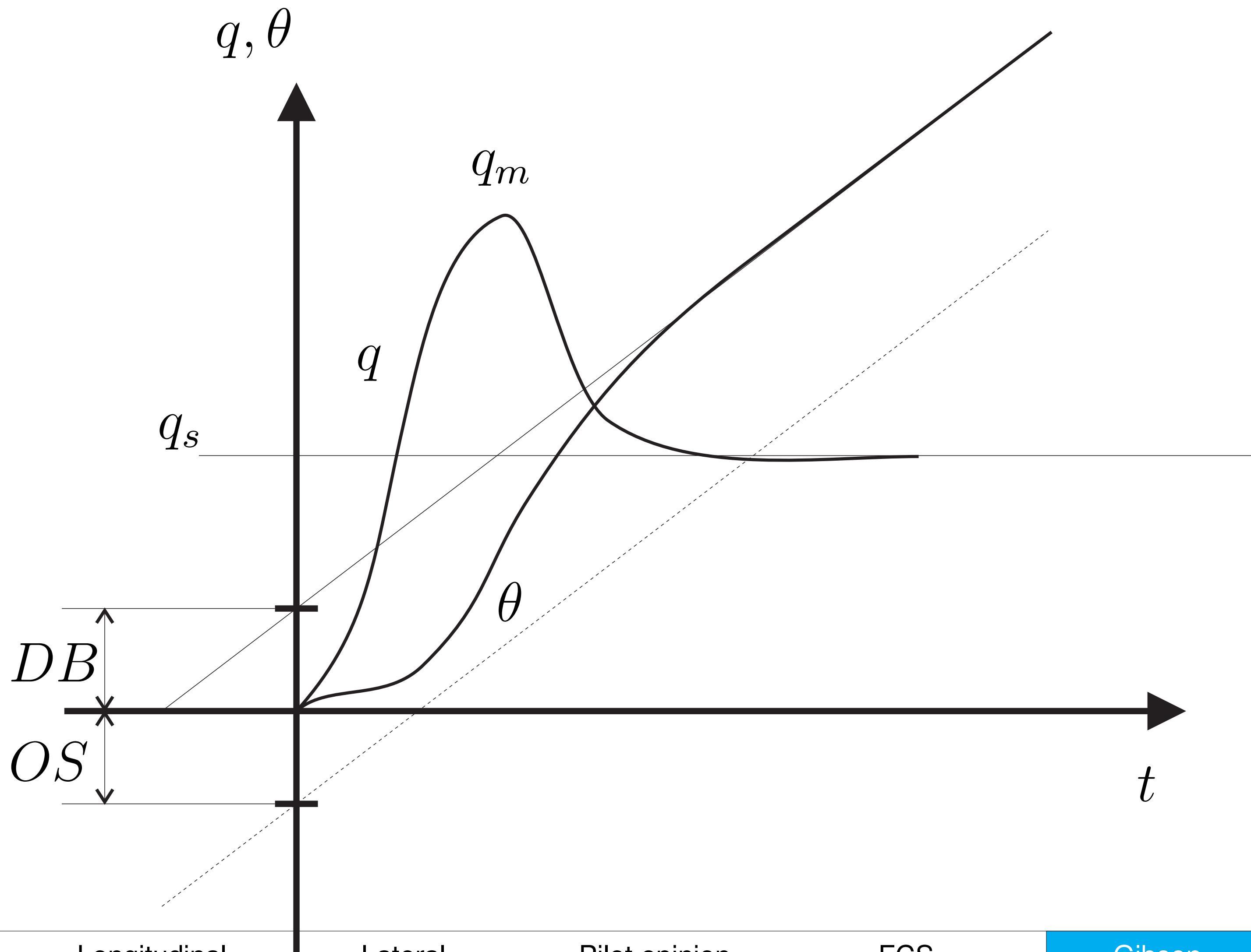


Figure 4: Gibson's Dropback

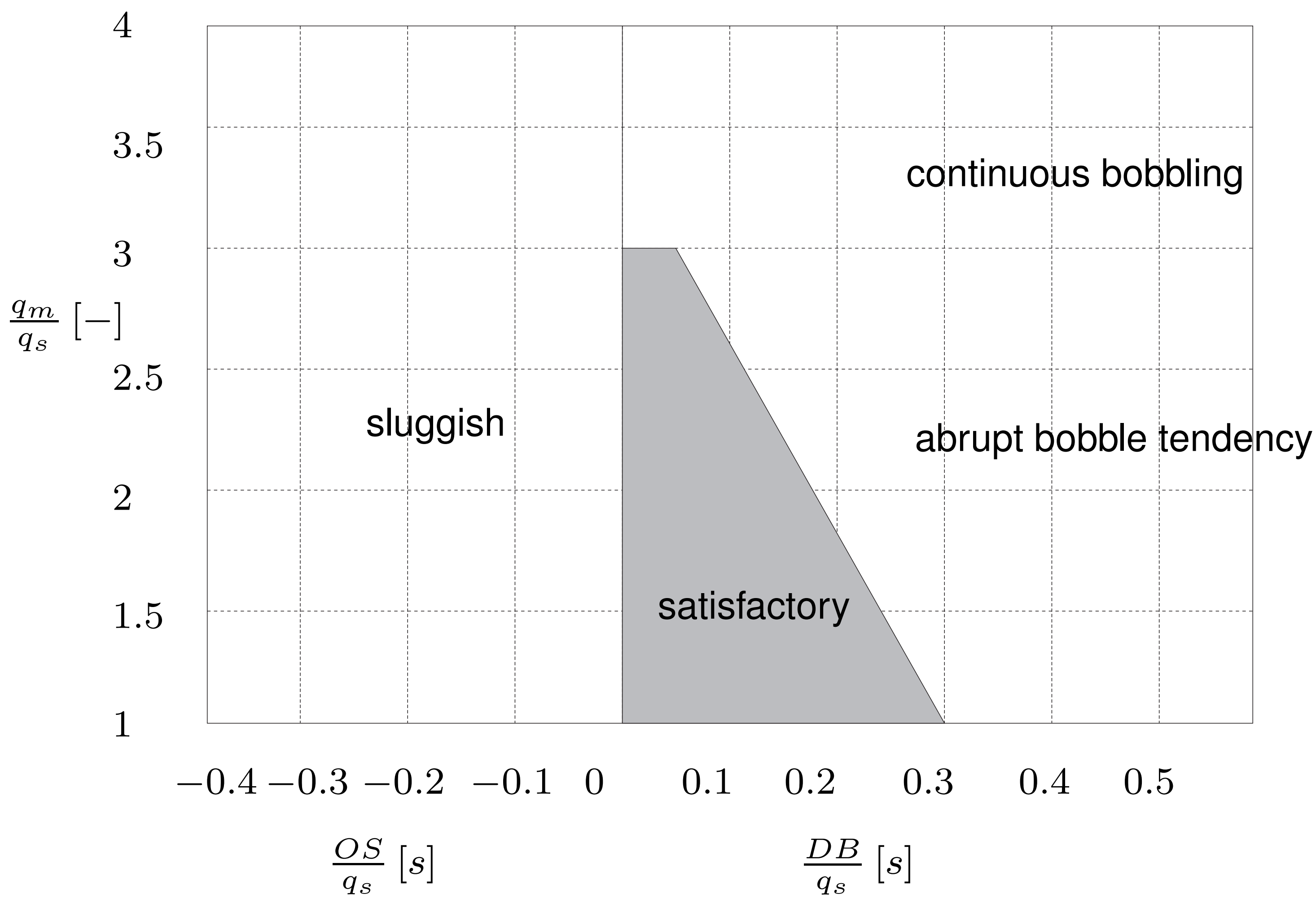
Gibson criterion

Dropback criterion

Relevant parameters:

- q_m: Maximum pitch rate.
- q_s: Steady state value of pitch rate.
- $\frac{q_m}{q_s}$: Pitch rate overshoot ratio.
- DB: Dropback, negative transition.
- OS: Overshoot, positive transition.

Gibson criterion



Gibson criterion

Dropback criterion

- No dropback if $\frac{q_m}{q_s} \leq 1.0$, lower part of satisfactory region not attainable.
- Zero dropback (DB=0) is optimal, minimal DB preferable over OS.
- Acceptable pitch rate overshoot values: $1.0 \leq \frac{q_m}{q_s} \leq 3.0$.

Gibson criterion

Phase rate criterion

- Pilot Induced Oscillations (PIO's): addition of pilot leads to instabilities.
- Likelihood of PIO's: degree of gain/phase compensation required.
- PIO probability analysis: closed loop gain and phase analysis close to resonant frequency of human pilot.

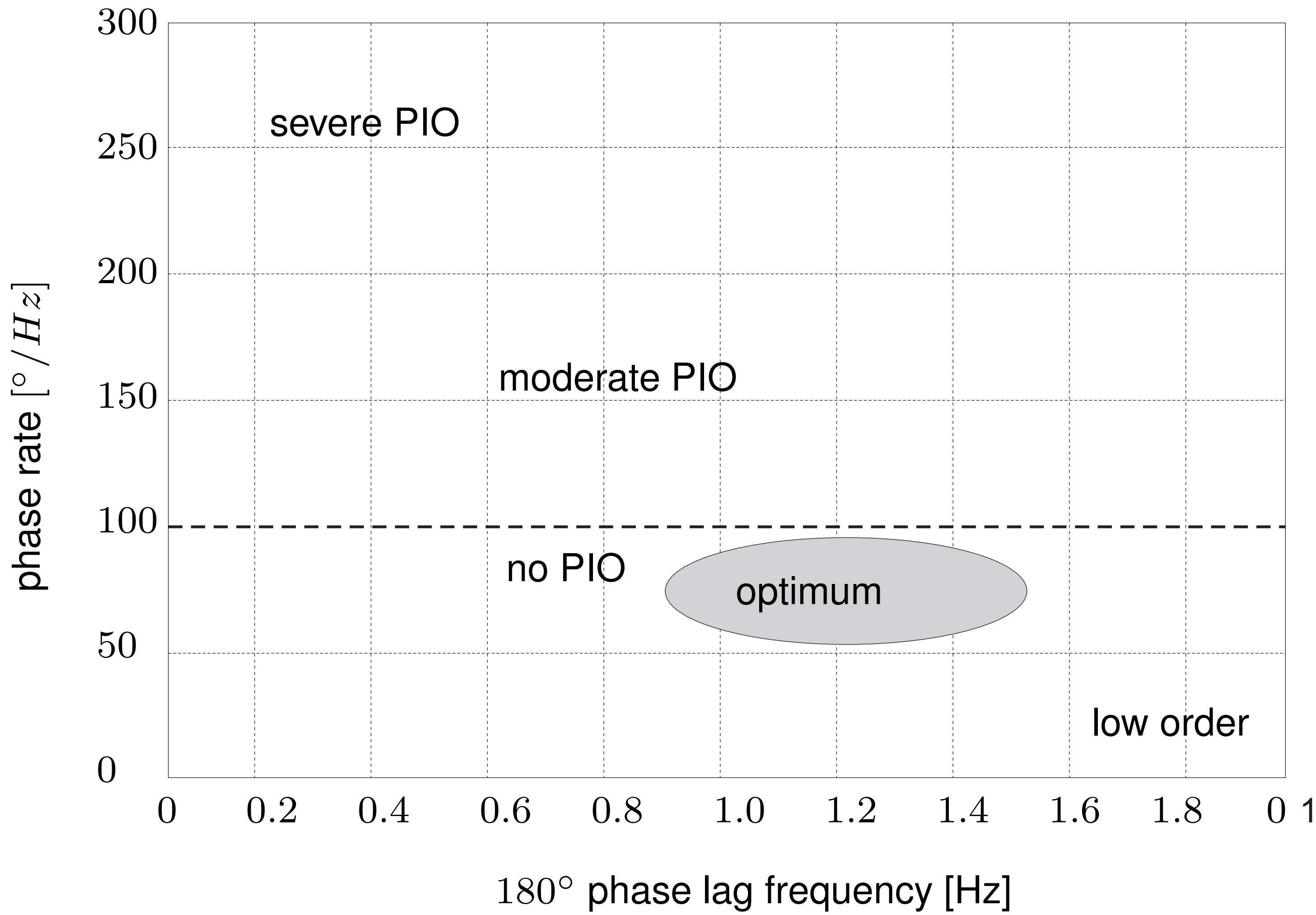
Gibson criterion

Phase rate criterion

Requirements:

- The 180° phase lag frequency: $\omega_{\phi=180^\circ} \approx 1Hz$.
- The phase rate at 180° phase lag: $\left(\frac{\partial\phi}{\partial\omega}\right)_{\phi=-180^\circ} \leq 100\frac{^\circ}{Hz}$

Gibson criterion



Summary

- Flying qualities requirements:
  - Differ according to airplane class
  - Distinction between flight phase categories.
  - Levels according to failure probability.
- Pilot opinion: Cooper-harper rating scale, flight test evaluation.
- Flight Control System influences flying qualities.
- Gibson dropback criterion: transient behaviour and PIO's.

What's next?

- Static/dynamic stability augmentation.
- Longitudinal autopilot modes.
- Lateral autopilot modes.

AE4-301 Automatic Flight Control System Design

Ewoud Smeur
Delft University of Technology - AE - C&S

September 23, 2020

AE4-301 Automatic Flight Control System Design

Ewoud Smeur
Delft University of Technology - AE - C&S

September 30, 2020

Course structure

- Introduction automated flight control and recap flight dynamics.
- Control theory.
- Performance and handling qualities.
- **Static/dynamic stability augmentation.**
- Longitudinal autopilot modes.
- Lateral autopilot modes.

Lecture 10 structure

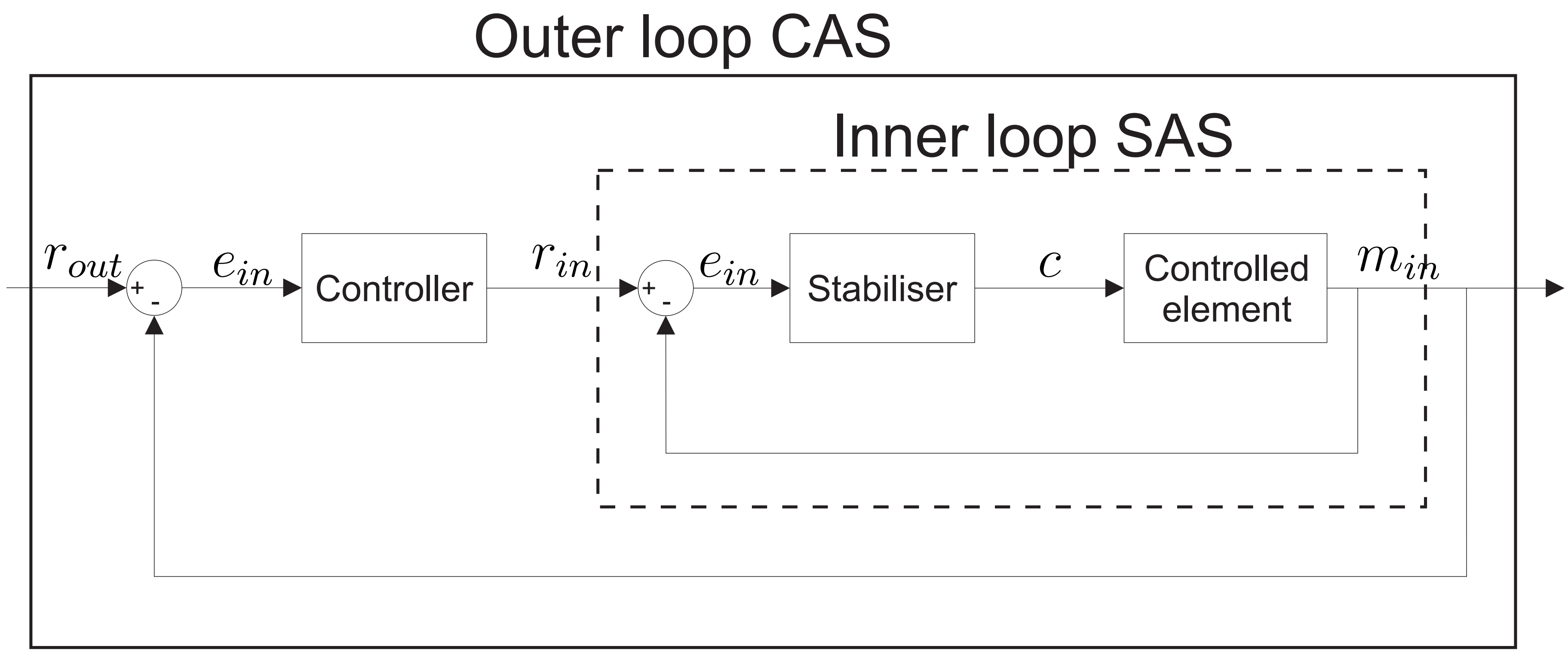
- Introduction.
- Pitch dampers.
- Phugoid dampers.
- Yaw dampers.
- Summary.

Introduction

Category	condition	tool
Static stability	$C_{m_\alpha} < 0$ $C_{n_\beta} > 0$	static SAS
Dynamic stability	eigenmotions stable $Re\left(\lambda\right) < 0$	dampers

Introduction

Position of the Stability Augmentation System in the global Flight Control System:



Introduction

- A Stability Augmentation System (SAS) augments open loop static/dynamic stability of the aircraft.
- SAS is necessary when inherent open loop characteristics parameters of the aircraft ($\omega, \zeta, ...$) do not comply with flying qualities requirements.
- Inner loop must be stable before continuing with the outer loop Control Augmentation System(CAS).

Pitch damper

High performance aircraft at low speed and high altitude can exhibit a rapid deterioration of the short period damping. Using a pitch damper improves the short period behaviour.

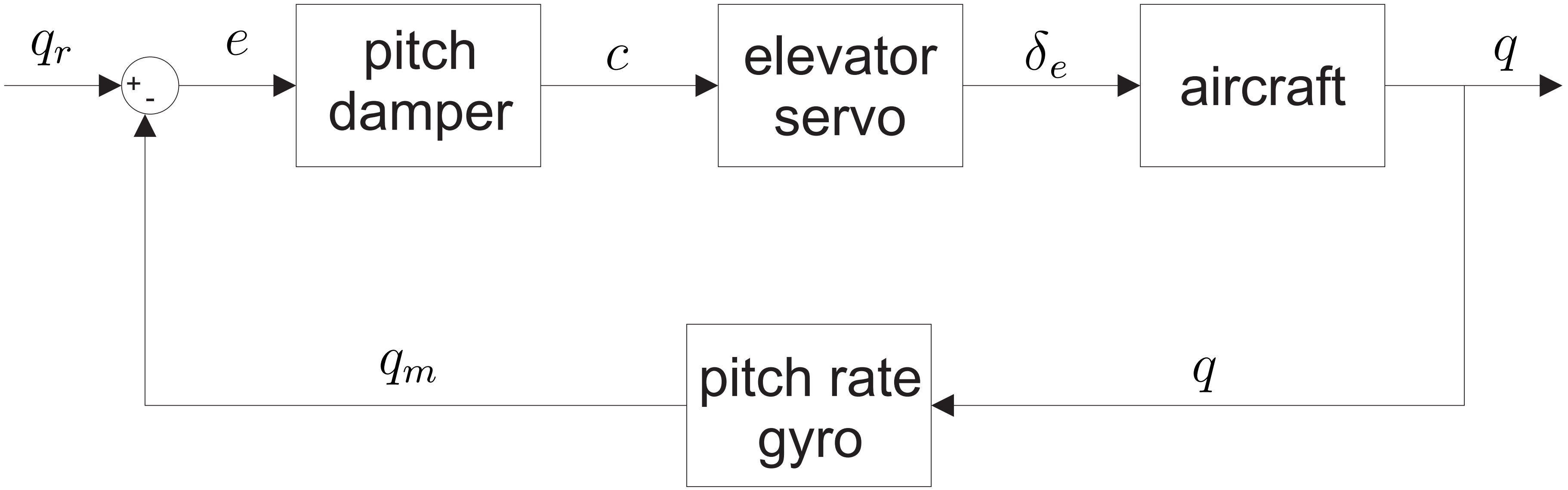
V[km/h]	$\omega_{n_{sp}}$	ζ_{sp}
1000	1.51	0.36
700	1.17	0.33
300	0.78	0.20

altitude h=12000 m

See **Homework assignment 10**

Pitch damper

Pitch rate feedback loop:



Pitch damper

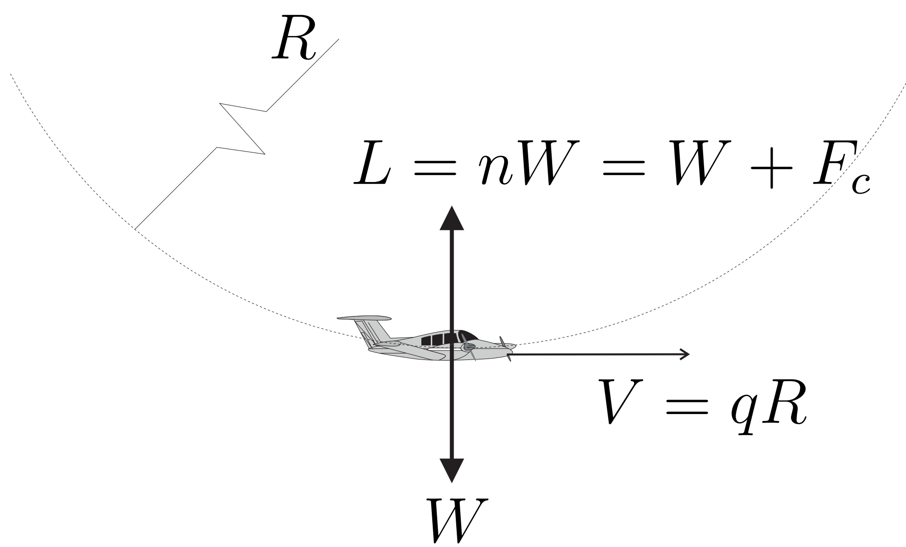
What about the reference input signal q?

- The pitch damper will force any pitch rate to zero, even when we want the aircraft to pull-up or push-over.
- There are two ways to prevent this from happening:
  - Compute a desired pitch rate for an intended manoever and feed this to the reference input.
  - Place a washout filter in the feedback loop, which limits the damper action in time.

Pitch damper

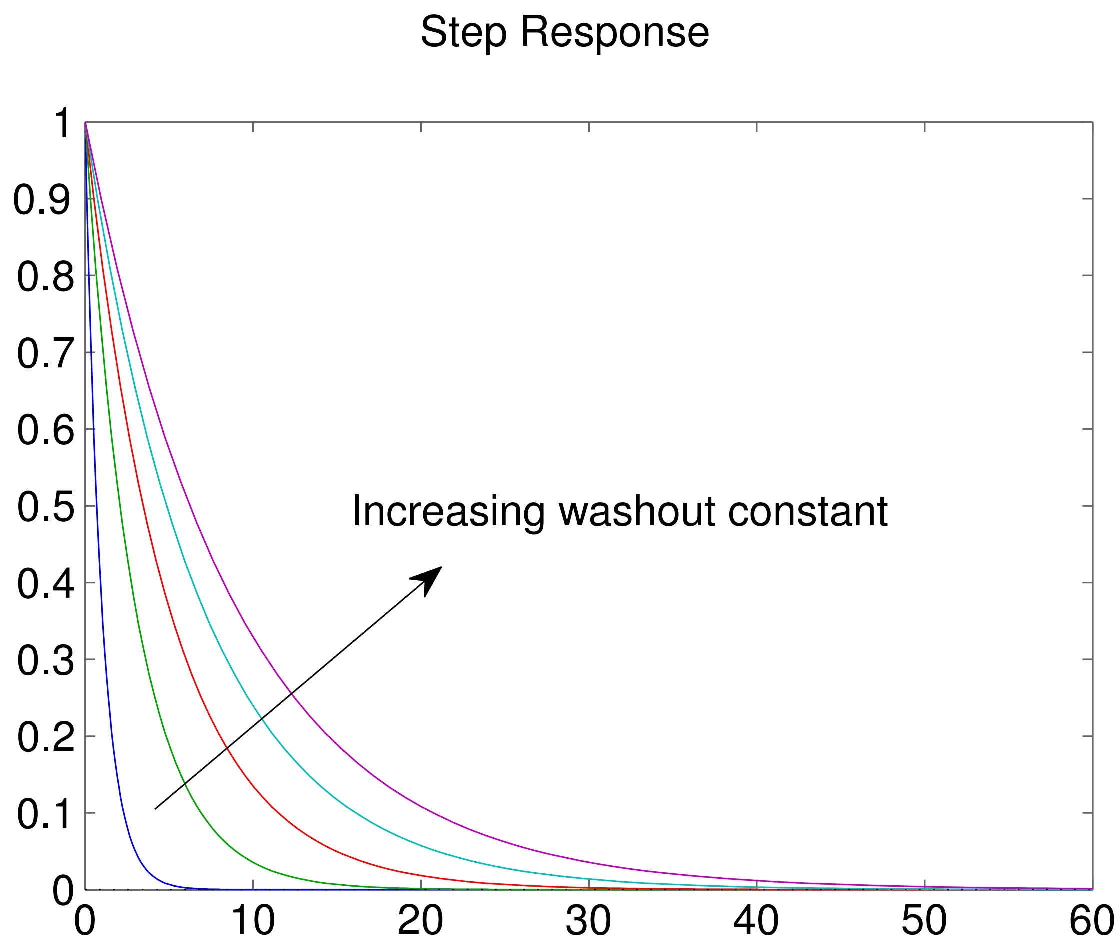
Compute desired pitch rate:

$$L = nW = W + F_c$$
$$nW = W + mq^2R$$
$$nW = W + mVq$$
$$nmg = mg + mVq$$
$$(n-1)\,g = Vq$$
$$q = \frac{g}{V}\,(n-1)$$



Pitch damper

Pitch rate washout filter: $H_{washout} = \frac{\tau s}{\tau s+1}$



Pitch damper

Pitch rate washout filter:

$$H_{washout} = \frac{\tau s}{\tau s + 1}$$

- τ very small: pitch damper will not work.
- τ very large: pitch damper will fight against the pilot setting up a pull-up or push-over maneuver.

Pitch damper

Elevator servo model:

- The actuator output always lags behind the input signal.
- This can be represented by a first order lag transfer function.

$$H_{servo} = \frac{K_{servo}}{1 + T_{servo}s}$$

T_{servo} depends on the type of actuator. Typical values are:

- Hydraulic: $T_{servo} = 0.05 - 0.1s$
- Electric: $T_{servo} = 0.25s$

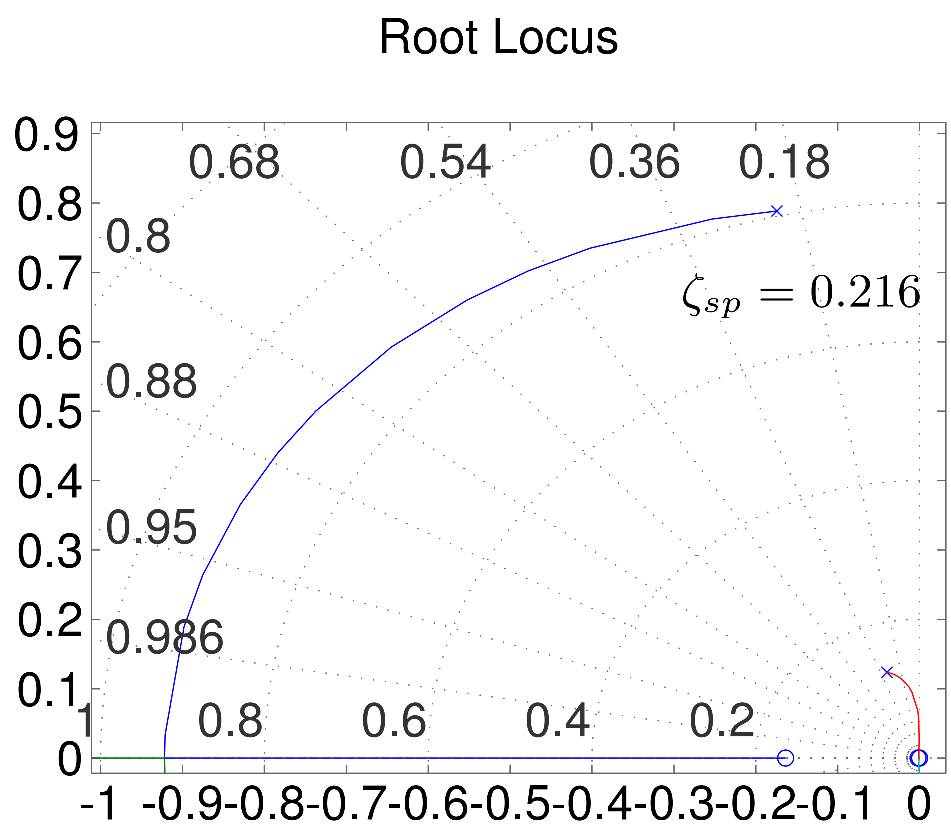
Pitch damper

Example: F-16, h=12km, V=330km/h

- **(Simulink demo)** F-16 model linearisation **(AE4-301P)**.
- **(Simulink demo)** Analysis of open loop response elevator to pitch rate.
- **(sisotool demo)** Design of pitch damper gain.
- **(Simulink demo)** Analysis of closed loop response elevator to pitch rate.

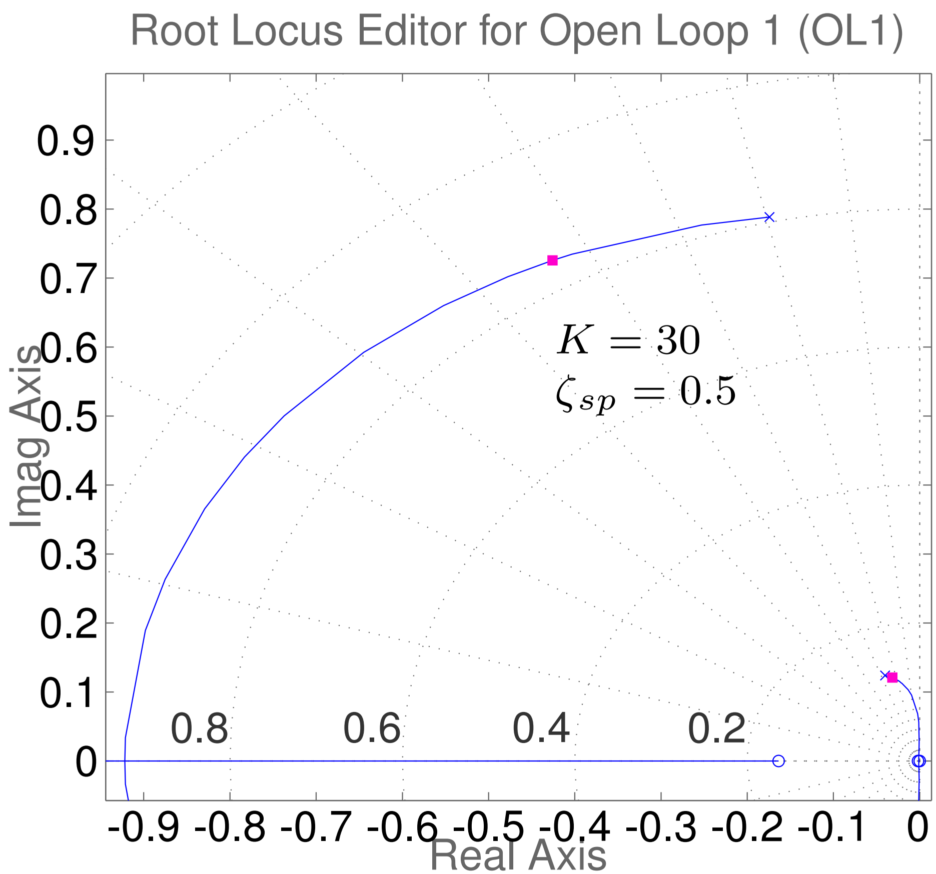
Pitch damper

Example: F-16, h=12km, V=330km/h



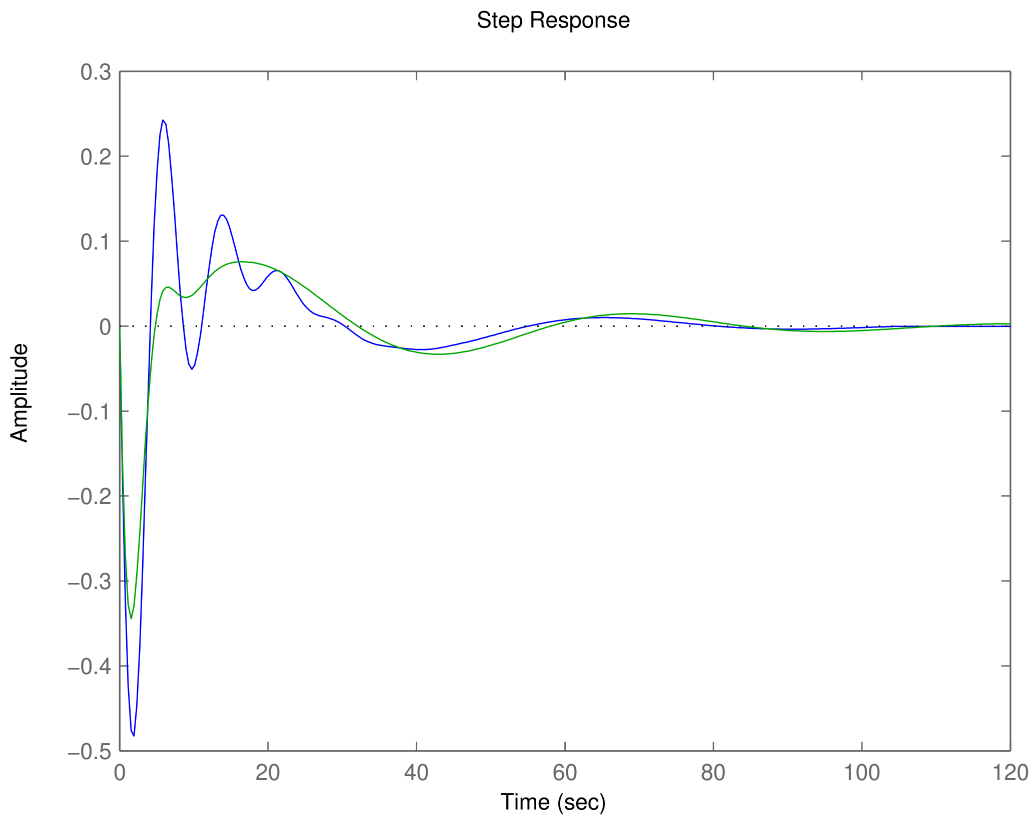
From req.: $0.3 < \zeta_{sp} < 2$ and $\zeta_{ph} > 0.04$. $\rightarrow$ $\zeta_{\mathbf{sp}}$ **is too low!**

Pitch damper



Set feedback gain to $K = 30$. $\rightarrow \zeta_{\mathbf{sp}}$ now satisfied.

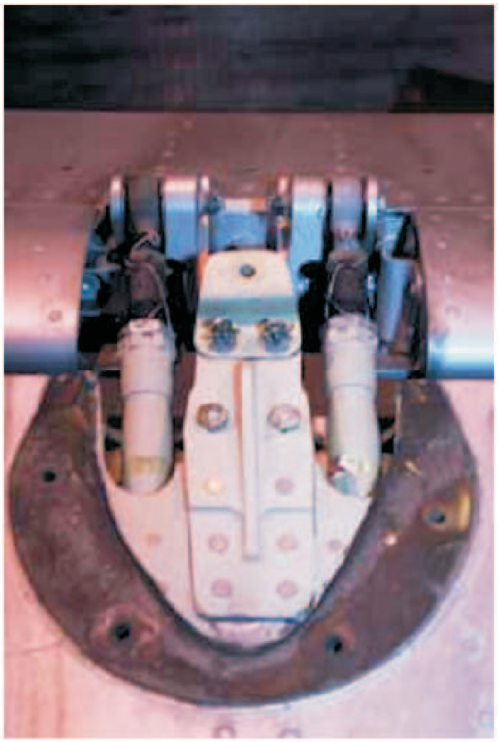
Pitch damper

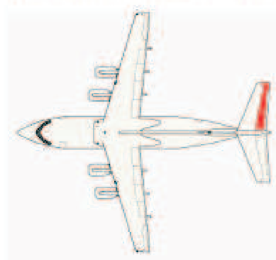


Open loop and closed loop step response: damping is improved.

Pitch damper

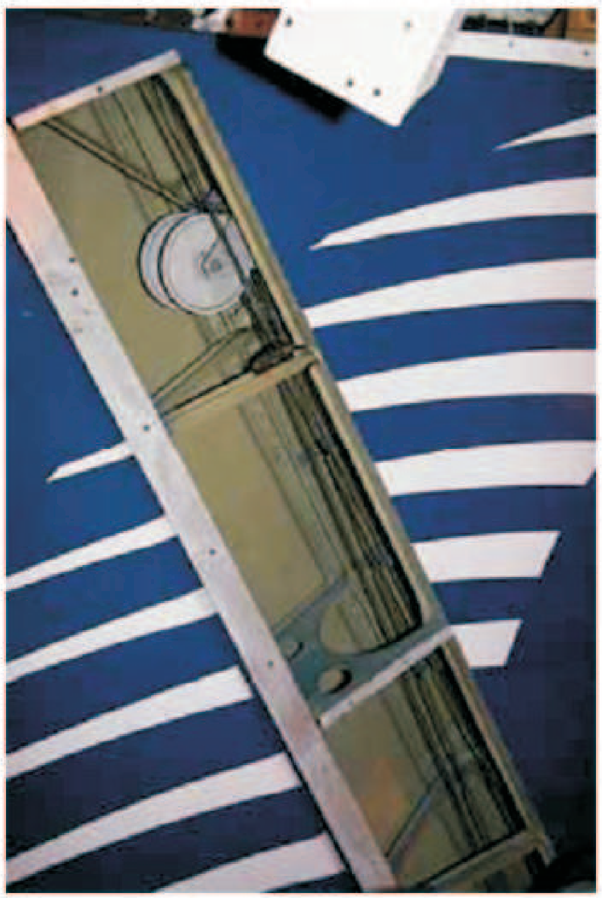
Elevator steering system of BAe RJ85:





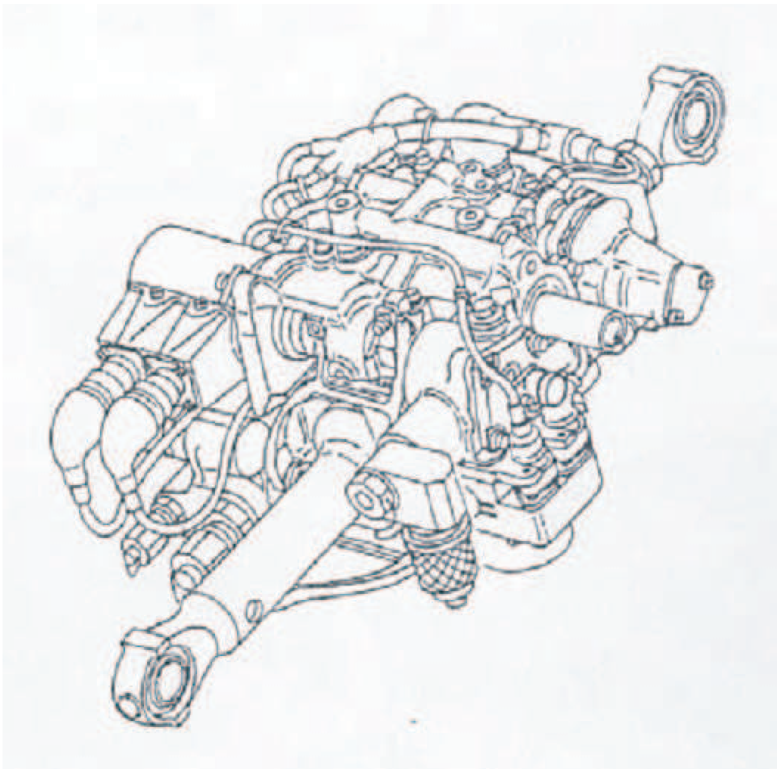
Pitch damper

Elevator steering system of BAe RJ85:



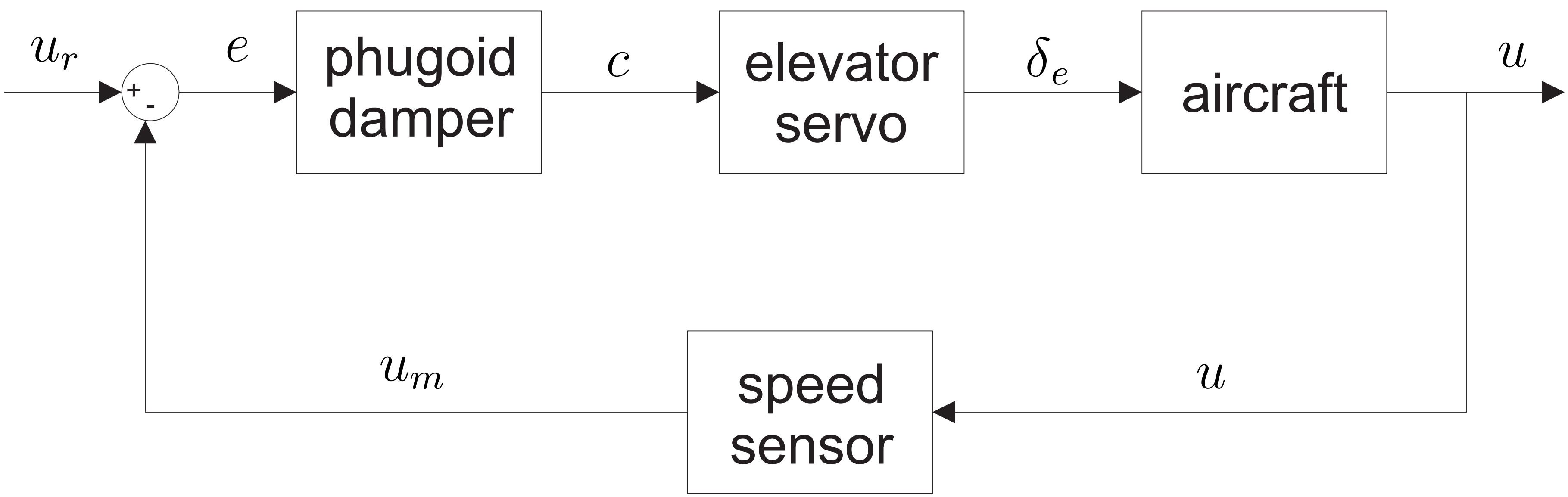
Pitch damper

Elevator servos of Airbus A320:



Phugoid damper

Sometimes the phugoid eigenmotion appears to be unstable or it does not satisfy the airworthiness regulations. In these cases, a phugoid damper can be used to improve the open loop phugoid behavior.



Phugoid damper

Sensor and servo models:

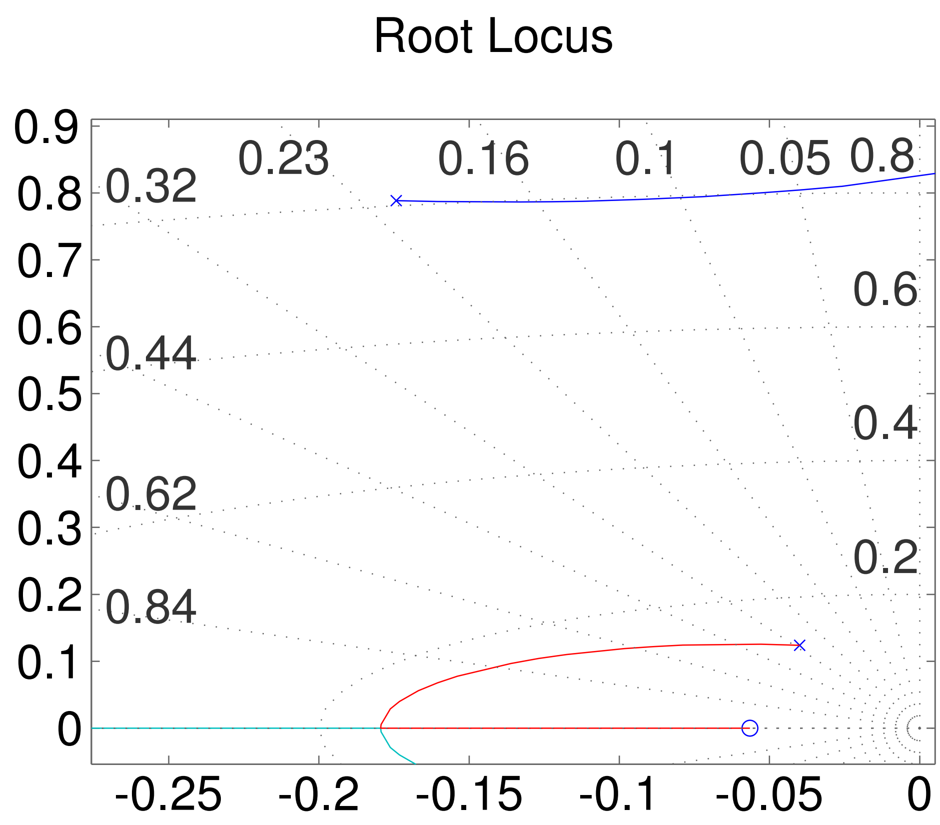
- Speed sensor: $H_{speed\ sensor} \approx 1$
- Elevator servo: $H_{servo} \approx \frac{1}{0.05s+1}$

What about reference input signal u?

- Feed computed velocity to the phugoid damper reference input.
- Use a washout circuit in the feedback path.

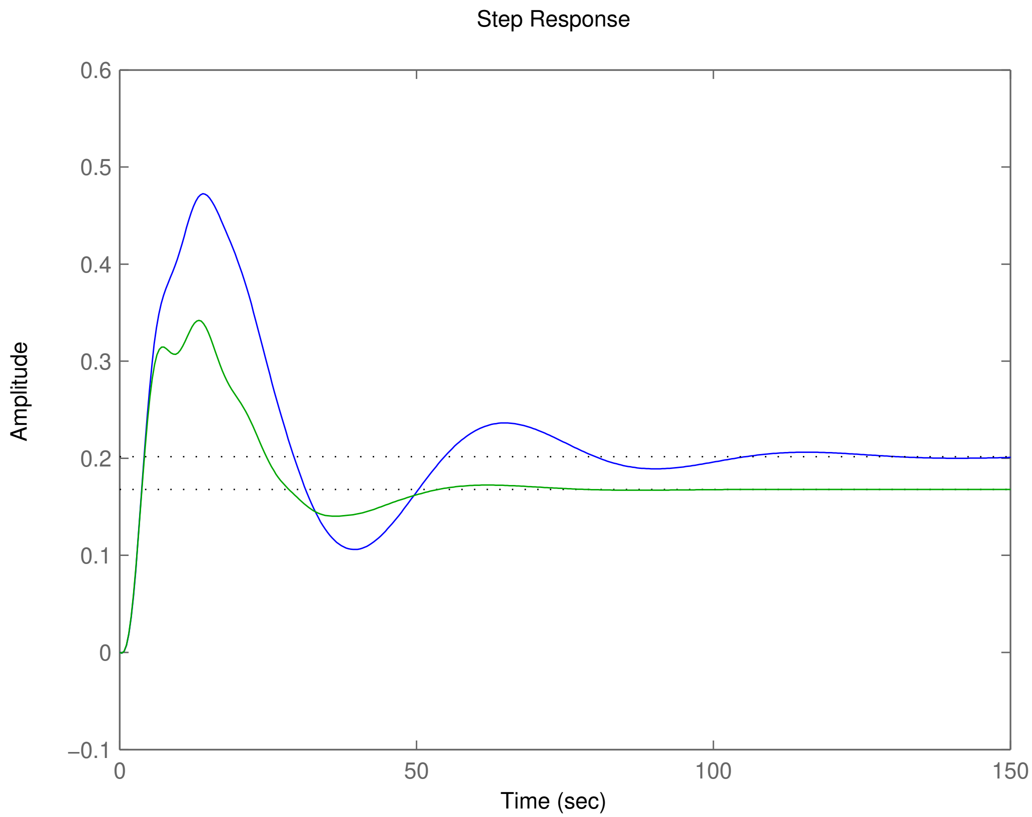
Phugoid damper

Example: F-16, h=12km, V=330km/h



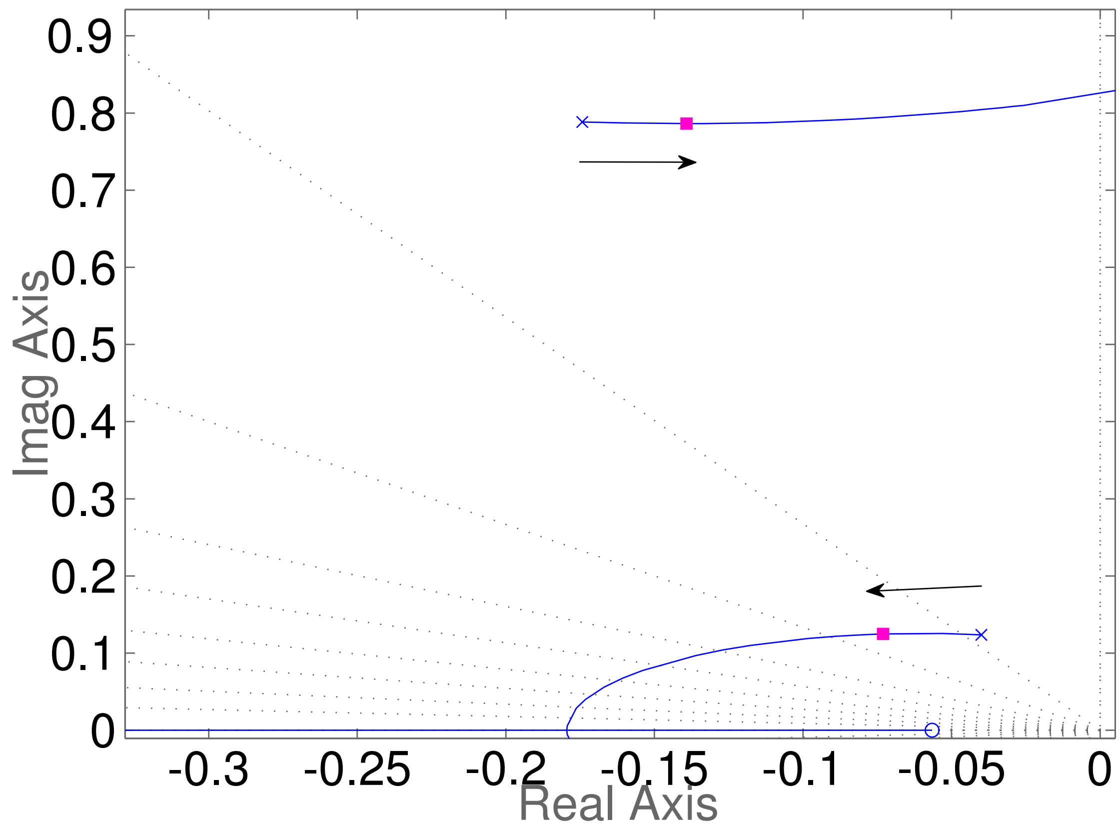
Phugoid damper

Increased phugoid damping by velocity feedback gain. Open and closed loop step response:



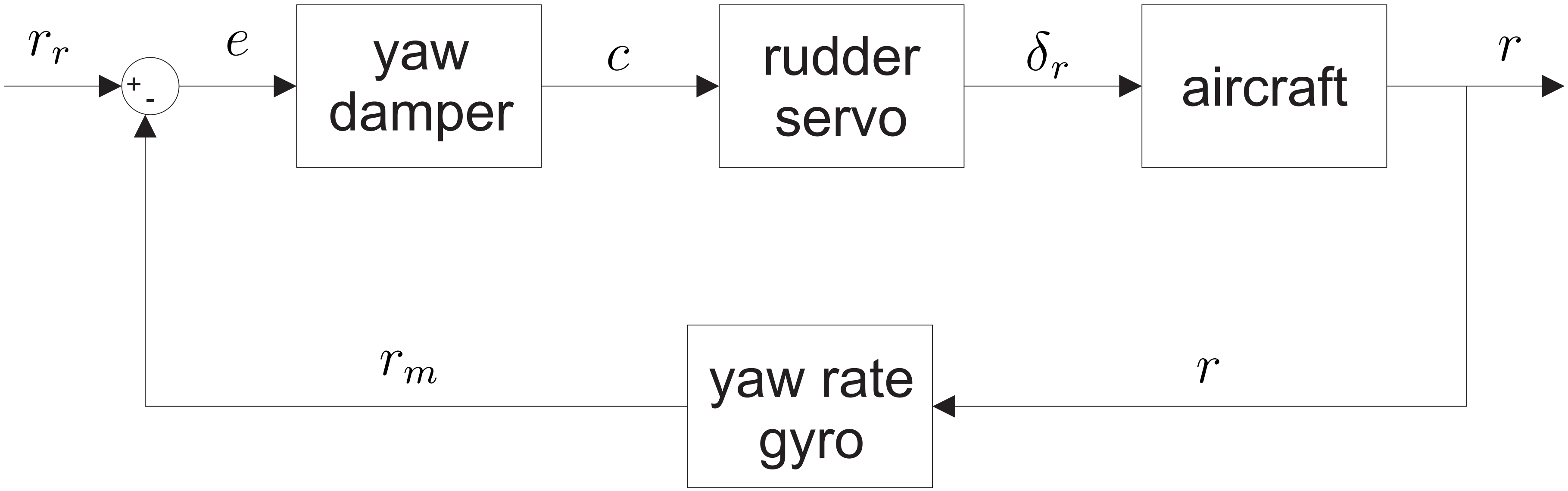
Phugoid damper

But, the velocity feedback gain also influences short period damping! Check the root locus plot:



Yaw damper

High performance aircraft at low speed and high altitude: rapid deterioration of Dutch roll damping. A yaw damper can improve the Dutch roll behaviour.



Yaw damper

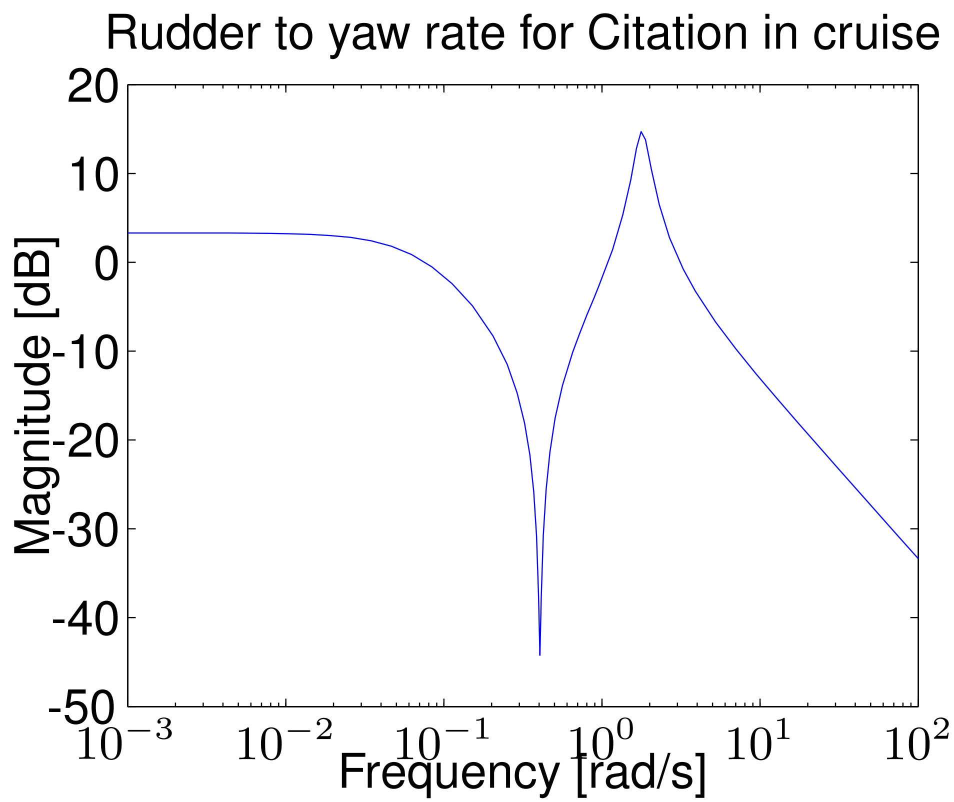
Yaw rate gyro model:

- Gyro is good in low frequency measurements.
- Deterioration in high frequency regions.
- Model as a low pass filter transfer function

$$H_{gyro} = K_{gyro} \frac{1}{s + \omega_{br}}$$

Where the gyro break frequency ω_{br} is selected such that the important aircraft yaw dynamics can be captured.

Yaw damper



So the gyro break frequency ω_{br} can be set somewhere at > 5 rad/s.

Yaw damper

$$H_{gyro} = K_{gyro} \frac{1}{s + \omega_{br}}$$

Worksheet: Plot the bode diagram of H_{gyro} for a gyro break frequency $\omega_{br} = 50 rad/s$ and $K_{gyro} = 40$

Yaw damper

$$H_{gyro} = K_{gyro} \frac{1}{s + \omega_{br}}$$

Worksheet: Plot the bode diagram of H_{gyro} for a gyro break frequency $\omega_{br} = 50 rad/s$ and $K_{gyro} = 40$

$$20 \log |H_{gyro}(j\omega)| = 20 \log \left| K_{gyro} \frac{1}{j\omega + \omega_{br}} \right| = -20 \log \left(\frac{\sqrt{\omega^2 + \omega_{br}^2}}{K_{gyro}} \right)$$

$$20 \log |H_{gyro}(0)| = -20 \log \left| \frac{50}{40} \right| = -1.94$$

$$20 \log |H_{gyro}(j50)| = -20 \log \left| \frac{50\sqrt{2}}{40} \right| = -4.94$$

Yaw damper

$$H_{gyro} = K_{gyro} \frac{1}{s + \omega_{br}}$$

Worksheet: Plot the bode diagram of H_{gyro} for a gyro break frequency $\omega_{br} = 50 rad/s$ and $K_{gyro} = 40$

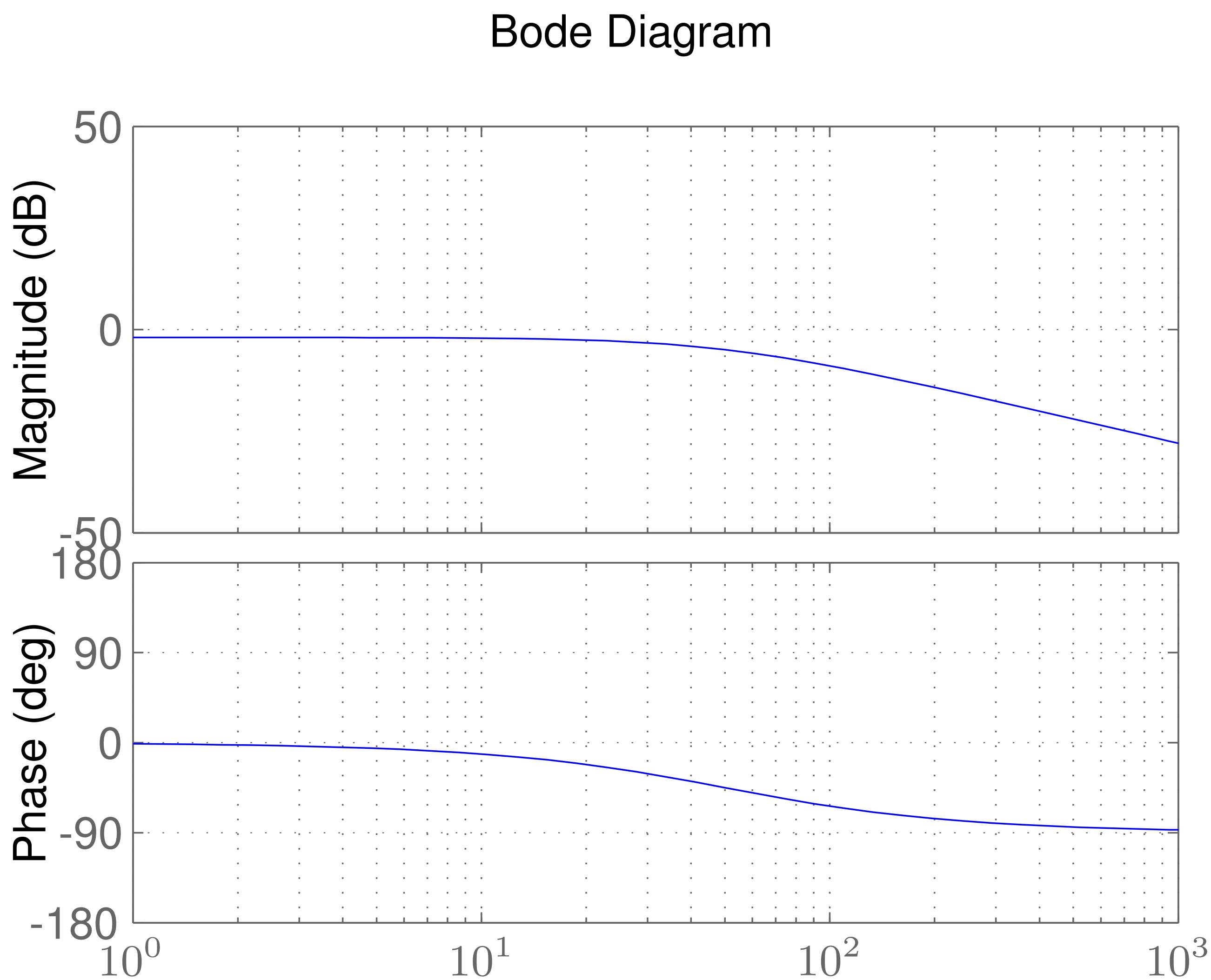
$$\angle H_{gyro}(j\omega) = \arctan \frac{\Im(H_{gyro}(j\omega))}{\Re(H_{gyro}(j\omega))}$$

$$\angle H_{gyro}(j\omega) = \angle \text{numerator} - \angle \text{denominator}$$

$$\angle H_{gyro}(j\omega) = -\arctan \frac{\omega}{\omega_{br}}$$

$$\angle H_{gyro}(0) = 0^o, \quad \angle H_{gyro}(j\infty) = -90^o, \quad \angle H_{gyro}(j50) = -45^o$$

Yaw damper


Bode Diagram
Magnitude (dB)
50
0
-50
Phase (deg)
180
90
0
-90
-180
10^0
10^1
10^2
10^3

Yaw damper

What about the reference input signal r?

- The yaw damper will force any yaw rate to zero, even when we want the aircraft to make a turn.
- There are two ways to prevent this from happening:
  - Compute a desired yaw rate for an intended turn and feed this to the reference input.
  - Place a washout filter in the feedback loop, which limits the damper action in time.

Yaw damper

Compute desired yaw rate

$$r = \dot{\psi} \cos \theta \cos \phi - \dot{\theta} \sin \phi$$

which requires measurements of:

- Pitch attitude angle θ.
- Bank angle ϕ.
- Rate of change of heading $\dot{\psi}$.
- Rate of change of pitch $\dot{\theta}$.

In a steady turn with $\dot{\theta} = 0$, the heading rate is: $\dot{\psi} = \frac{g \tan \phi}{V}$

Yaw damper

Yaw rate washout filter:

$$H_{washout} = \frac{\tau s}{\tau s + 1}$$

- τ very small: yaw damper will not work.
- τ very large: yaw damper will fight against the pilot setting in a turn.

Yaw damper

Rudder servo model:

- The actuator output always lags behind the input signal.
- This can be represented by a first order lag transfer function.

$$H_{servo} = \frac{K_{servo}}{1 + T_{servo}s}$$

T_{servo} depends on the type of actuator. Typical values are:

- Hydraulic: $T_{servo} = 0.05 - 0.1s$
- Electric: $T_{servo} = 0.25s$

Yaw damper

Rudder servo model:

- $H_{servo} = \frac{4}{s+4}$

Worksheet:

- Sketch the time response of H_{servo} to a step input
- What is the magnitude of the response after 0.25 seconds?

Laplace transforms:

$$\begin{aligned} 1 &\rightarrow \frac{1}{s} \\ e^{-at} &\rightarrow \frac{1}{s+a} \end{aligned}$$

Yaw damper

Rudder servo model:

- $H_{servo} = \frac{4}{s+4}$

Laplace transforms:

$$\begin{array}{lll} 1 & \rightarrow & \frac{1}{s} \\ e^{-at} & \rightarrow & \frac{1}{s+a} \end{array}$$

$$U(s) = \frac{1}{s} \rightarrow Y(s) = \frac{4}{s+4}\frac{1}{s} = \frac{a}{s} + \frac{b}{s+4}$$

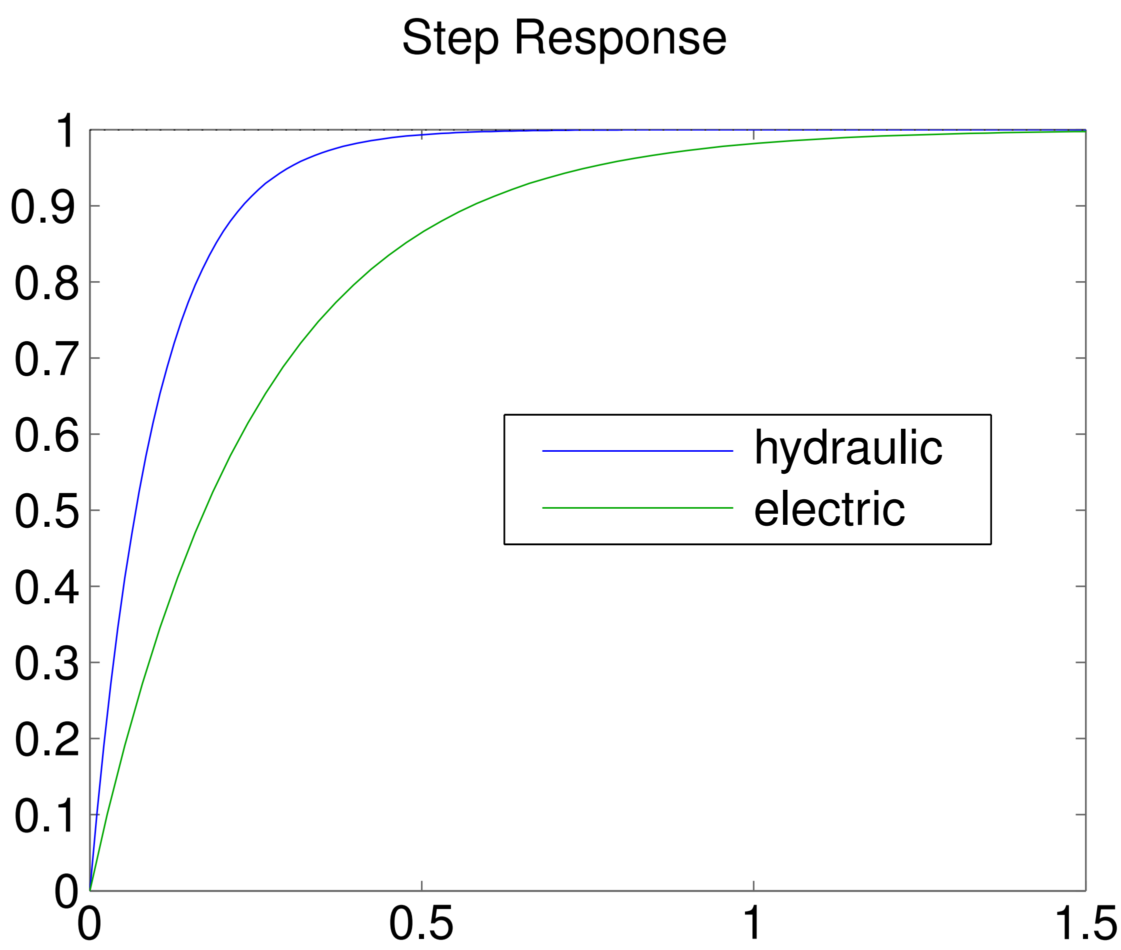
$$a(s+4) + bs = 4 \rightarrow a = 1, \quad b = -1$$

$$h_{servo}(t) = 1 - e^{-4t}$$

$$h_{servo}(0.25) = 1 - e^{-1} = 0.63$$

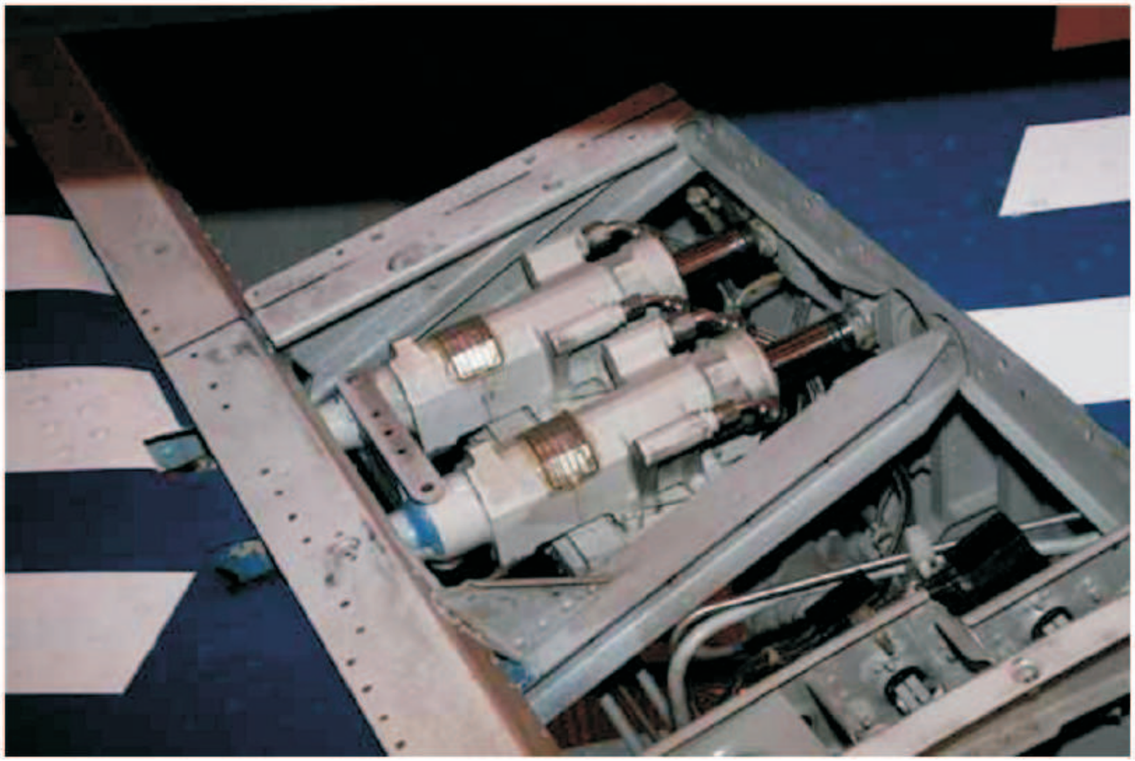
Yaw damper

Rudder servo model:



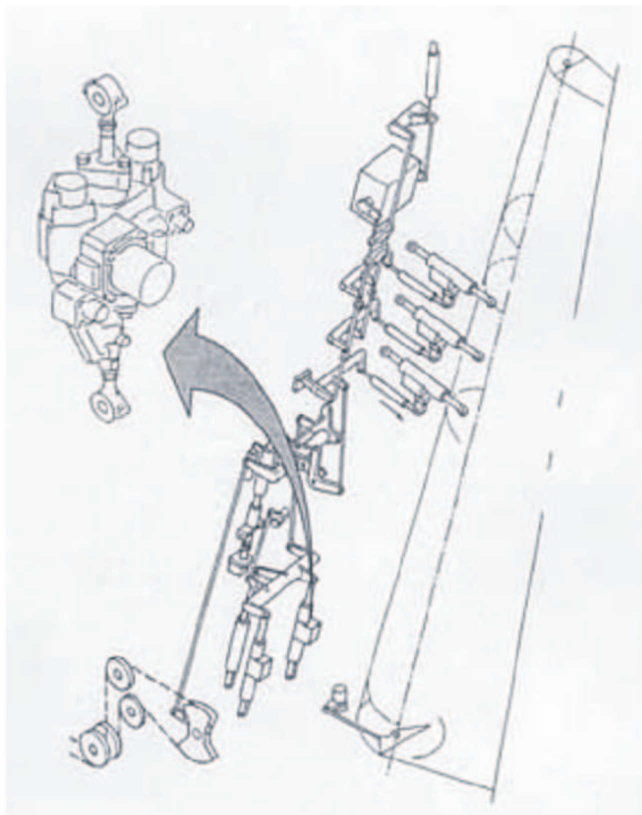
Yaw damper

Rudder servo of British Aerospace RJ85:



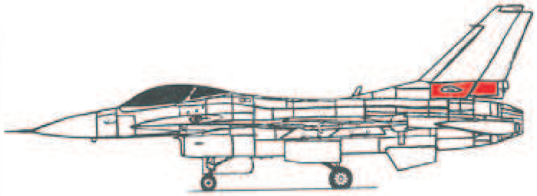
Yaw damper

Yaw damper servos location of Airbus A320:



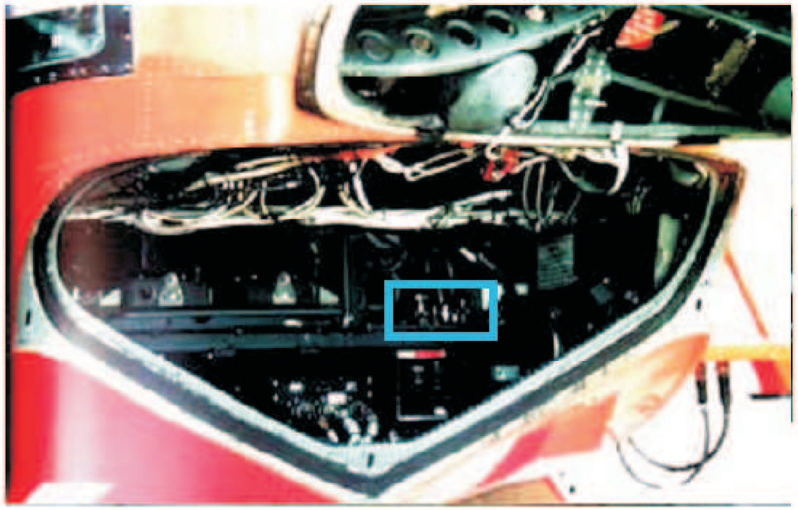
Yaw damper

Rudder servo of Lockheed F-16:



Yaw damper

Yaw rate gyro of Westland Sea King:



Yaw damper

Bunting UAV:

$$\frac{\frac{rb}{2V}}{\delta_r}\frac{\delta_r}{u_{servo}} = \frac{83.97\mathrm{s}^3+485.2\mathrm{s}^2+131.9\mathrm{s}+2275}{\mathrm{s}^5+9.528\mathrm{s}^4+37.04\mathrm{s}^3+182.1\mathrm{s}^2+489.9\mathrm{s}+1.702}$$

Yaw damper

Bunting UAV:

$$\frac{\frac{rb}{2V}}{\delta_r}\frac{\delta_r}{u_{servo}} = \frac{83.97\text{s}^3{+}485.2\text{s}^2{+}131.9\text{s} + 2275}{\text{s}^5{+}9.528\text{s}^4{+}37.04\text{s}^3{+}182.1\text{s}^2{+}489.9\text{s} + 1.702}$$

```
pole(H)
ans =
   0.3690 + 4.4042i
   0.3690 - 4.4042i
  -6.2635
  -3.9991
  -0.0035
```

Which eigenmodes are these?
Worksheet: ζ and ω_n of the Dutch Roll?

Yaw damper

```
pole(H)
ans =
   0.3690 + 4.4042i
   0.3690 - 4.4042i
  -6.2635
  -3.9991
  -0.0035
```

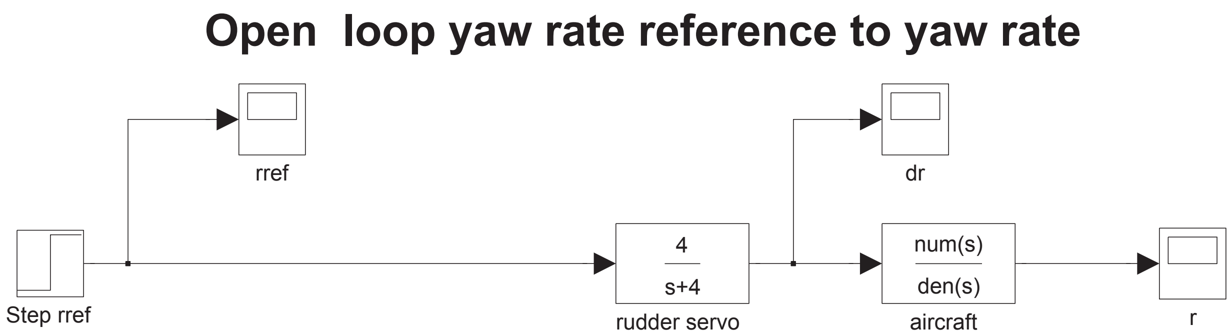
$$s = -\zeta\omega_n \pm i\omega_n\sqrt{1-\zeta^2}$$

$$\omega_n = \sqrt{(\zeta\omega_n)^2 + \left(\omega_n\sqrt{1-\zeta^2}\right)^2} = 4.4196\text{rad/s}$$

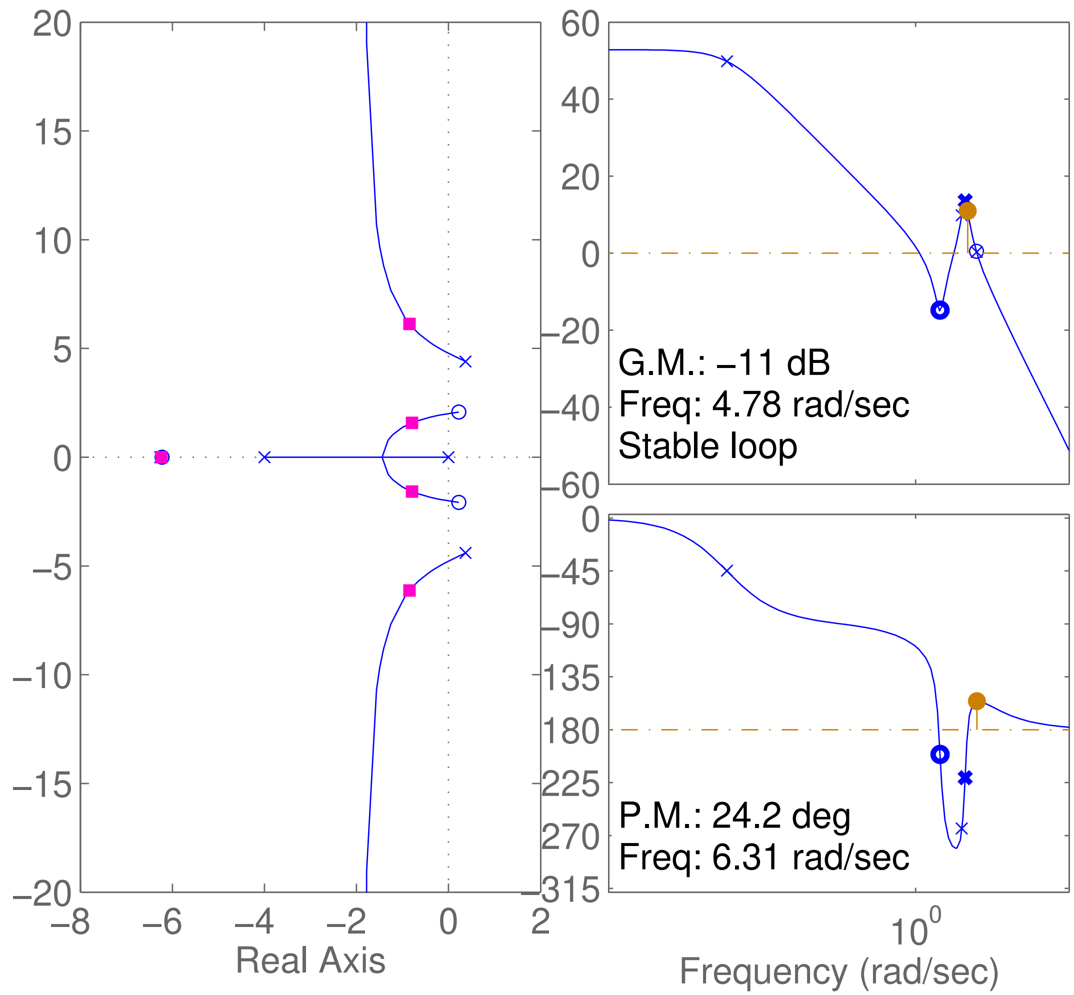
$$\zeta = \frac{\zeta\omega_n}{\omega_n} = \frac{-0.3690}{4.4196} = -0.0879$$

Yaw damper

- **(Simulink demo)** Analysis of open loop response yaw rate reference to yaw rate.
- **(sisotool demo)** Design of yaw damper gain.
- **(Simulink demo)** Analysis of closed loop response yaw rate reference to yaw rate.

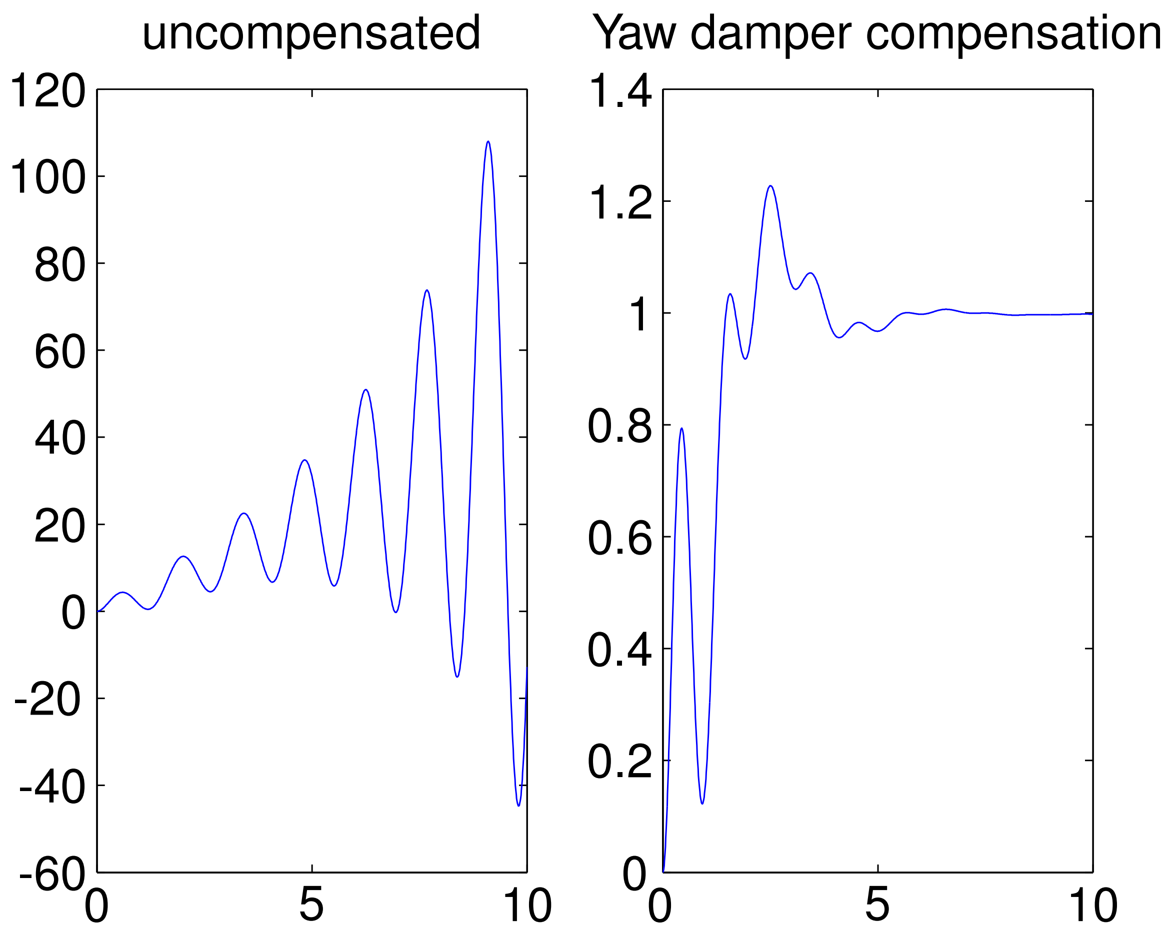


Yaw damper



20
15
10
5
0
−5
−10
−15
−20
−8 −6 −4 −2 0 2
Real Axis
60
40
20
0
−20
−40
−60
G.M.: −11 dB
Freq: 4.78 rad/sec
Stable loop
0
−45
−90
135
180
225
270
315
P.M.: 24.2 deg
Freq: 6.31 rad/sec
10^0
Frequency (rad/sec)

Yaw damper



uncompensated
120
100
80
60
40
20
0
-20
-40
-60
0
5
10
Yaw damper compensation
1.4
1.2
1
0.8
0.6
0.4
0.2
0
0
5
10

Yaw damper

The value of the yaw damper constant in cruise $K_{cruise} = 0.329$, does not guarantee compliance with airworthiness regulations in other flight conditions. In fact the yaw damper constant is different in each flight condition.

Flight condition	K
launch	0.439
climb	0.336
cruise	0.329
descent	0.336
recovery	0.6

Gain Scheduling is required.

Yaw damper

Gain scheduling

Gain value is varied based upon values of some relevant parameter like velocity or altitude. The range of these values coincides with the aircraft flight envelope.

Gain values are computed for a select number of points. Afterwards, interpolation techniques can be used to find the gains for parts of the flight envelope inbetween the computed points.

Summary

- SAS augments open loop aircraft stability
- Damper channels:
  - Pitch damper: $q \rightarrow \delta_e$
  - Phugoid damper: $u \rightarrow \delta_e$
  - Yaw damper: $r \rightarrow \delta_r$
- Damper reference signal:
  - Computed signal: either from pilot or autopilot, can be complex.
  - Washout filter: time limited SAS action, but simpler.

Summary

- Dampers improve the damping ratio of the eigenmodes.
- Damper models: Proportional controllers.
- Gains depend on flight condition: **Gain scheduling**.
- Dampers can be sensitive to servo and sensor time constants.
- Analyse time response to interpret the effect of different gains.

AE4-301P

Practical assignment AE4-301P

Objective:
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.

Case study: Lockheed F-16 Fighting Falcon.

- Compulsory for all C&O students.
- Homework Matlab/Simulink assignment.
- Teams of **max 3** students allowed.

Practical assignment AE4-301P

Background information:

- All information for AE4301p is posted on Brightspace page of AE4301.
- Lecture slides.
- Control theory book: Ogata, Nise, other.
- F-16 model documentation on Brightspace.

Practical assignment AE4-301P

Assignment setup:

1. Choice of flight condition.
2. Trim and linearisation.
3. Open loop analysis.
4. Pitch rate command system satisfying CAP/Gibson.
5. Glideslope and flare controller.
6. Reporting.

Practical assignment AE4-301P

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

Table 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 (you can decide who that will be).

Practical assignment AE4-301P

Reporting:

- Written report includes chosen procedures, answers to all questions, and numerical results.
- Your report must be delivered as a pdf file.
- All code that was written to generate the results in the report must be included in a zip-file.
- So 2 files (pdf+zip) in 1 submission!

Practical assignment AE4-301P

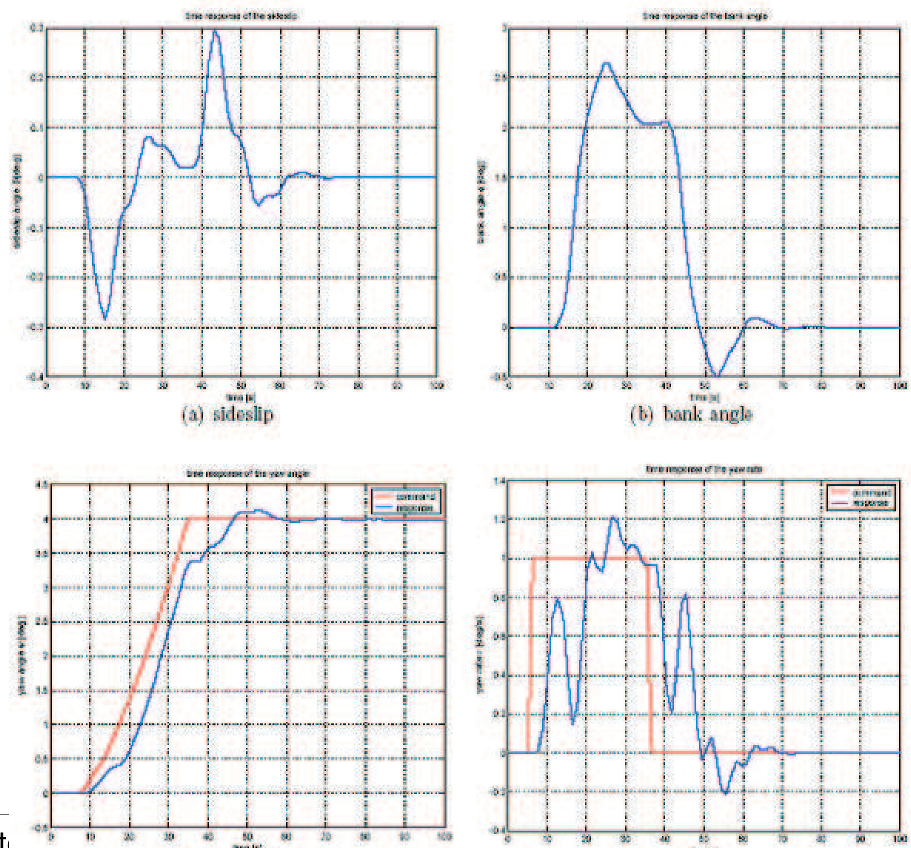
Reporting continued:

- Include Introduction, Conclusions, References etc. like in any technical report.
- Figure captions should clearly explain what is to be seen in the figure.

Practical assignment AE4-301P

Reporting continued:

- Points deducted for unreadable notations in figures:


(a) sideslip (b) bank angle
(c) yaw angle (d) yaw rate

Practical assignment AE4-301P

Deadline:

- Deadline for handing in the report is **17:00pm Sunday February 7th 2021**! (last day before start of Q3)
- All reports and code must be submitted through Brightspace. Submission instructions will be placed in the Assignments section of the Brightspace page for AE4301.

Practical assignment AE4-301P

Support:

- There will be a weekly walk-in question hour on Zoom for support.
- The walk-in hour is there to help with some technical problems (get the model running) and to receive some feedback when you show the things you have implemented.
- The walk-in hour is not there to give you the answers to the questions in the assignment!
- The walk-in question hours will start next week, until the submission deadline. Every Friday from 9:30-10:30.
- History shows that the last walk-in hour before the deadline is extremely busy and not everyone can be helped, so plan accordingly.
- Zoom link: https://tudelft.zoom.us/j/91798910643?pwd=VHJKOG9qbWx5MUhDeTJYK0xjT2c1Zz09
- Password: 040235

Practical assignment AE4-301P

First steps:

- Go to Brightspace AE4301 –> content –> AE4301P Assignment (will be up soon)
- Download and read the Practical Assignment document.
- Download F-16 Simulation model and documentation.
- Play around with the model and try to understand how it is constructed.
- Follow the steps in the Practical Assignment document.

AE4-301 Automatic Flight Control System Design

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September 30, 2020

AE4-301 Automatic Flight Control System Design

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October 4, 2020

Course structure

- Introduction automated flight control and recap flight dynamics.
- Control theory.
- Performance and handling qualities.
- **Static/dynamic stability augmentation.**
- **Exam training**
- Longitudinal autopilot modes.
- Lateral autopilot modes.
- Navigational autopilot systems.

Lecture 11 structure

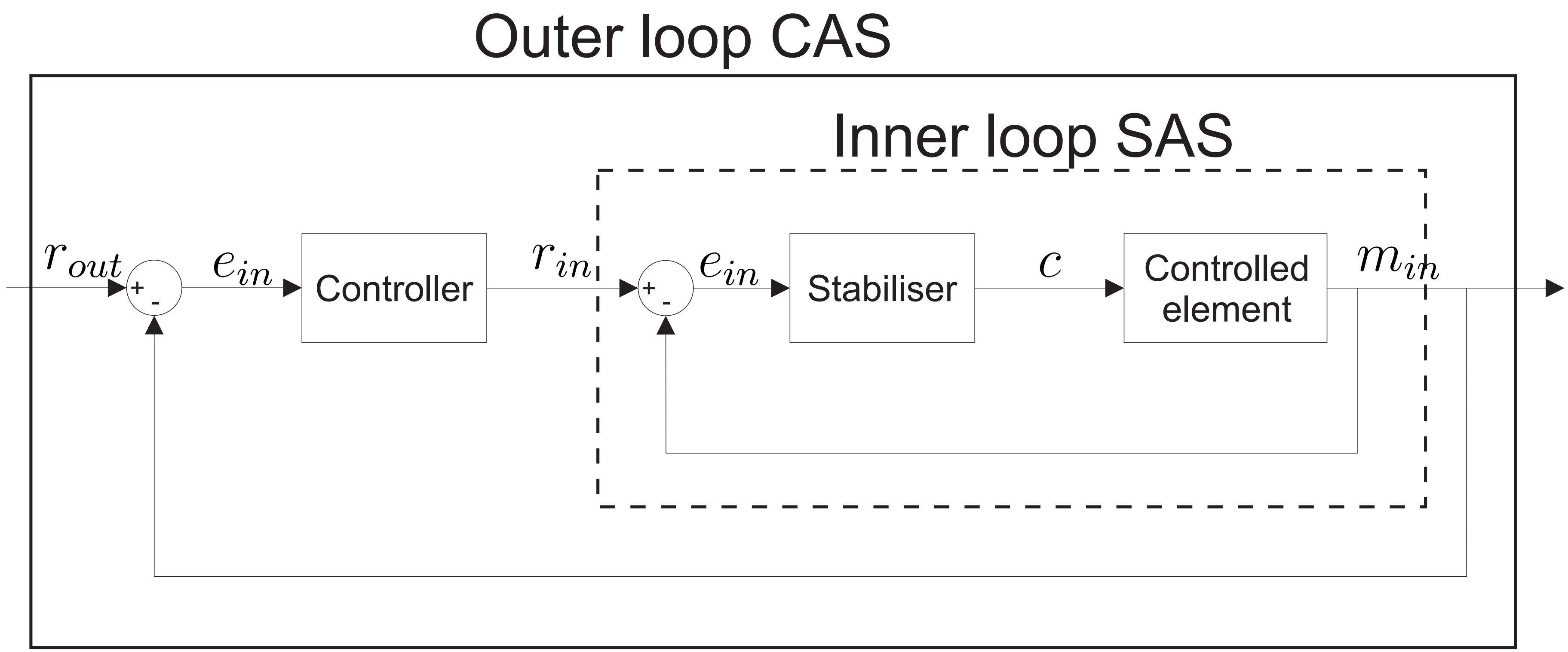
- Introduction.
- Angle of attack feedback
- Load factor feedback
- Sideslip feedback
- Summary
- Exam Training

Stability Augmentation Overview

Category	condition	tool
Static stability	$C_{m_\alpha} < 0$ $C_{n_\beta} > 0$	static SAS
Dynamic stability	eigenmotions stable $Re(\lambda) < 0$	dampers

Introduction

Position of the Stability Augmentation System in the global Flight Control System:



Introduction

- SAS augments open loop static and dynamic stability of the aircraft.
- SAS is necessary when inherent open loop characteristics $(\omega, \zeta, \tau, ...)$ do not comply with flying qualities requirements.
- Discussed last week:
  - Yaw damper: $r \rightarrow \delta_r$
  - Pitch damper: $q \rightarrow \delta_e$
  - Phugoid damper: $u \rightarrow \delta_e$

Introduction

- Fighter aircraft have enhanced manoevrability at a cost of reduced static longitudinal/lateral inherent aircraft stability.
- In the longitudinal case there is often a negative *static margin*, so stability augmentation is required.
- Augmentation forms:
  - Longitudinal: Angle of attack feedback, acceleration feedback.
  - Lateral/directional: Sideslip feedback.
- What type of aircraft will have problems with directional stability?

Introduction

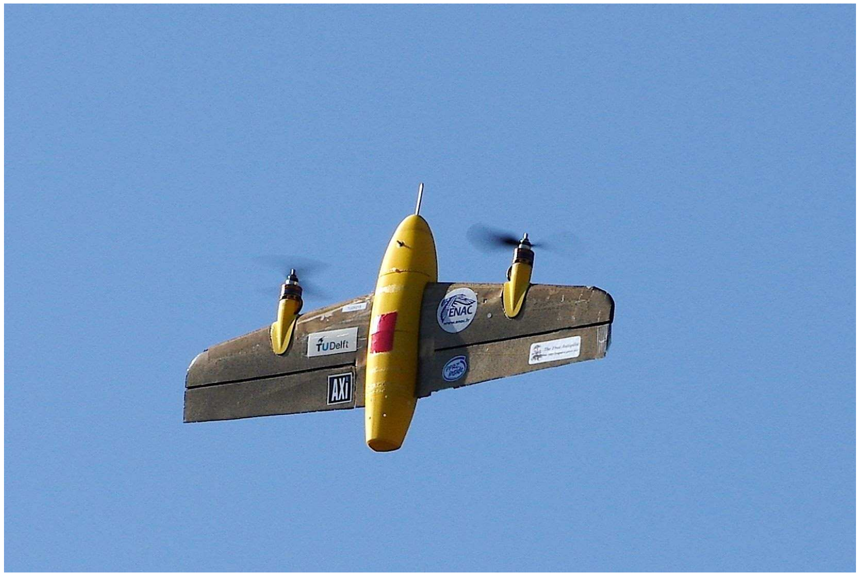
Inherent static directional instability, e.g. for tailless aircraft.

Northrop B-2 Spirit

Boeing X-36

Introduction

Inherent static directional instability, e.g. for tailless aircraft.



The Cyclone UAV uses differential thrust for sideslip feedback.

Introduction

Inherent stable aircraft:

F-15 Eagle

F-4 Phantom

F-5 Tiger

A-4 Skyhawk

Introduction

Inherent unstable aircraft:

F-16 Fighting Falcon

F-117 Nighthawk

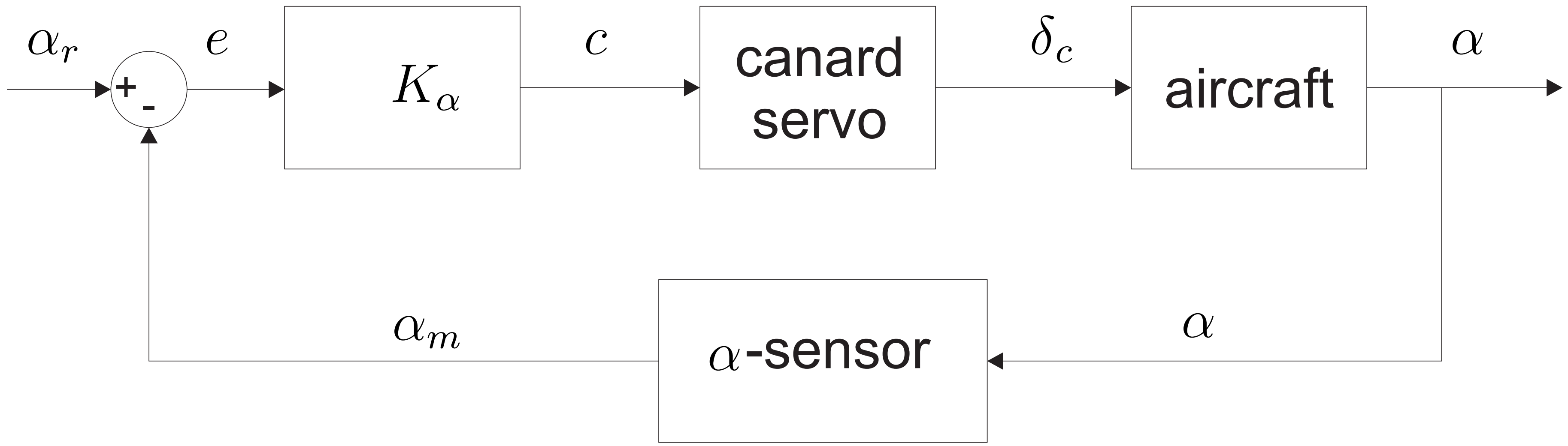
Eurofighter Typhoon

F-22 Raptor

Angle of attack feedback

Angle of attack feedback

Block diagram of the Saab JAS-39 Gripen:



Angle of attack feedback

- Canard to angle-of-attack transfer function at 950 km/h, 13.7 km:

$$\frac{\alpha(s)}{\delta_c(s)} = \frac{-35.4s^3 + 5786s^2 + 11.5s + 22.5}{871s^4 + 608s^3 - 9065s^2 - 43.1s - 43.34}$$

- Servo transfer function:

$$H_{servo} \approx \frac{1}{0.025s + 1} \qquad (\omega_{br} = 40\ rad/s)$$

- Angle-of-attack sensor dynamics:

$$H_{\alpha-sensor} \approx 1$$

Angle of attack feedback

- From flying qualities requirements **(see lecture 9)**:

$$2 \leq \omega_{n_{sp}} \leq 3 \;\; rad/s$$

- Use root locus to find appropriate feedback gain
- $K_{alpha} = 2.5$
- Is this magnitude of the feedback gain acceptable?

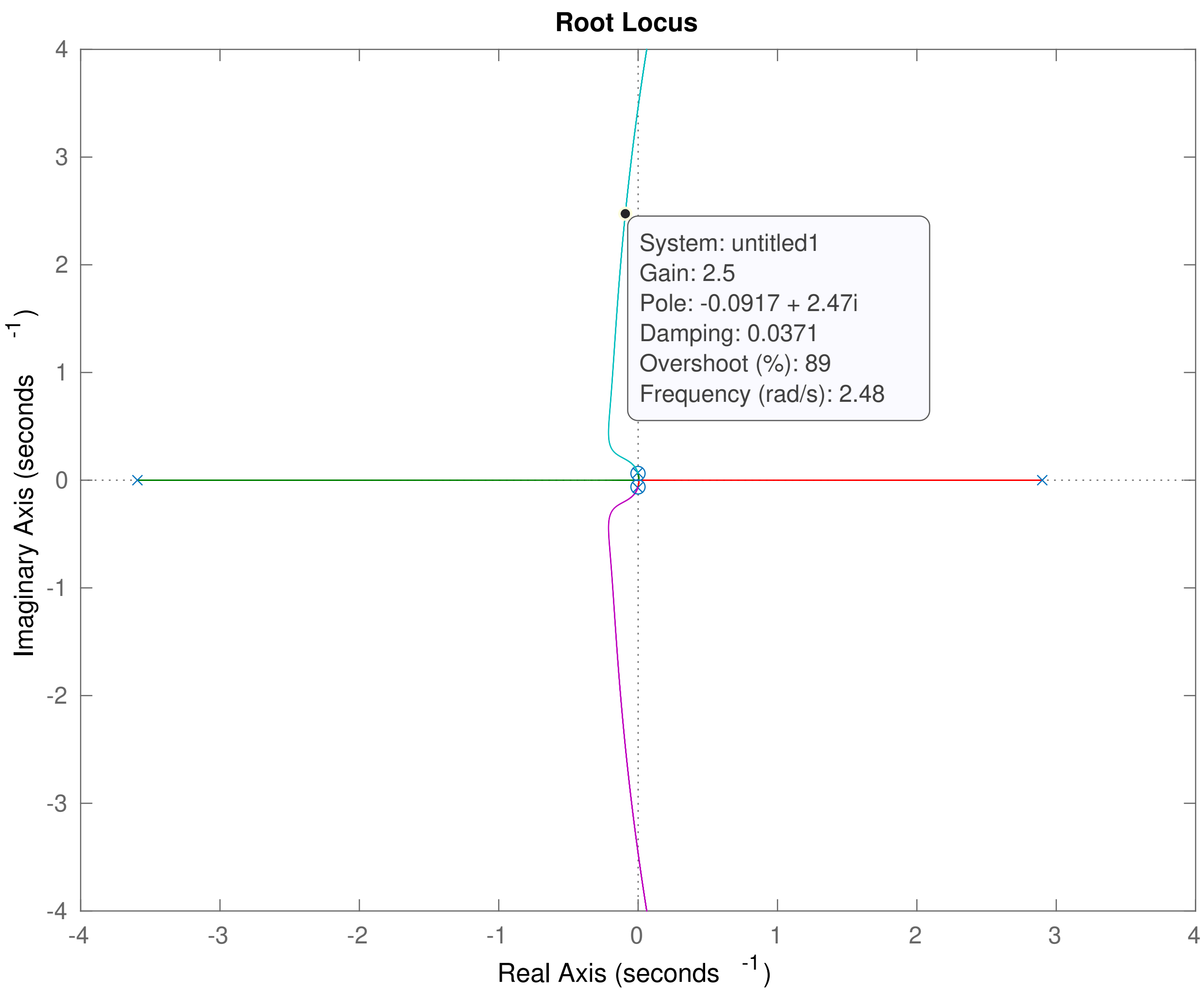
Angle of attack feedback

- From flying qualities requirements **(see lecture 9)**:

$$2 \leq \omega_{n_{sp}} \leq 3 \ \ rad/s$$

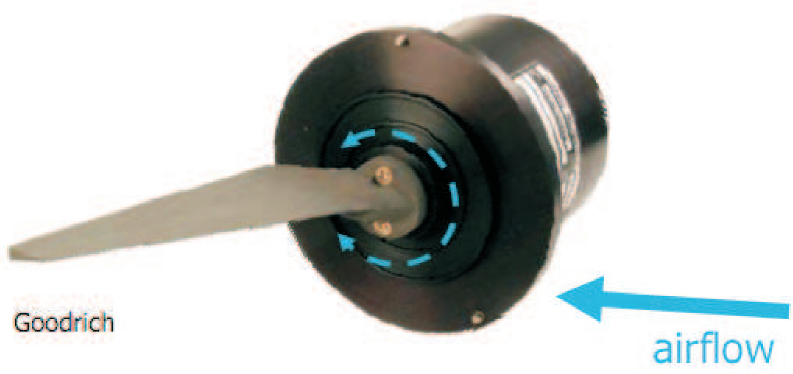
- Use root locus to find appropriate feedback gain
- $K_{alpha} = 2.5$
- Is this magnitude of the feedback gain acceptable?
- Vertical gust of $4.572 m/s$ (MIL-F-8785C) at $V = 950$ km/h.
- $\alpha_{induced} = \mathrm{atan}\left(\frac{4.572}{265.8}\right) = 0.985°$
- $\delta_c = K_\alpha \alpha_{induced} = 2.46°$, which is acceptable.

Angle of attack feedback



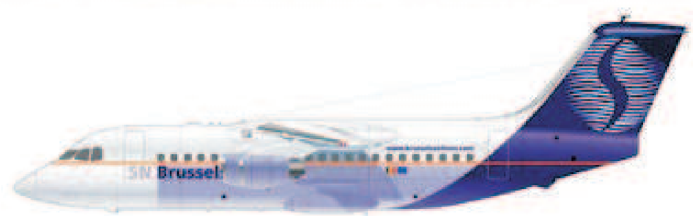
Angle of attack feedback

- α-sensors have poblems with local airflows.
- Output filtering is required.
- An alternative is to use the load factor as a feedback signal.



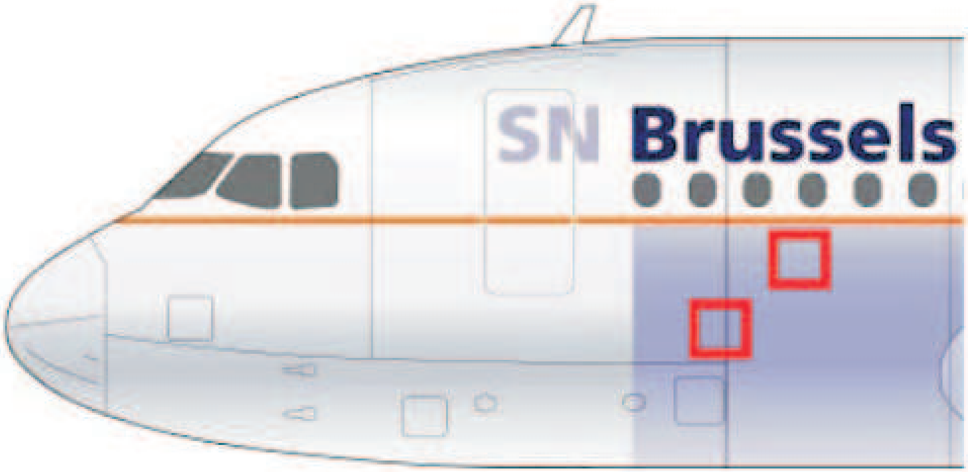
Angle of attack feedback

Picture of the angle of attack sensor of the BAe RJ85:



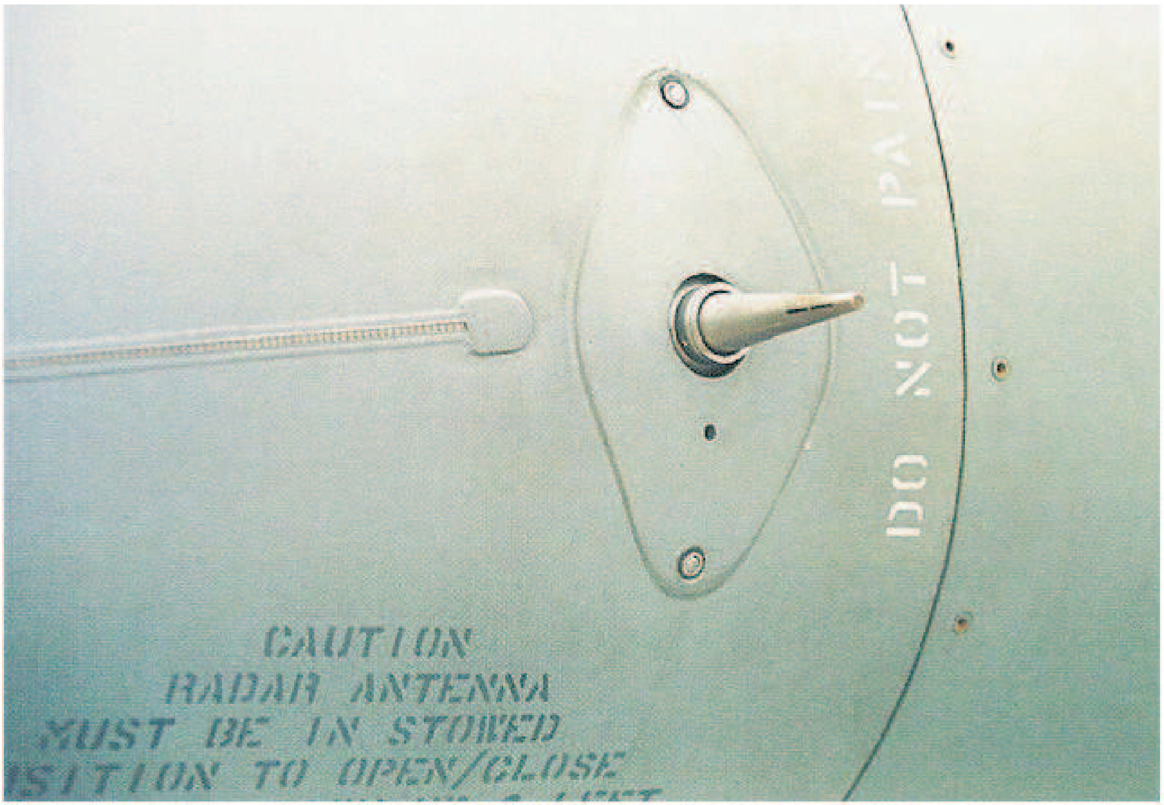
Angle of attack feedback

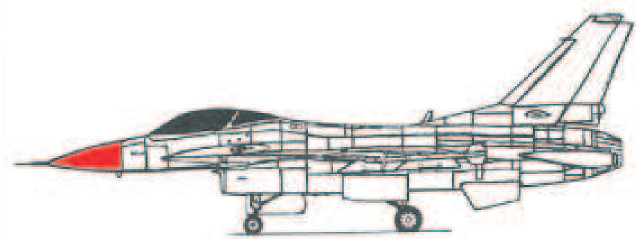
Picture of the angle of attack sensor of the Airbus A320:



Angle of attack feedback

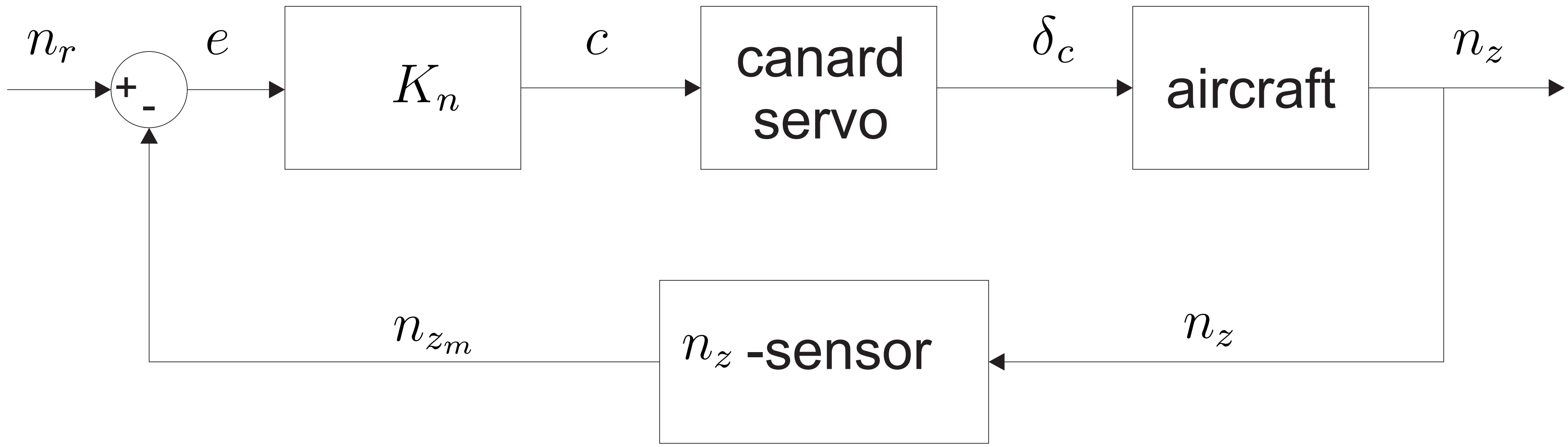
Picture of the angle of attack sensor of the Lockheed F-16:





Load factor feedback.

Block diagram of the Saab JAS-39 Gripen:



Load factor feedback.

- Load factor in terms of rate of change of γ:

$$n = \frac{U_0 \dot{\gamma}}{9.81}$$

- Corresponding load factor to canard transfer function:

$$\frac{n(s)}{\delta_c(s)} = \frac{U_0}{9.81}\frac{\dot{\gamma}(s)}{\delta_c(s)} = \frac{U_0 s}{9.81}\frac{\gamma(s)}{\delta_c(s)}$$

- Since $\gamma = \theta - \alpha$: $\frac{n(s)}{\delta_c(s)} = \frac{U_0 s}{9.81}\left(\frac{\theta(s)}{\delta_c(s)} - \frac{\alpha(s)}{\delta_c(s)}\right)$

Load factor feedback.

- Canard to load factor transfer function:

$$\frac{n\left(s\right)}{\delta_c\left(s\right)} = 27.07s\frac{35.4s^3 + 25s^2 + 2145.5s - 22.4}{871s^4 + 608s^3 - 9065s^2 - 43.1s - 43.34}$$

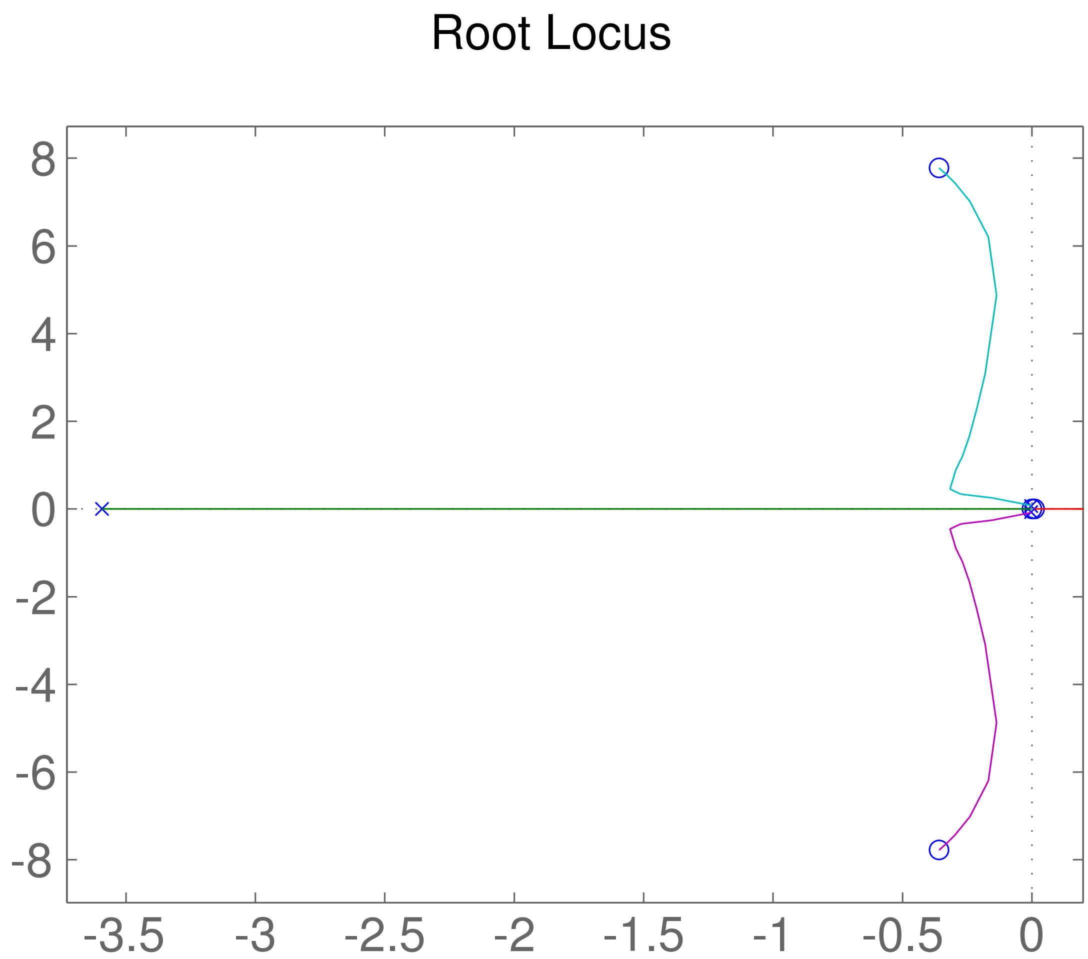
- Servo transfer function:

$$H_{servo} \approx \frac{1}{0.025s + 1} \qquad (\omega_{br} = 40 \text{ rad/s})$$

- Load factor sensor dynamics:

$$H_{n_z - sensor} \approx 1$$

Load factor feedback.



Load factor feedback.

- From flying qualities requirements:

$$2 \leq \omega_{n_{sp}} \leq 3 \ \ \text{rad/s}$$

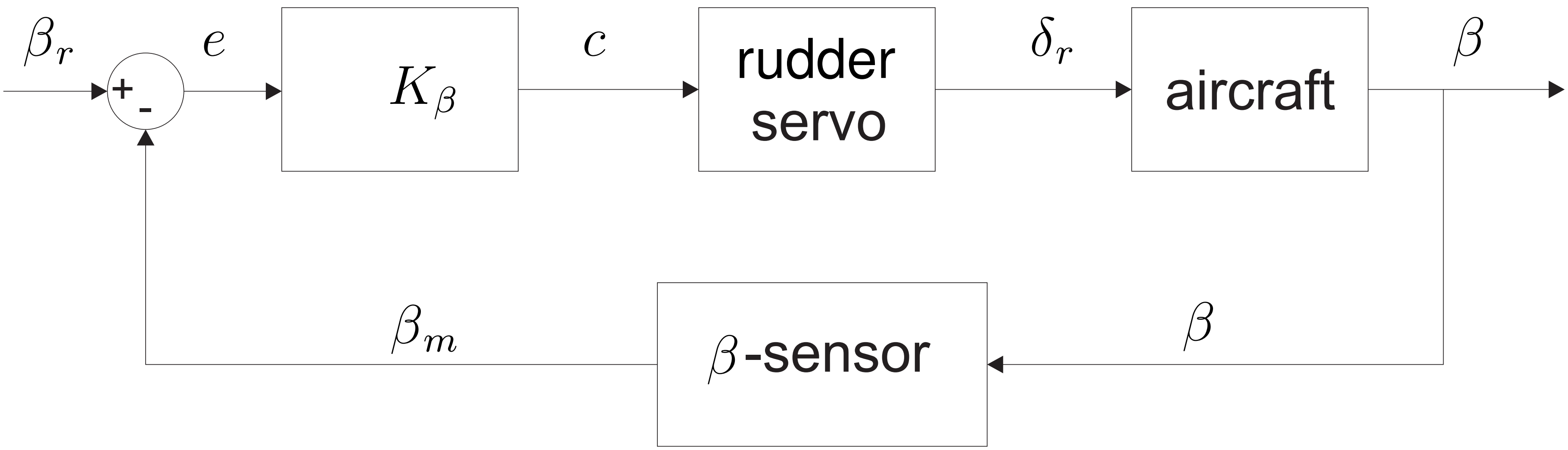
- Use root locus to find appropriate feedback gain
- $K_n = 0.279$
- Is this level of feedback gain acceptable?
- Assume 5g perturbation:

$$\delta_c = K_n \Delta n = 0.279 * 5 \approx 1.4^\circ$$

This is acceptable!

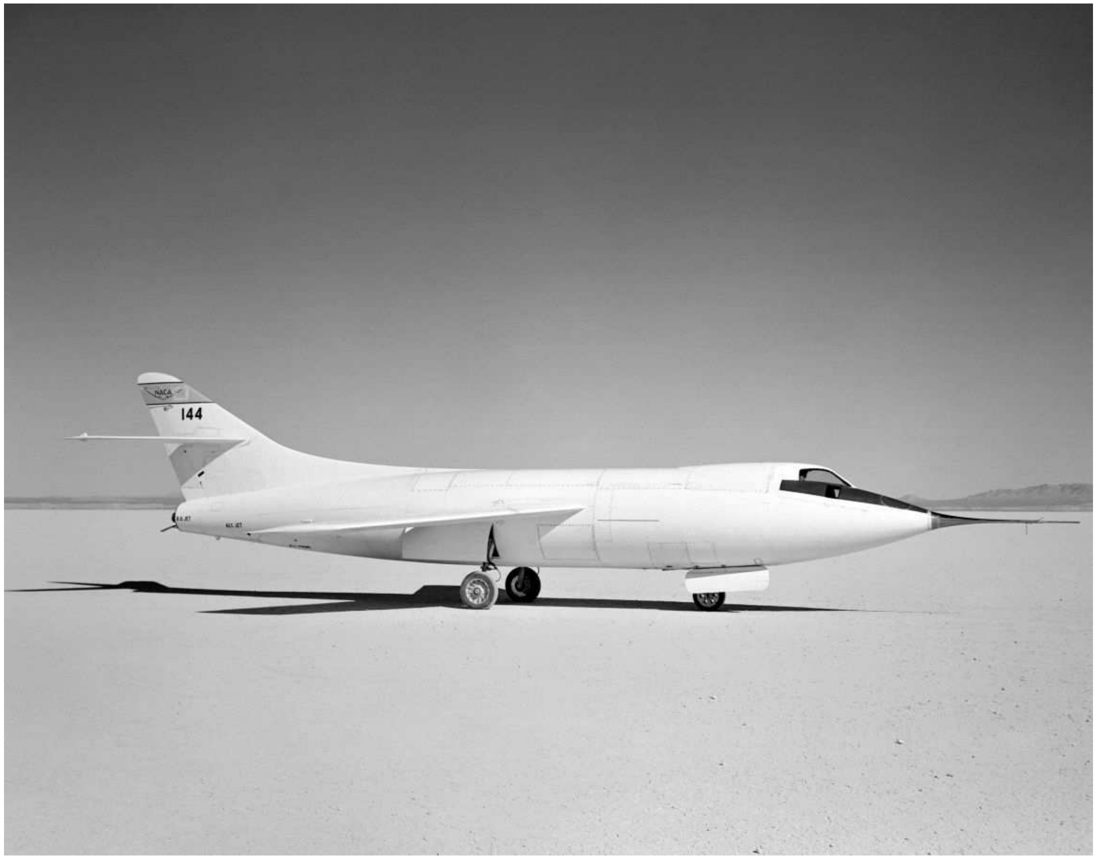
Sideslip feedback

Block diagram of sideslip feedback:



Sideslip feedback

Douglas D-558-II airplane:



Sideslip feedback

- Rudder to angle-of-sideslip transfer function in case of a reduced vertical stabilizer:

$$\frac{\beta(s)}{\delta_r(s)} = \frac{4.19s^3 + 924.3s^2 + 34.36s - 0.108}{1364.1s^4 + 521.5s^3 + 40.43s^2 + 8.648s + 0.36}$$

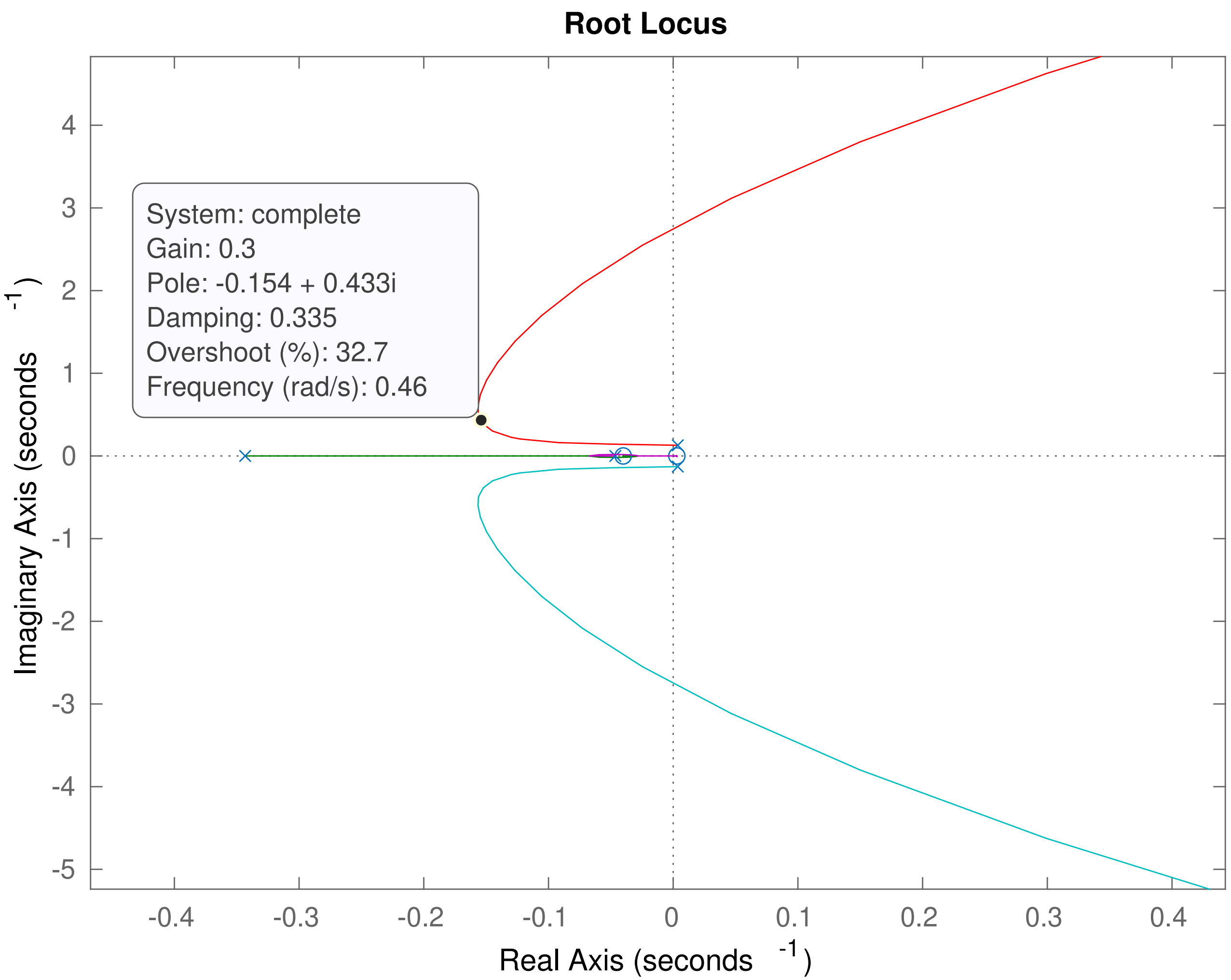
- Servo transfer function:

$$H_{servo} \approx \frac{1}{0.05s + 1} \qquad (\omega_{br} = 20 \text{ rad/s})$$

- Sideslip sensor dynamics:

$$H_{\beta-sensor} \approx 1$$

Sideslip feedback



Lateral requirements

Dutch roll frequency and damping, MIL-F-8785C

Level	Category	Class	min ζ_d	min $\zeta_d \omega_{n_d}$	min ω_{n_d}
Level 1	A(C,GA)	IV	0.4	–	1.0
	A	I and IV	0.19	0.35	1.0
		II and III	0.19	0.35	0.4
	B	all	0.08	0.15	0.4
	C	I,II-C,IV	0.08	0.15	1.0
		II-L,III	0.08	0.10	0.4
Level 2	all	all	0.02	0.05	0.4
Level 3	all	all	0	–	0.4

Sideslip feedback

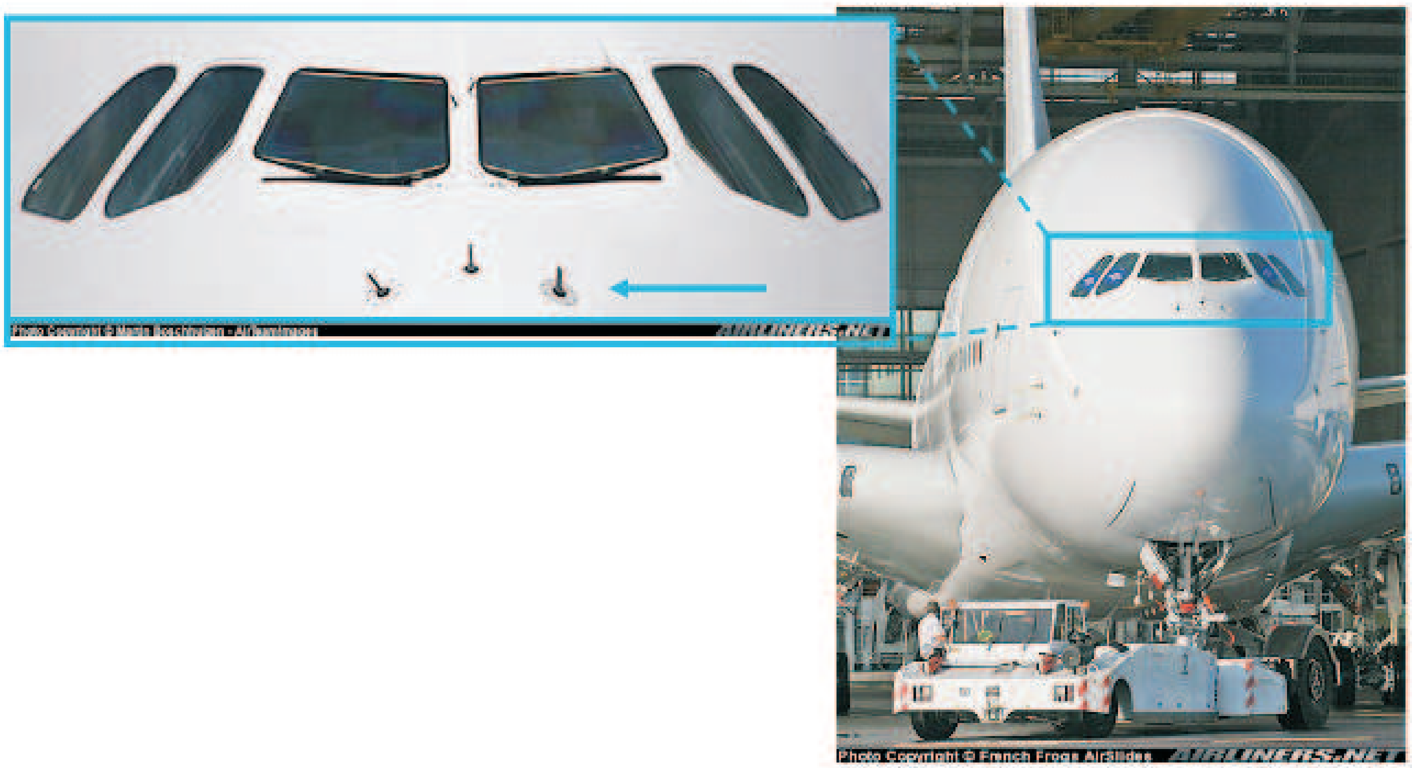
- From flying qualities requirements:

$$\omega_{dr} \geq 0.4 \text{ rad/s}$$

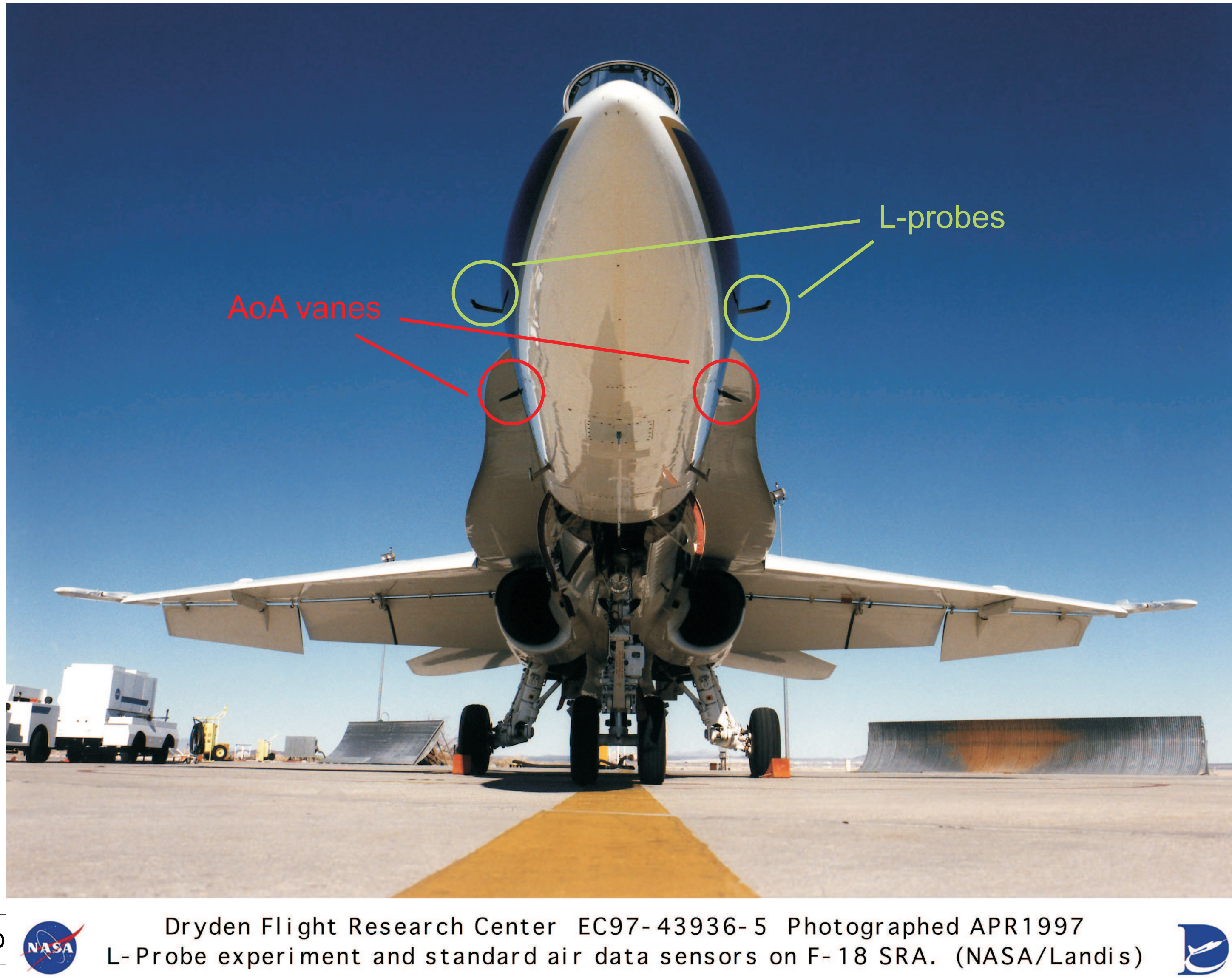
- Use root locus to find appropriate feedback gain
- $K_\beta = 0.3$
- Is this level of feedback gain acceptable?
- Assume 5° sideslip disturbance: $\delta_r = K_\beta \Delta\beta = 1.5^\circ$
  This is acceptable!

Sideslip feedback

Picture of the sideslip sensor of the Airbus A380:



Sideslip feedback



Summary

- Static SAS: compensate for inherent negative static stability margin.
- Different augmentation forms:
  - Longitudinal: Angle-of-attack feedback, load-factor feedback.
  - Lateral: Sideslip feedback
- Proportional feedback gains only.
- Check sensitivity of feedback gains to disturbances.

What's next?

- Exam training
- Longitudinal autopilot modes.
- Lateral autopilot modes.
- Navigational autopilot systems.

AE4-301 Automatic Flight Control System Design

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AE4-301 Automatic Flight Control System Design

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October 8, 2020

Course structure

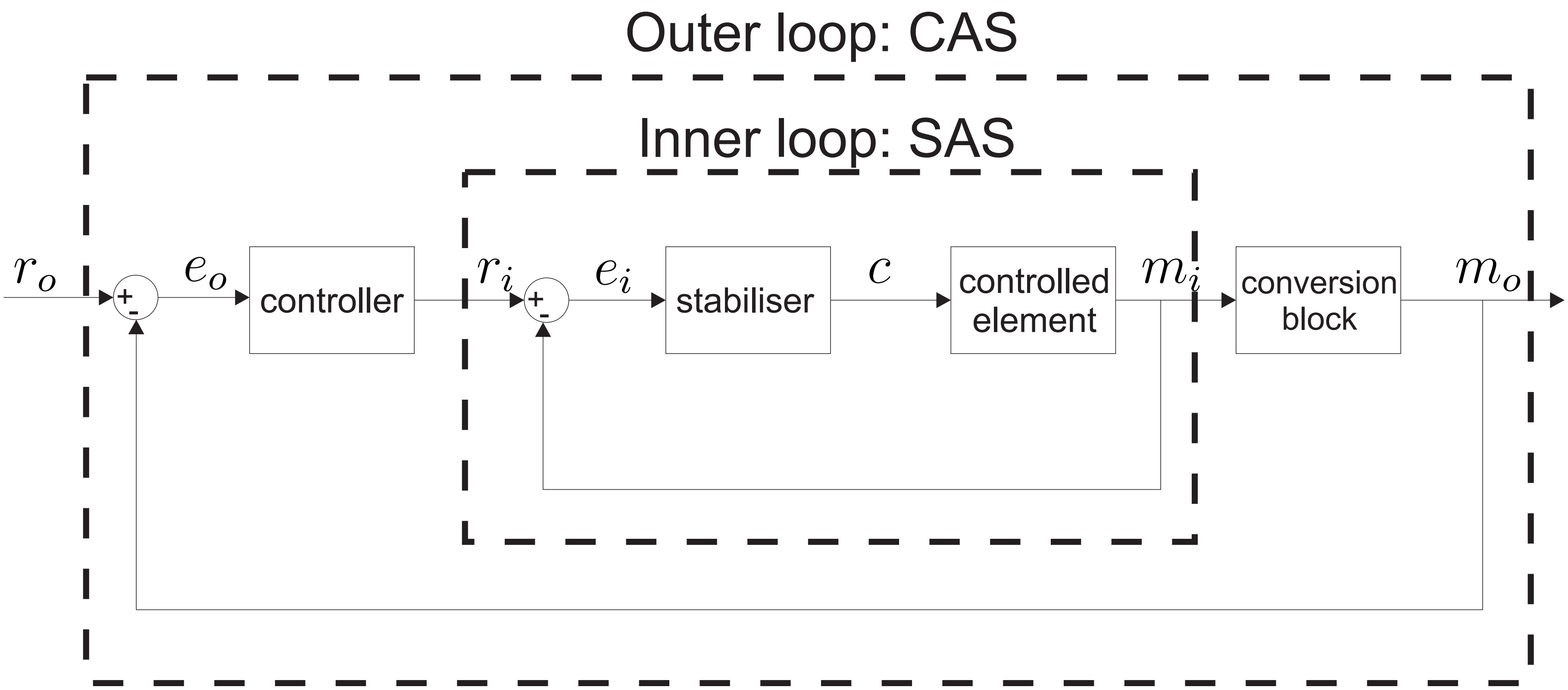
- Introduction automated flight control and recap flight dynamics.
- Control theory.
- Performance and handling qualities.
- Static/dynamic stability augmentation.
- **Longitudinal autopilot modes.**
- Lateral autopilot modes.
- Navigational autopilot systems.

Lecture 12 structure

- Introduction
- Longitudinal control/hold functions
  - Pitch attitude
  - Altitude
  - Airspeed
  - Climb or descend rate
- Summary

Introduction

Position of the Control Augmentation System (CAS) in the global Flight Control System.



Introduction

Autopilot system use:

- For safety reasons.
- Lowers pilot workload (on long flights).

Autopilot system categories:

- Control or hold certain flight parameters **(Lectures 12+13)**.
- Perform navigational tasks **(Lectures 13+14)**.

Introduction

Longitudinal control/hold functions: Operated through Mode Control Panel.

- Pitch attitude
- Altitude
- Airspeed
- Climb or descend rate

Pitch attitude hold mode

Use: prevents pilot having to constantly control pitch attitude in turbulent air (e.g. TURB switch in 747-400)

Autopilot mode analysis/design, 2 important questions:

- Which type of control device do we use?

Pitch attitude hold mode

Use: prevents pilot having to constantly control pitch attitude in turbulent air (e.g. TURB switch in 747-400)

Autopilot mode analysis/design, 2 important questions:

- Which type of control device do we use? $\longrightarrow$**Elevator**
- What type of feedback loop structure (sensor)?

Pitch attitude hold mode

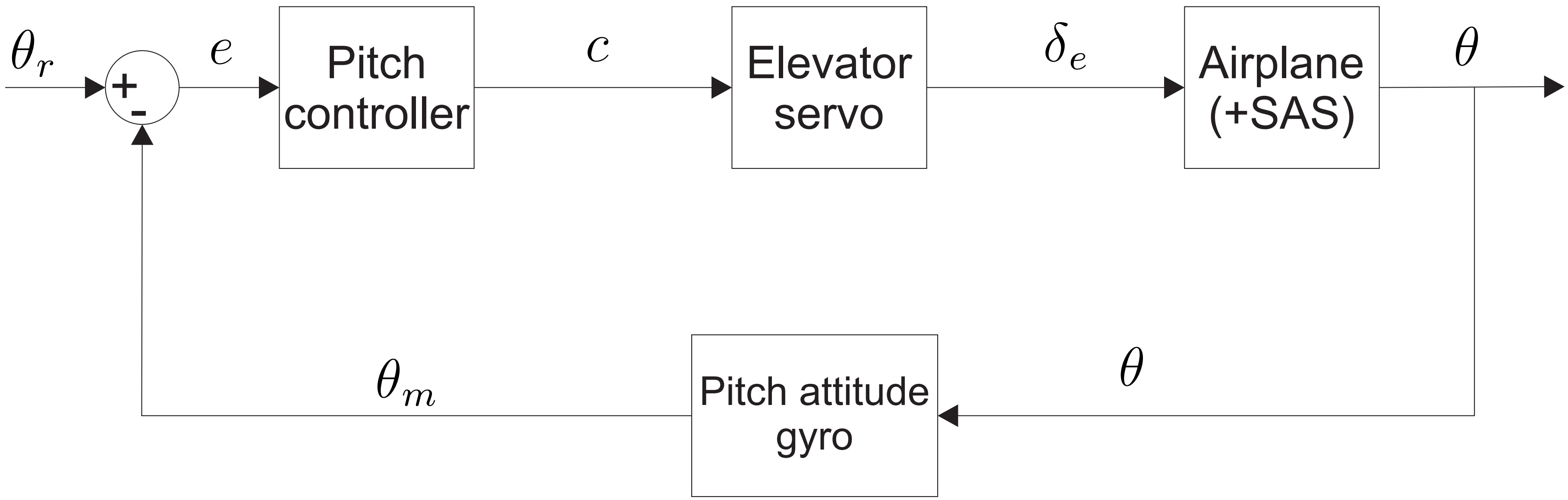
Use: prevents pilot having to constantly control pitch attitude in turbulent air (e.g. TURB switch in 747-400)

Autopilot mode analysis/design, 2 important questions:

- Which type of control device do we use? $\longrightarrow$**Elevator**
- What type of feedback loop structure (sensor)? $\longrightarrow$**Vertical gyro**

Pitch attitude hold mode

Block diagram of a pitch attitude hold loop:



Pitch attitude hold mode

Gyro and servo model representations(as before):

- Pitch attitude gyro: $H_{\text{gyro}} \approx 1$.
- Elevator servo: $H_{\text{servo}} \approx \frac{1}{\tau_{\text{servo}} s+1}$.

The pitch attitude controller incorporates three types of action:

- Proportional: reduces rise time.
- Integral: eliminates steady state error.
- Derivative: improves transient response(damping).

$$K\left(s\right) = K_P + \frac{K_I}{s} + K_D s = \frac{K_D s^2 + K_P s + K_I}{s}$$

Pitch attitude hold mode

What about the reference input signal θ_r?
In the TURB mode: the governing attitude when the TURB mode is activated is kept constant.

Examples:

- Cessna 620
- Boeing 747
- Bunting UAV

Pitch attitude hold mode

Cessna 620

Dimensions:

- Wing area: 31.6 m^2
- Mean chord: 2 m
- Wingspan: 16.8 m

Flight condition: Approach

- Altitude: 0 m
- Speed: 207.4 km/h
- Weight: 68 kN

Pitch attitude hold mode

<u>Cessna 620</u>
Reduced elevator deflection to pitch attitude transfer function in approach:

$$\frac{\theta(s)}{\delta_e(s)} = \frac{-(1841.3s + 1540.3)}{189.3s^3 + 740.35s^2 + 1412.74s}$$

- Open loop $\frac{q(s)}{\delta_e(s)}$ damping: $\zeta = 0.716$.
- Open loop $\frac{q(s)}{\delta_e(s)}$ frequency: $\omega_{sp} = 2.73 rad/s$.
- Required short period damping: $0.35 \leq \zeta \leq 1.30$.
- Required short period frequency: $0.9 \leq \omega_{sp} \leq 4 rad/s$.

Pitch attitude hold mode

Cessna 620 pitch attitude hold mode:

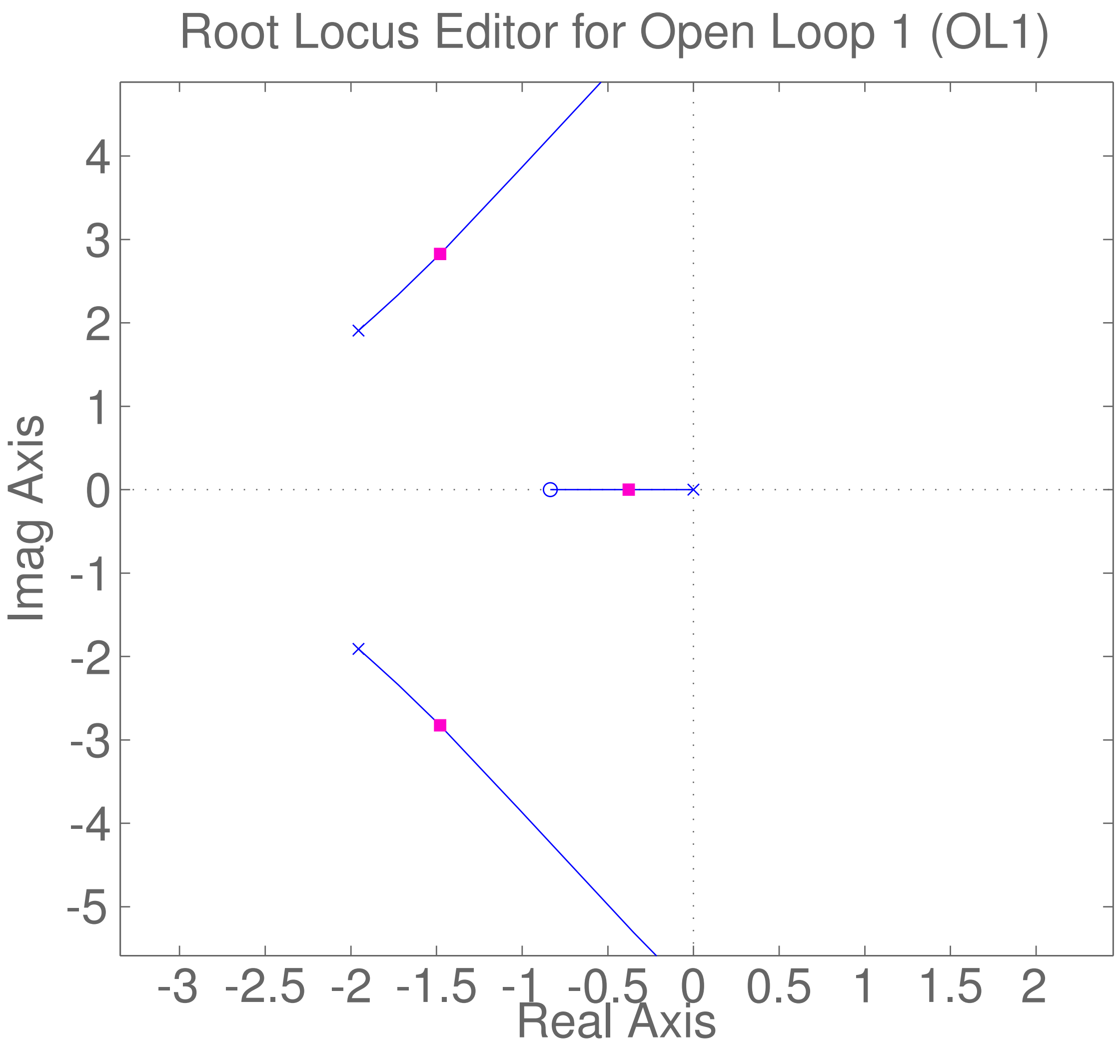
- **Simulink demo**: Open loop θ_{ref} to θ.
- **Sisotool demo**: Pitch feedback gain determination.
- **Simulink demo**: Closed loop θ_{ref} to θ.

Pitch attitude hold mode

Cessna 620 pitch attitude hold mode:

- Selecting pitch feedback gain: trade-off between rise time and damping.
- In this example: $K = 0.5$ is selected.
- Short period damping goes from $\zeta = 0.716$ to $\zeta = 0.464$, which still satisfies the requirements, so **no inner loop pitch rate control is required**.

Pitch attitude hold mode



Pitch attitude hold mode

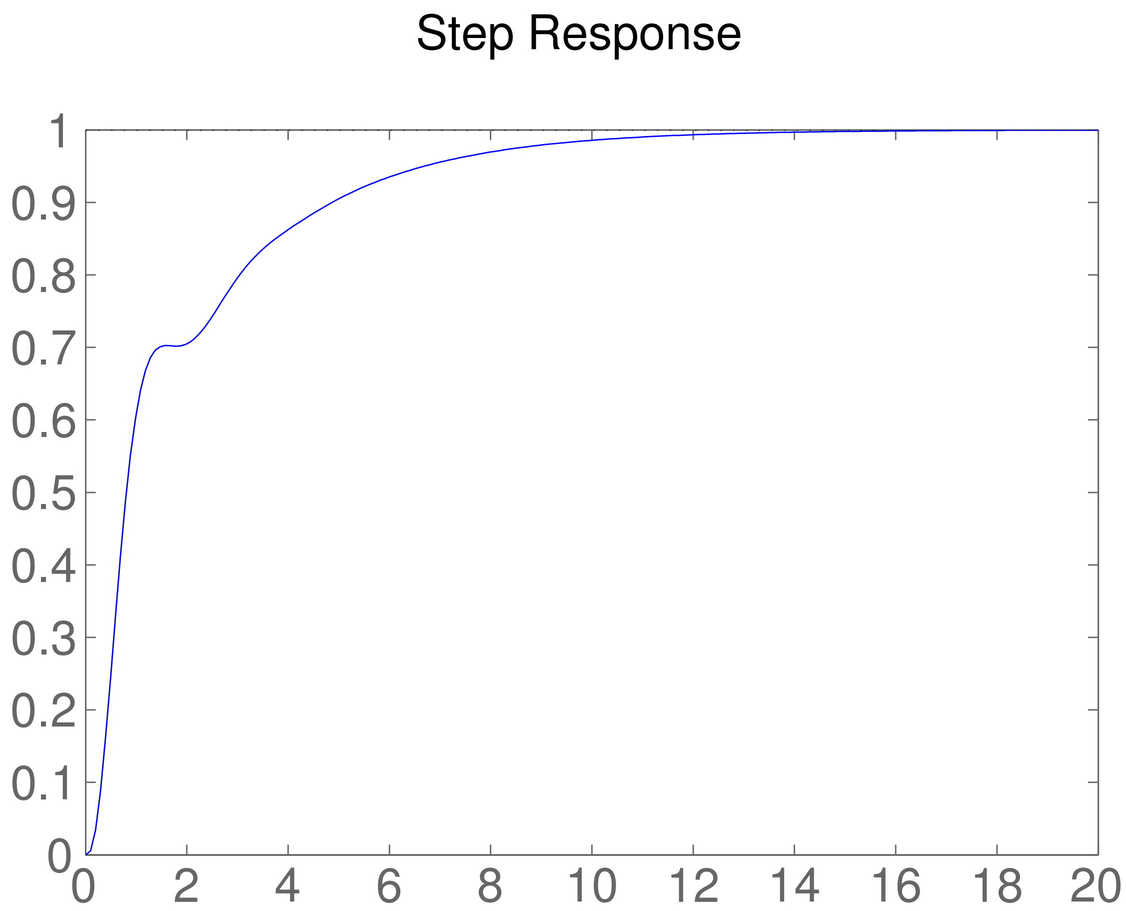


Figure 1: Pitch angle step response

Pitch attitude hold mode

Boeing 747-200

Dimensions:
-wing area: 511 m^2
-mean chord: 8.3 m
-wingspan: 59.7m
Flight condition: high cruise
-altitude: 12200 m
-speed: 956 km/h
-weight: 2.888MN

Pitch attitude hold mode

Inherent aircraft properties:

- Open loop system damping: $\zeta = 0.35$.
- Open loop system frequency: $\omega_{sp} = 1.33 rad/s$.

Airworthiness requirements:

- Required short period damping: $0.30 \leq \zeta \leq 2.00$.
- Required short period frequency: $1.1 \leq \omega_{sp} \leq 8 rad/s$.

Pitch attitude hold mode

Reduced elevator deflection to pitch attitude transfer function for Boeing 747-200 in cruise:

$$\frac{\theta\left(s\right)}{\delta_e\left(s\right)} = \frac{-\left(1072.8s + 380.6\right)}{871.5s\left(s^2 + 0.936s + 1.78\right)}$$

- **Simulink demo**: Open loop θ_{ref} to θ.
- **Sisotool demo**: Pitch feedback gain determination.
- **Sisotool demo**: Inner loop pitch rate gain determination.
- **Simulink demo**: Closed loop θ_{ref} to θ.

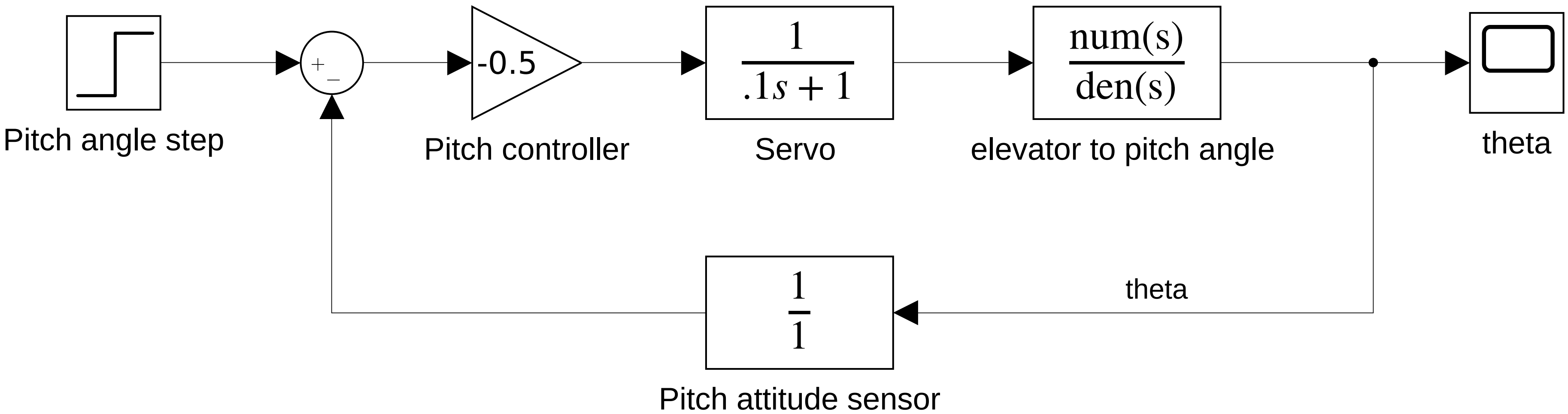
Pitch attitude hold mode

B747-200 pitch attitude hold mode:

- Selecting pitch feedback gain: trade-off between rise time and damping.
- In this example: $K = 0.5$ is selected.
- Short period damping goes from $\zeta = 0.35$ to $\zeta = 0.27$, which **does not** satisfy the requirements, so **inner loop pitch rate control is required**.
- The inner loop gain $K_q = 2$ increases the short damping ratio to $\zeta = 0.92$.

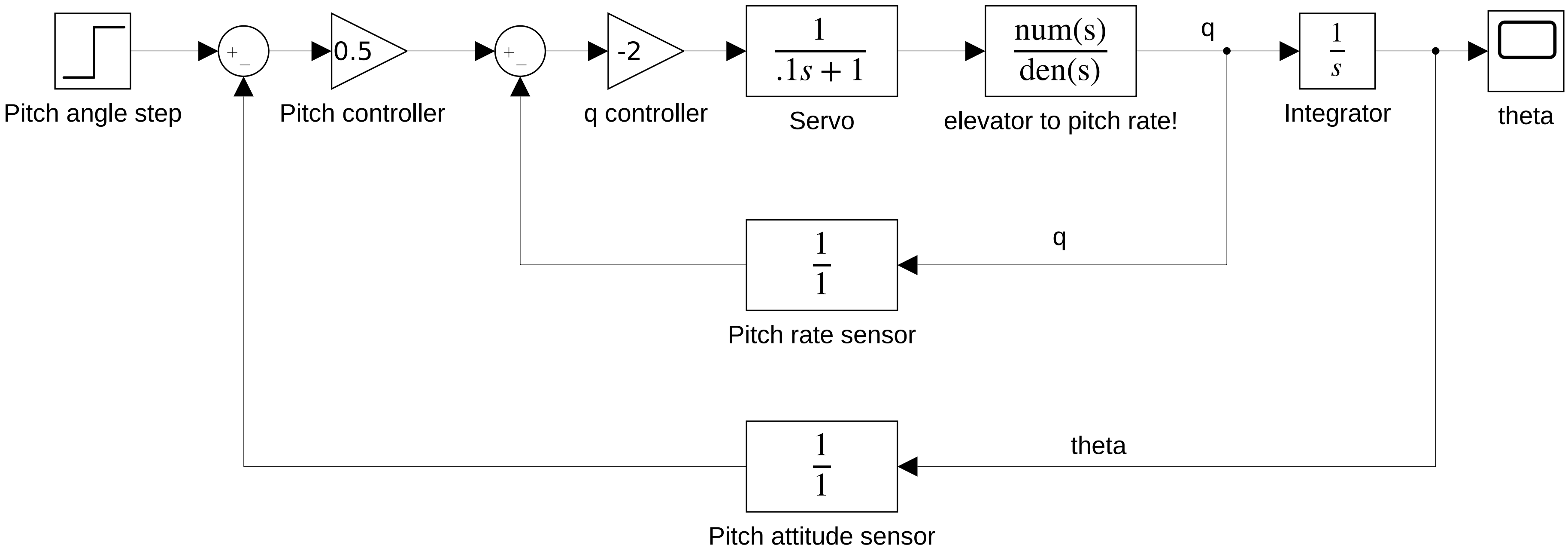
Pitch attitude hold mode

B747-200 outer loop only control.

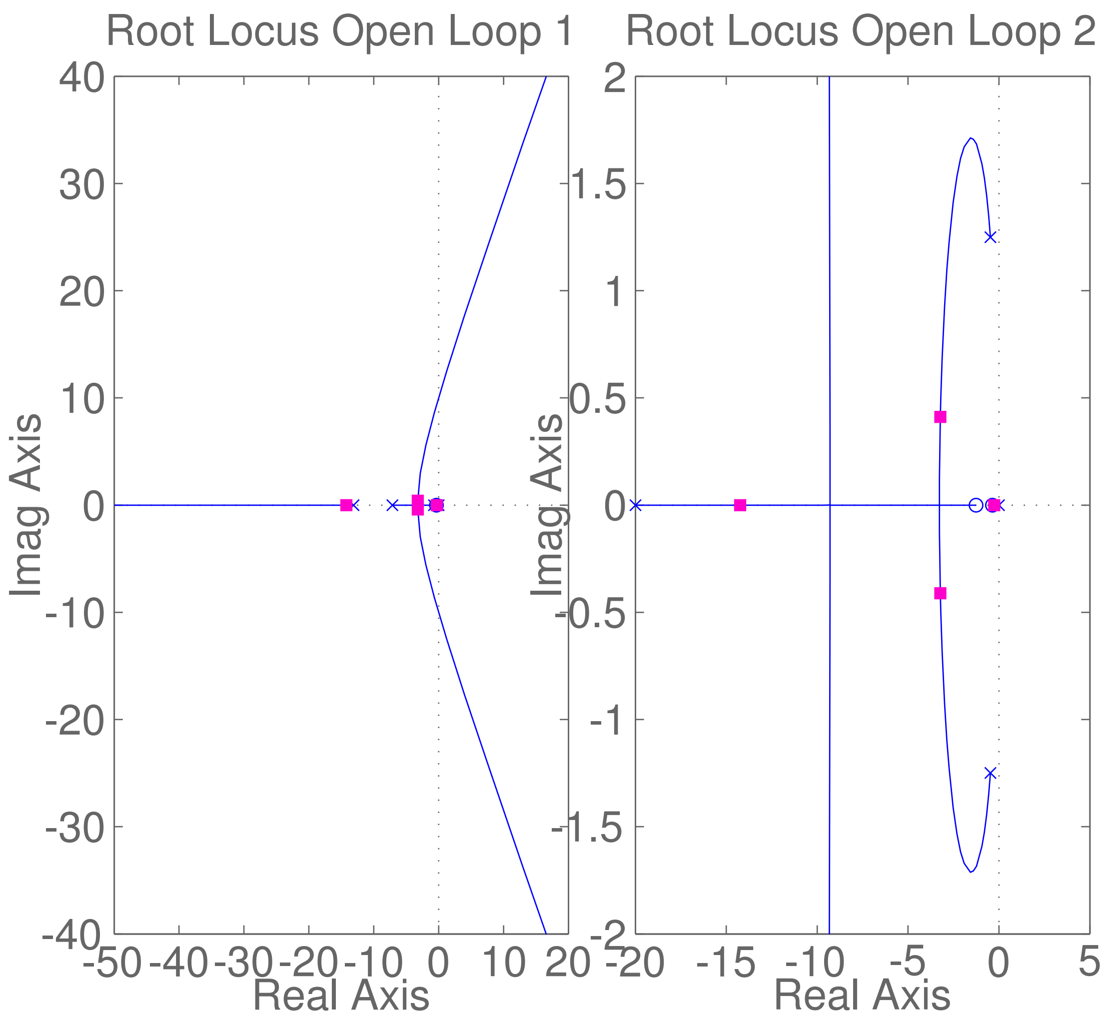


Pitch attitude hold mode

B747-200 inner loop and outer loop control.



Pitch attitude hold mode



Pitch attitude hold mode

B747-200 Closed loop step response

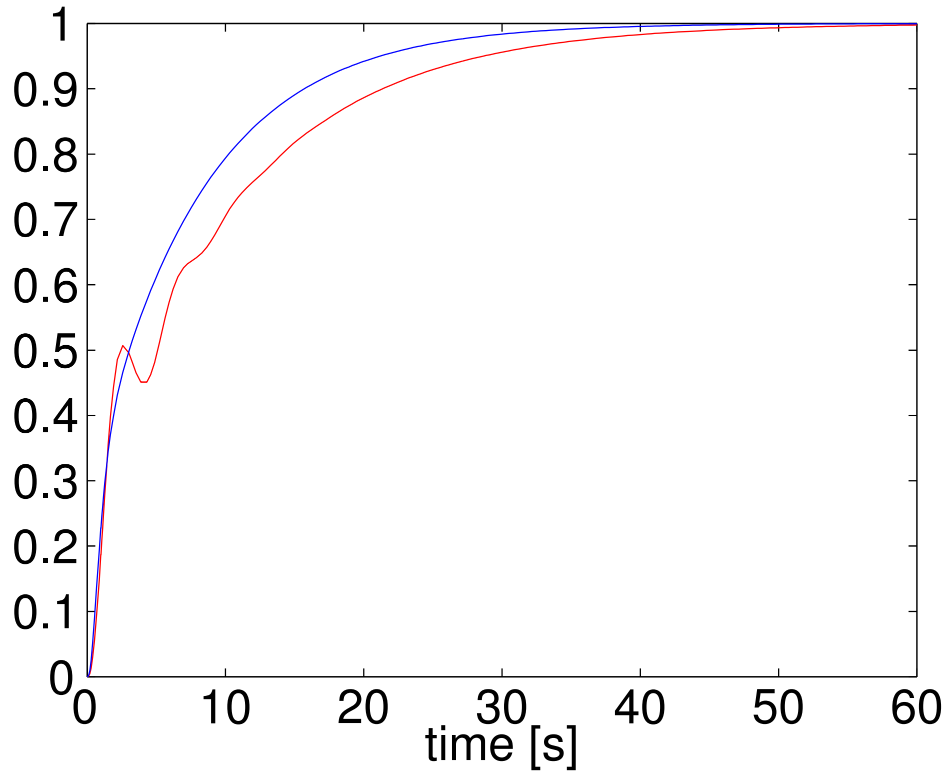
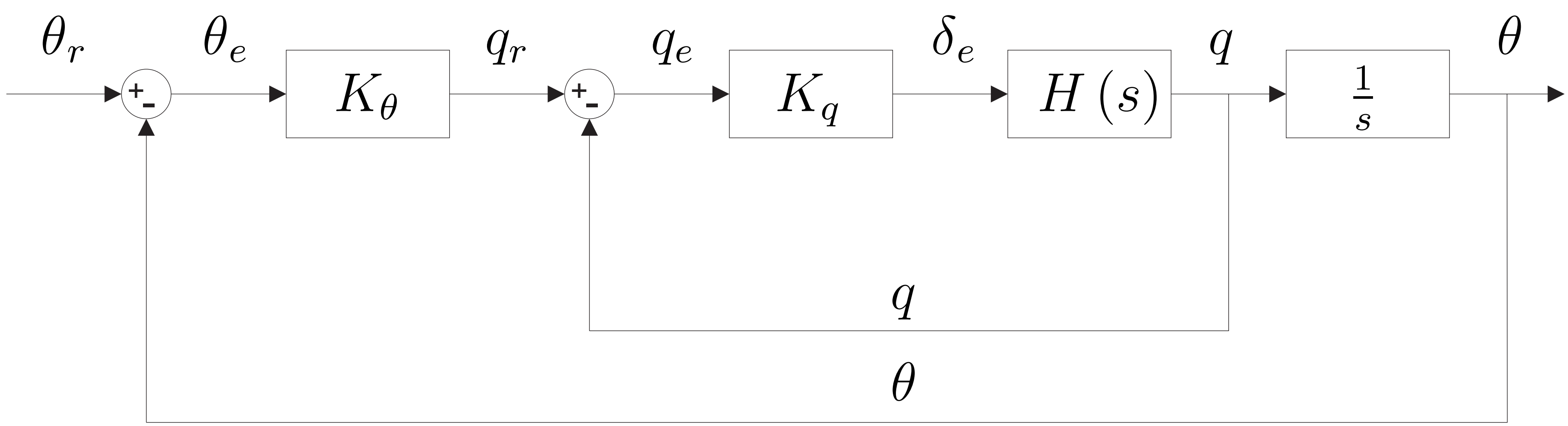


Figure 2: Red: Outer loop only. Blue: Outer+inner loop

Pitch attitude hold mode

General structure pitch attitude hold mode:

- Inner loop: pitch rate feedback.
- Outer loop: pitch angle feedback.



Pitch attitude hold mode

If the gains are kept constant, the performance of the pitch attitude hold mode controller will vary with:

- The servo break frequency.
- The flight condition:
  - Airspeed
  - Altitude

Pitch attitude hold mode

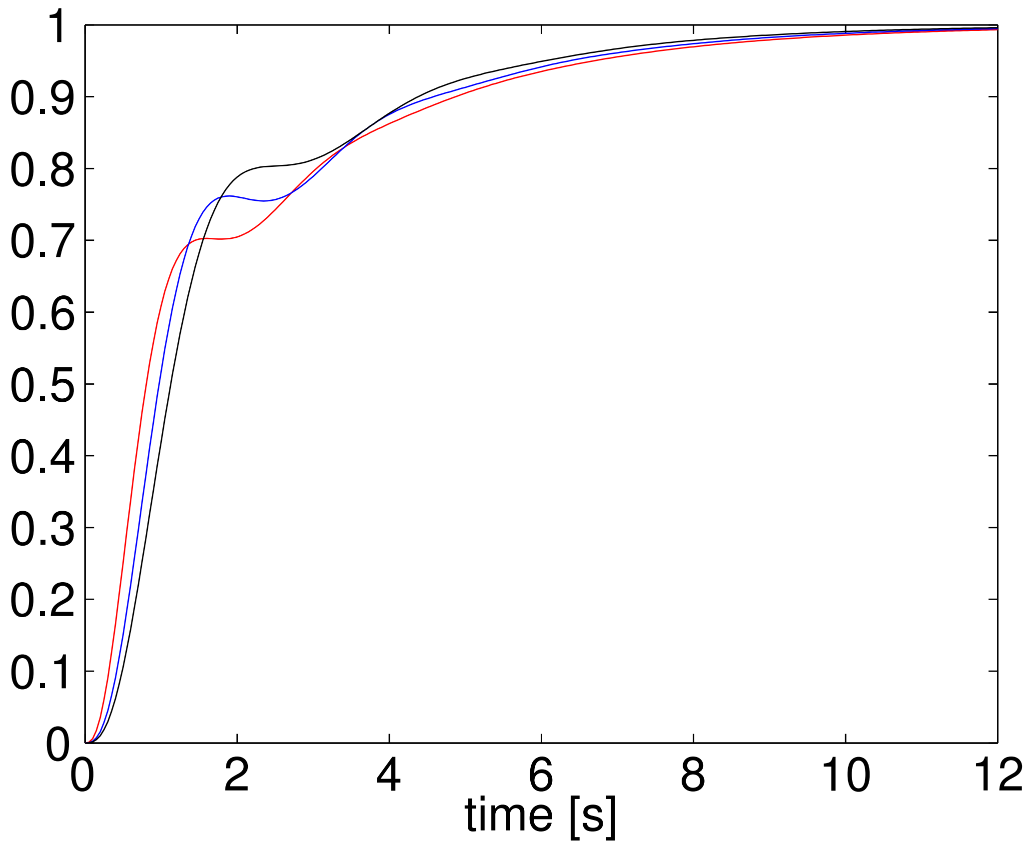
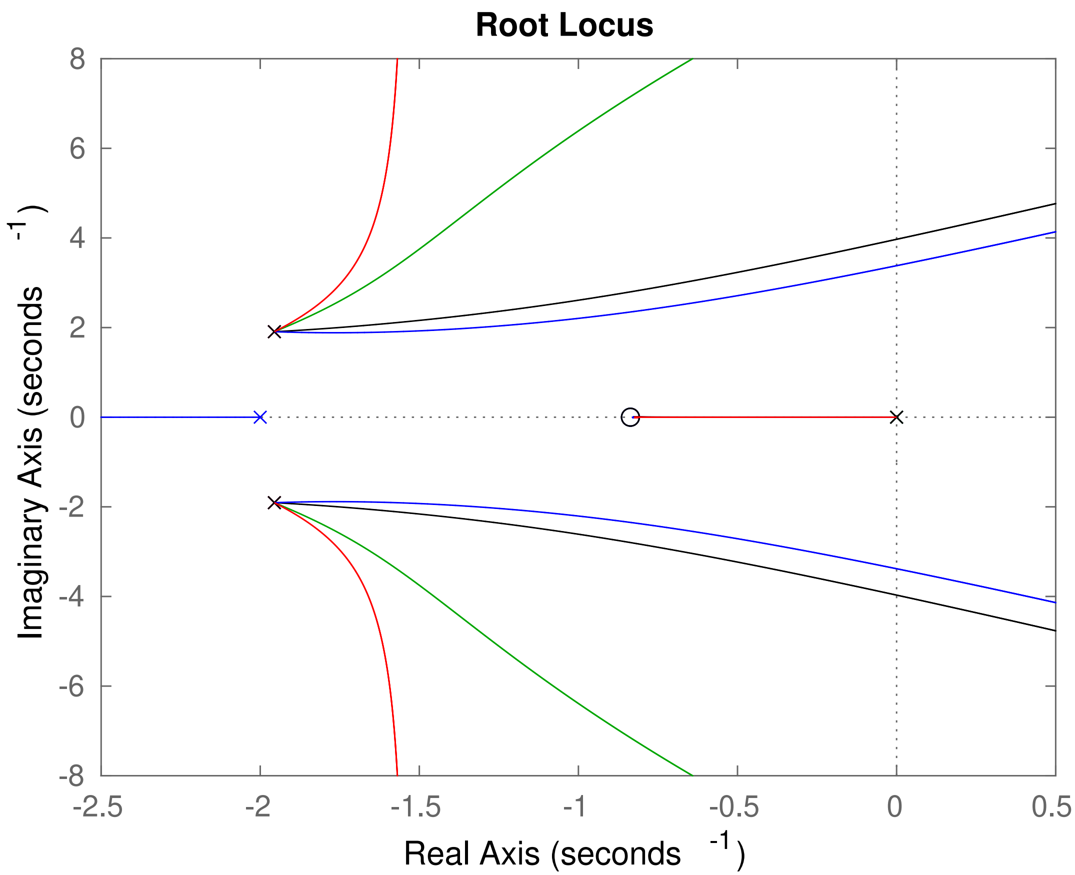


Figure 3: Cessna 620 effect of servo break frequency. Red: $\tau = 0.1$, blue: $\tau = 0.3$, black: $\tau = 0.5$

Pitch attitude hold mode



Cessna 620 effect of servo break frequency. Blue: $\tau = 0.5$, black: $\tau = 0.1$, green: $\tau = 0.03$, red: no actuator dynamics.

Pitch attitude hold mode

Pitch attitude gyro: part of Inertial Navigation System(INS)

Ring Laser Gyro Inertial Reference System

Global Navigation Air Data Inertial Reference Unit

Attitude Heading Reference System

Altitude hold mode

Use: prevents pilot having to constantly control altitude during the flight.

Autopilot mode analysis/design, 2 important questions:

- Which type of control device do we use?

Altitude hold mode

Use: prevents pilot having to constantly control altitude during the flight.

Autopilot mode analysis/design, 2 important questions:

- Which type of control device do we use? $\longrightarrow$**Elevator**
- What type of feedback loop structure (sensor)?

Altitude hold mode

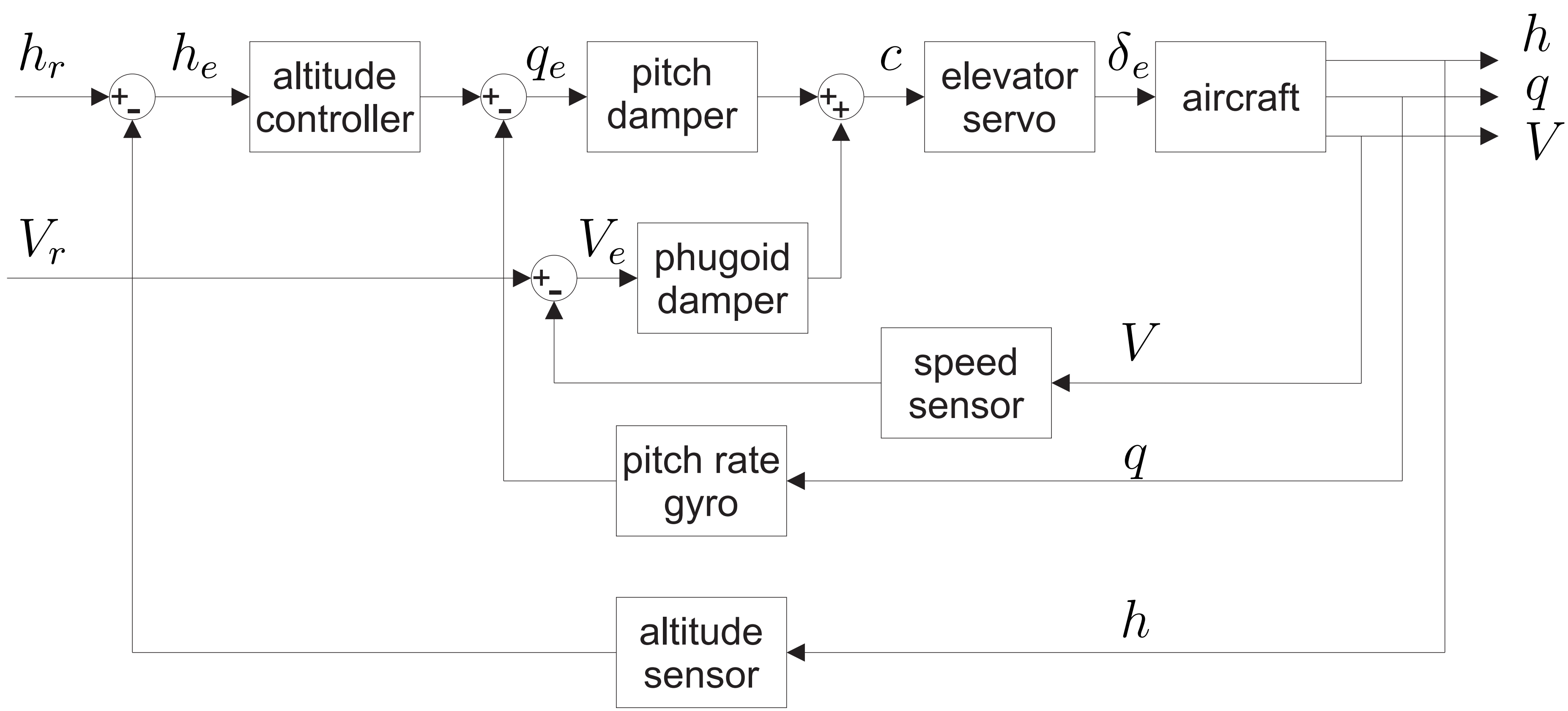
Use: prevents pilot having to constantly control altitude during the flight.

Autopilot mode analysis/design, 2 important questions:

- Which type of control device do we use? $\longrightarrow$**Elevator**
- What type of feedback loop structure (sensor)? $\longrightarrow$**altimeter**

Altitude hold mode

Block diagram of an altitude hold loop **(Worksheet)**:



Altitude hold mode

Altitude sensor model representations:

- Radar or GPS altimeter:

$$H_{\text{altimeter}} \approx 1$$

- Barometric altimeter:

$$H_{\text{altimeter}} \approx \frac{1}{\tau_{\text{altimeter}} s + 1}$$

Altitude hold mode

- How to get the transfer function $\frac{h(s)}{\delta_e(s)}$?
- For the longitudinal direction we have the states u, α, θ, q.
- State space model:

$$\begin{bmatrix} \dot{\hat{u}} \\ \dot{\alpha} \\ \dot{\theta} \\ \frac{\dot{q}\bar{c}}{V} \end{bmatrix} = \begin{bmatrix} A_{11} & A_{12} & A_{13} & A_{14} \\ A_{21} & A_{22} & A_{23} & A_{24} \\ A_{31} & A_{32} & A_{33} & A_{34} \\ A_{41} & A_{42} & A_{43} & A_{44} \end{bmatrix} \begin{bmatrix} \hat{u} \\ \alpha \\ \theta \\ \frac{q\bar{c}}{V} \end{bmatrix} + \begin{bmatrix} B_1 \\ B_2 \\ B_3 \\ B_4 \end{bmatrix} \delta_e$$

Altitude hold mode

Instead of h, look at the change in altitude $\dot{h}$.

$$\dot{h} = V_0 \sin \gamma \approx V_0 \gamma$$

$$sh(s) = V_0 \gamma(s)$$

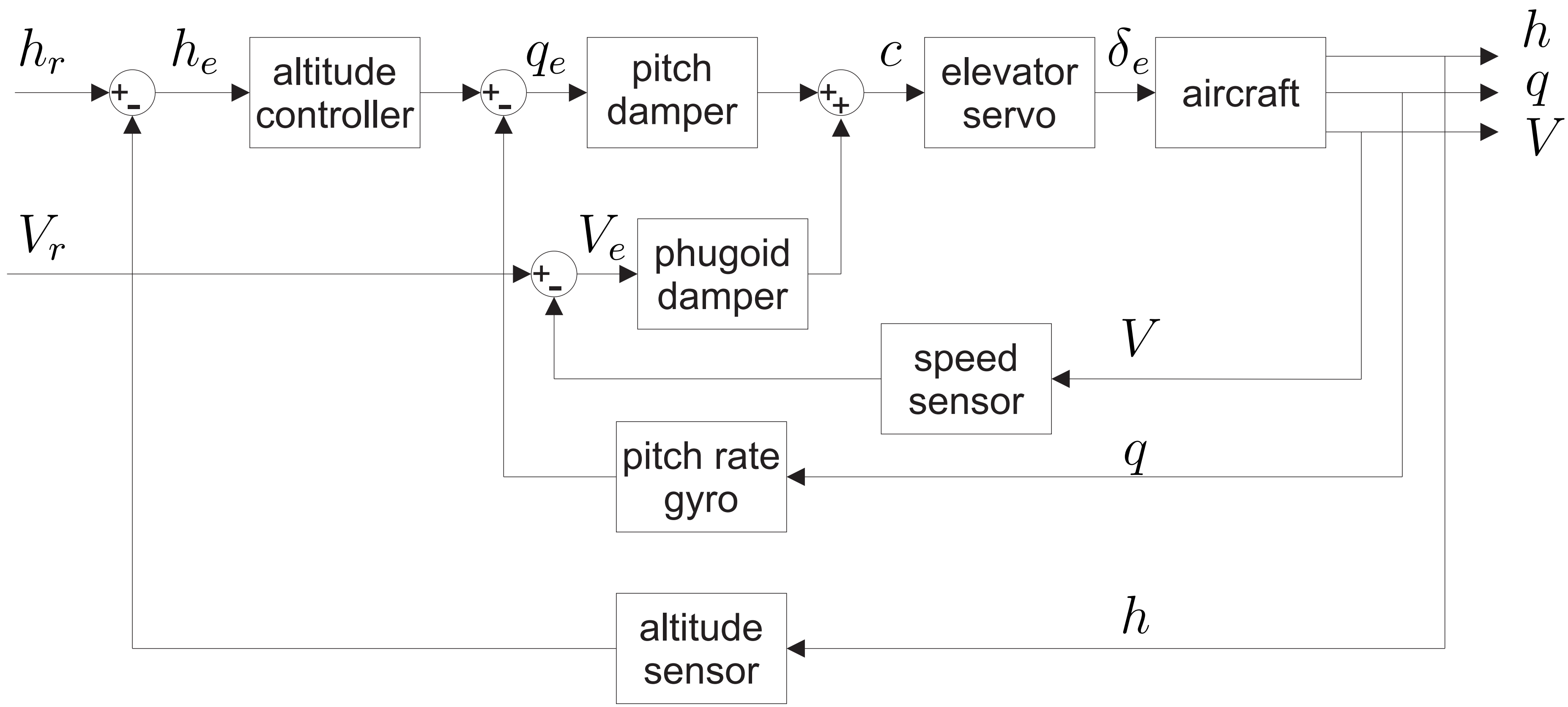
With the well known relationship $\gamma(s) = \theta(s) - \alpha(s)$, we get:

$$h(s) = \frac{V_0}{s}\left(\theta(s) - \alpha(s)\right)$$

$$\frac{h(s)}{\delta_e(s)} = \frac{V_0}{s}\left(\frac{\theta(s)}{\delta_e(s)} - \frac{\alpha(s)}{\delta_e(s)}\right)$$

Altitude hold mode

Block diagram of an altitude hold loop:



Altitude hold mode

The altitude hold mode controller incorporates three types of action:

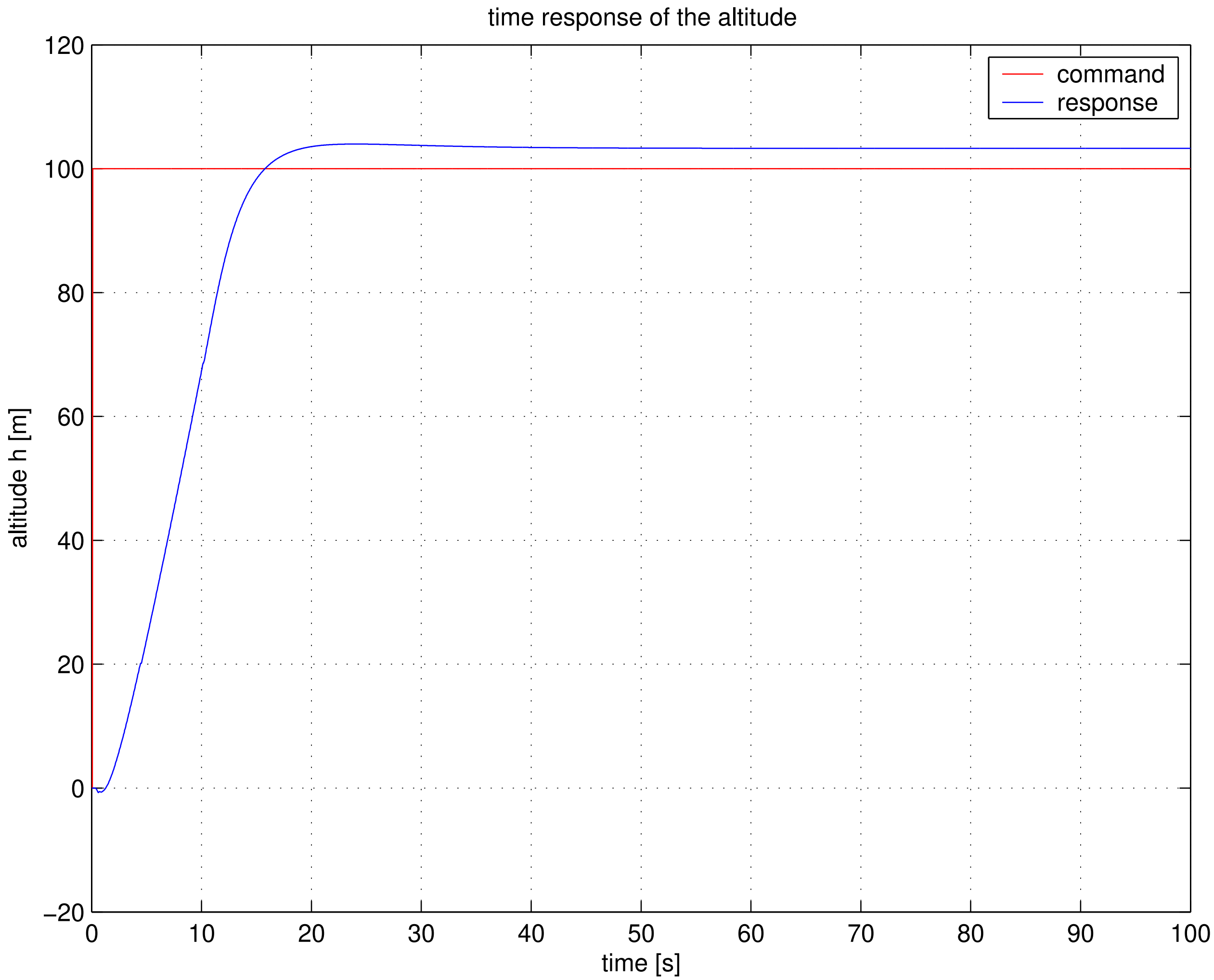
- Proportional: reduces rise time.
- Integrating: eliminates steady state error.
- Derivative: improves transient response(damping).

$$K(s) = K_P + \frac{K_I}{s} + K_D s = \frac{K_D s^2 + K_P s + K_I}{s}$$

The input reference altitude h_r is set by the pilot through the Mode Control Panel.

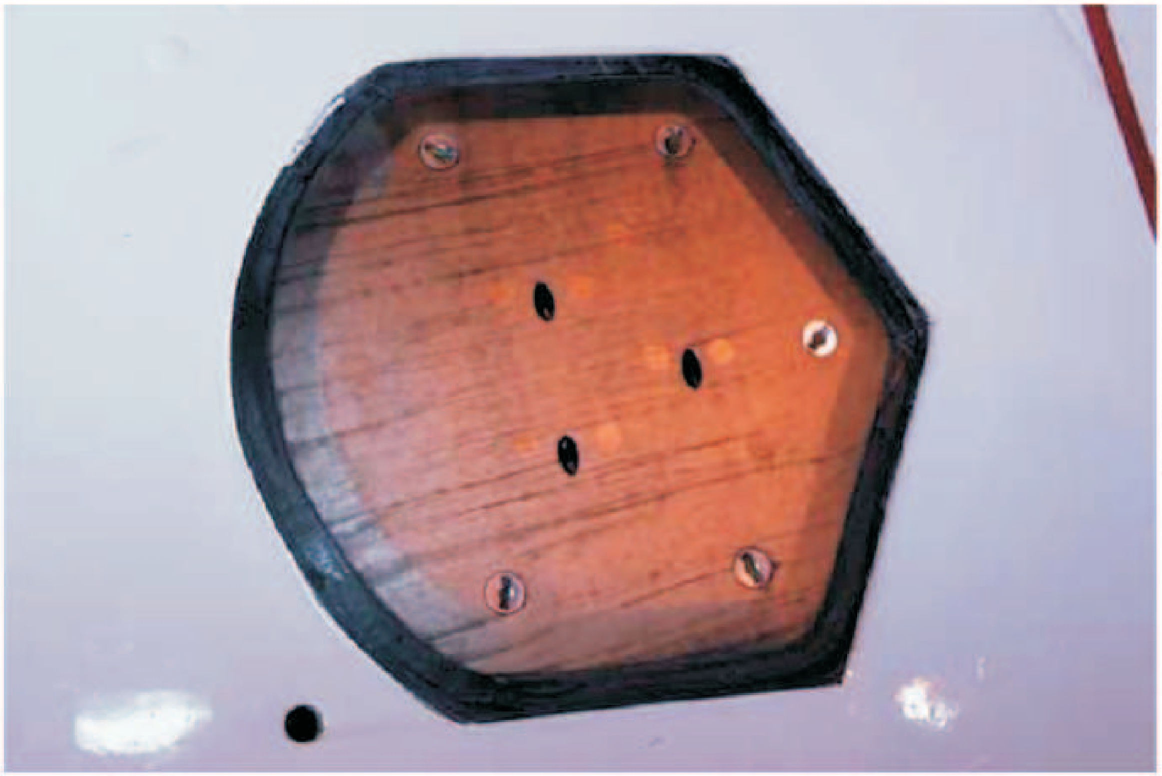
Altitude hold mode

Example Bunting UAV (what additional control is needed?):



Altitude hold mode

Picture of static holes for altitude sensor of BAe RJ 85.

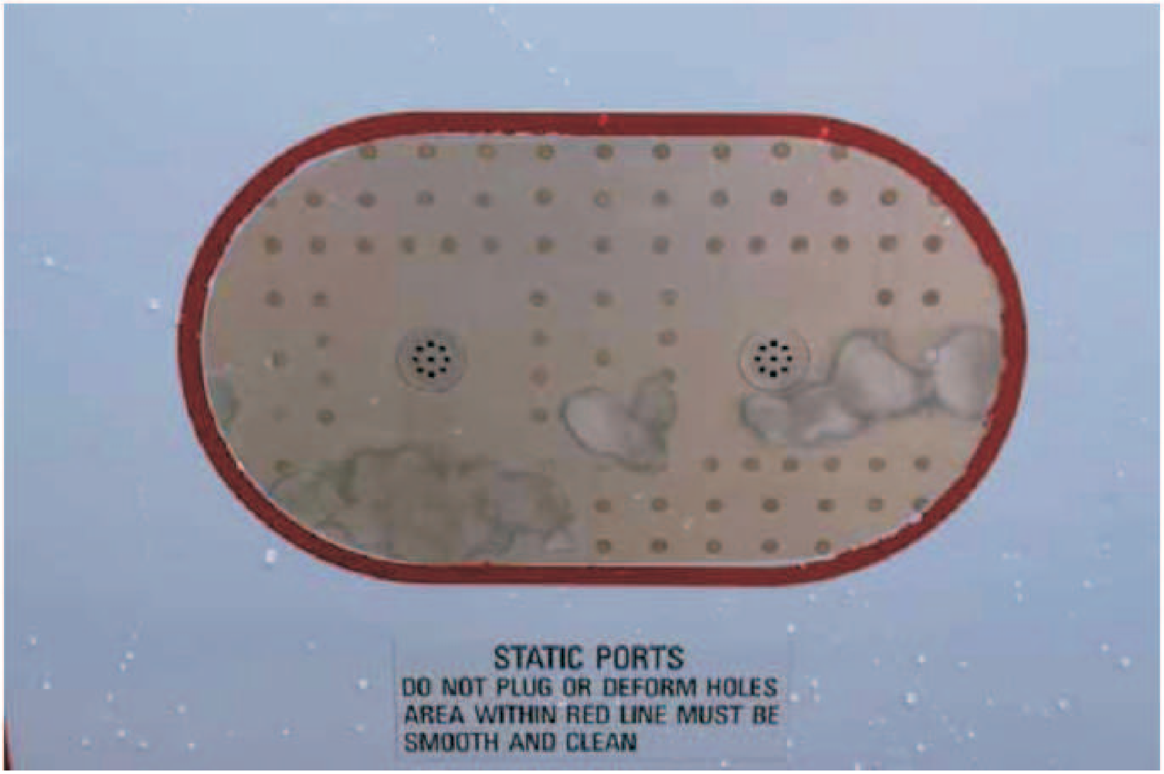


Altitude hold mode

Picture of GPS antenna for altitude sensor of BAe RJ 85.

Altitude hold mode

Picture of static holes for altitude sensor of Airbus A319.



Airspeed hold mode

Use: prevents pilot having to constantly control airspeed during the flight.

Autopilot mode analysis/design, 2 important questions:

- Which type of control device do we use?

Airspeed hold mode

Use: prevents pilot having to constantly control airspeed during the flight.

Autopilot mode analysis/design, 2 important questions:

- Which type of control device do we use? $\longrightarrow$**throttle**
- What type of feedback loop structure (sensor)?

Airspeed hold mode

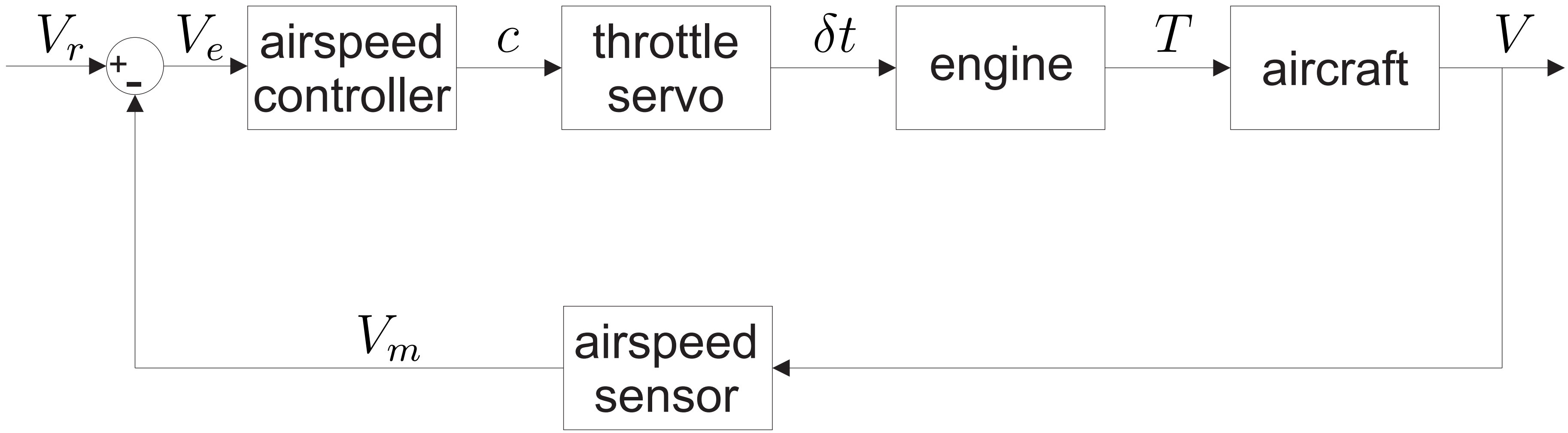
Use: prevents pilot having to constantly control airspeed during the flight.

Autopilot mode analysis/design, 2 important questions:

- Which type of control device do we use? $\longrightarrow$**throttle**
- What type of feedback loop structure (sensor)? $\longrightarrow$**airspeed sensor**

Airspeed hold mode

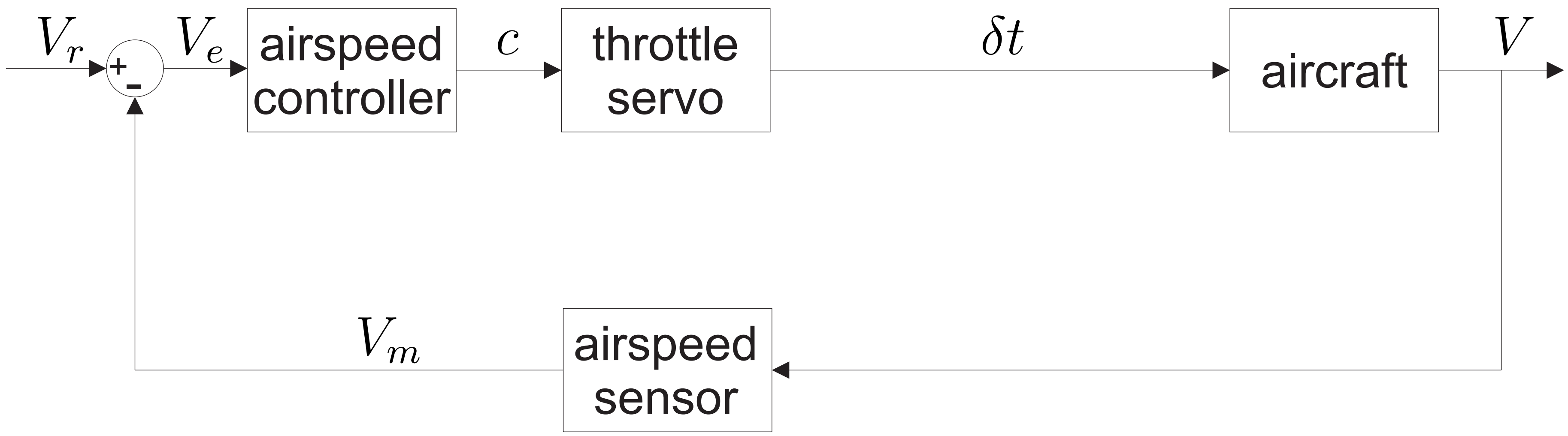
Block diagram of an airspeed hold loop:



Or lump engine + aircraft model together:

Airspeed hold mode

Block diagram of an airspeed hold loop:



Airspeed hold mode

The airspeed hold mode controller incorporates three types of action:

- Proportional: reduces rise time.
- Integrating: eliminates steady state error.
- Derivative: improves transient response(damping).

$$K(s) = K_P + \frac{K_I}{s} + K_D s = \frac{K_D s^2 + K_P s + K_I}{s}$$

The input reference airspeed V_r is set by the pilot through the Mode Control Panel.

Airspeed hold mode

Airspeed sensor model representations:

- GPS airspeed calculations:

$$H_{\text{airspeedsensor}} \approx 1$$

- Pitot static tube:

$$H_{\text{airspeedsensor}} \approx \frac{1}{\tau_{\text{airspeedsensor}} s + 1}$$

Airspeed hold mode

Picture of pitot-tube for airspeed sensor on BAe RJ 85.

Airspeed hold mode

Engine and throttle servo model representations:

- Throttle servo:

$$H_{\text{servo}} \approx \frac{1}{\tau_{\text{servo}} s + 1}$$

- Engine model (simplified!):

$$H_{\text{engine}} = \frac{\Delta T\left(s\right)}{\delta_T\left(s\right)} \approx K_T \frac{1}{\tau_{\text{engine}} s + 1}$$

- **Worksheet.**

Airspeed hold mode

$$H_{\text{engine}} =\approx K_T \frac{1}{\tau_{\text{engine}} s + 1}$$

- With K_T = 500000 and τ_{engine} = 3
- How many seconds after applying full throttle (i.e. a step on δ_T) does the engine deliver 400kN of thrust?

Laplace transforms:

$$\begin{array}{lcl} 1 & \rightarrow & \frac{1}{s} \\ e^{-at} & \rightarrow & \frac{1}{s+a} \\ t & \rightarrow & \frac{1}{s^2} \end{array}$$

Airspeed hold mode

$$U(s) = \frac{1}{s} \rightarrow Y(s) = K_T \frac{1}{\tau s + 1} \frac{1}{s}$$

Airspeed hold mode

$$U(s) = \frac{1}{s} \rightarrow Y(s) = K_T \frac{1}{\tau s + 1} \frac{1}{s}$$

$$Y(s) = K_T \frac{a}{\tau s + 1} + K_T \frac{b}{s}$$

$$as + b(\tau s + 1) = 1 \rightarrow b = 1, \quad a = -\tau$$

$$Y(s) = K_T \frac{-\tau}{\tau s + 1} + K_T \frac{1}{s}$$

$$h_{\text{engine}}(t) = -K_T e^{-\frac{1}{\tau} t} + K_T$$

$$t = 4.8 \text{ s}$$

Airspeed hold mode

$$H_{\text{engine}} = \approx K_T \frac{1}{\tau_{\text{engine}} s + 1}$$

- With K_T = 500000 and τ_{engine} = 3
- If the throttle is inreased linearly **from 0 to 1** during 10 seconds, what will be the thrust error at 10 seconds?

Laplace transforms:

$$\begin{array}{lcl} 1 & \rightarrow & \frac{1}{s} \\ e^{-at} & \rightarrow & \frac{1}{s+a} \\ t & \rightarrow & \frac{1}{s^2} \end{array}$$

Airspeed hold mode

$$U(s) = \frac{1}{10}\frac{1}{s^2} \rightarrow Y(s) = \frac{K_T}{10}\frac{1}{\tau s + 1}\frac{1}{s^2}$$

Airspeed hold mode

$$U(s) = \frac{1}{10}\frac{1}{s^2} \rightarrow Y(s) = \frac{K_T}{10}\frac{1}{\tau s + 1}\frac{1}{s^2}$$

$$Y(s) = \frac{K_T}{10}\left(\frac{a}{\tau s + 1} + \frac{b}{s^2} + \frac{c}{s}\right)$$

$$as^2 + b(\tau s + 1) + cs(\tau s + 1) = 1 \rightarrow b = 1, \quad c = -\tau \quad a = \tau^2$$

$$Y(s) = \frac{K_T}{10}\left(\tau\frac{\tau}{\tau s + 1} + \frac{1}{s^2} - \tau\frac{1}{s}\right)$$

$$h_{\text{engine}}(t) = -\frac{K_T}{10}\left(\tau e^{-\frac{1}{\tau}t} + t - \tau\right)$$

$$h_{\text{engine}}(10) = 3.6 \cdot 10^5 \text{ N}$$

Airspeed hold mode

The effects of the throttle can also be entered in the state space models:

$$\dot{u} \sim \delta_{th} \quad \Rightarrow \quad \dot{u} = f\left(\hat{u}, \alpha, \theta, \frac{q\bar{c}}{V}\right) + K_{th}\delta_{th}$$

Extended state space system:

$$\begin{bmatrix} \dot{\hat{u}} \\ \dot{\alpha} \\ \dot{\theta} \\ \frac{\dot{q}\bar{c}}{V} \end{bmatrix} = A \begin{bmatrix} \hat{u} \\ \alpha \\ \theta \\ \frac{q\bar{c}}{V} \end{bmatrix} + \begin{bmatrix} B & \begin{matrix} K_{th} \\ 0 \\ 0 \\ 0 \end{matrix} \end{bmatrix} \begin{bmatrix} \delta_e \\ \delta_{th} \end{bmatrix}$$

Flight path angle hold mode

Use: prevents pilot having to constantly control climb or descent rate during a climb or descent.

Autopilot mode analysis/design, 2 important questions:

- Which type of control device do we use?

Flight path angle hold mode

Use: prevents pilot having to constantly control climb or descent rate during a climb or descent.

Autopilot mode analysis/design, 2 important questions:

- Which type of control device do we use? $\longrightarrow$**Elevator**
- What type of feedback loop structure (sensor)?

Flight path angle hold mode

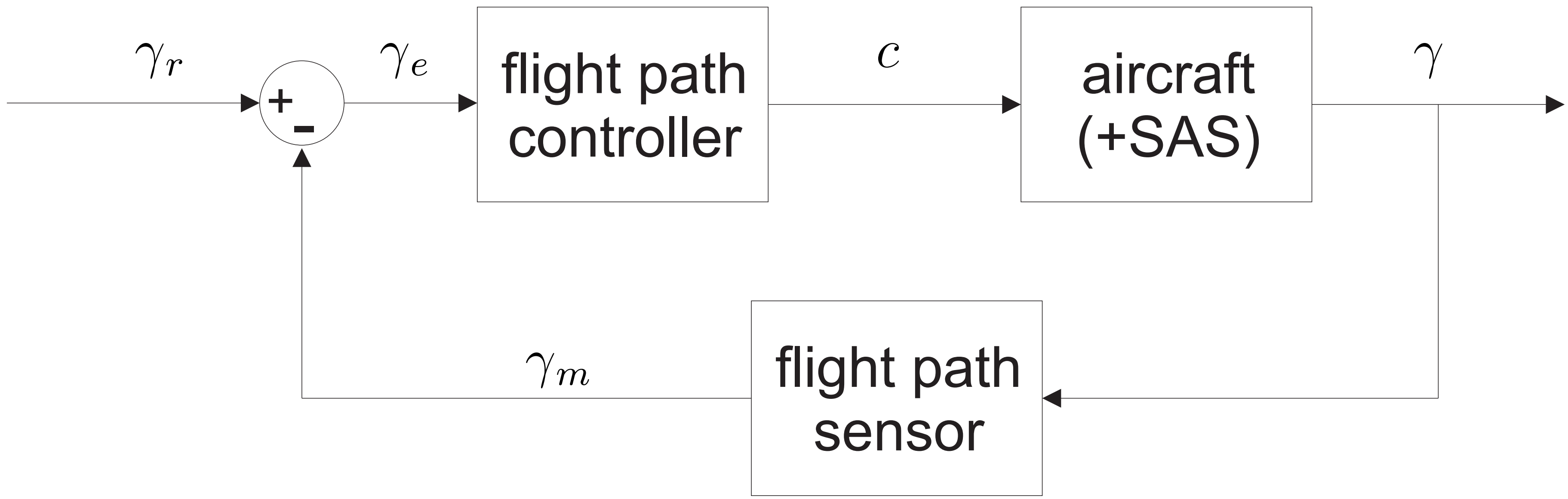
Use: prevents pilot having to constantly control climb or descent rate during a climb or descent.

Autopilot mode analysis/design, 2 important questions:

- Which type of control device do we use? $\longrightarrow$ **Elevator**
- What type of feedback loop structure (sensor)? $\longrightarrow \gamma$
- Flight path angle is not directly measurable, so use: $\gamma = \theta - \alpha$, vertical gyro and an angle of attack sensor.

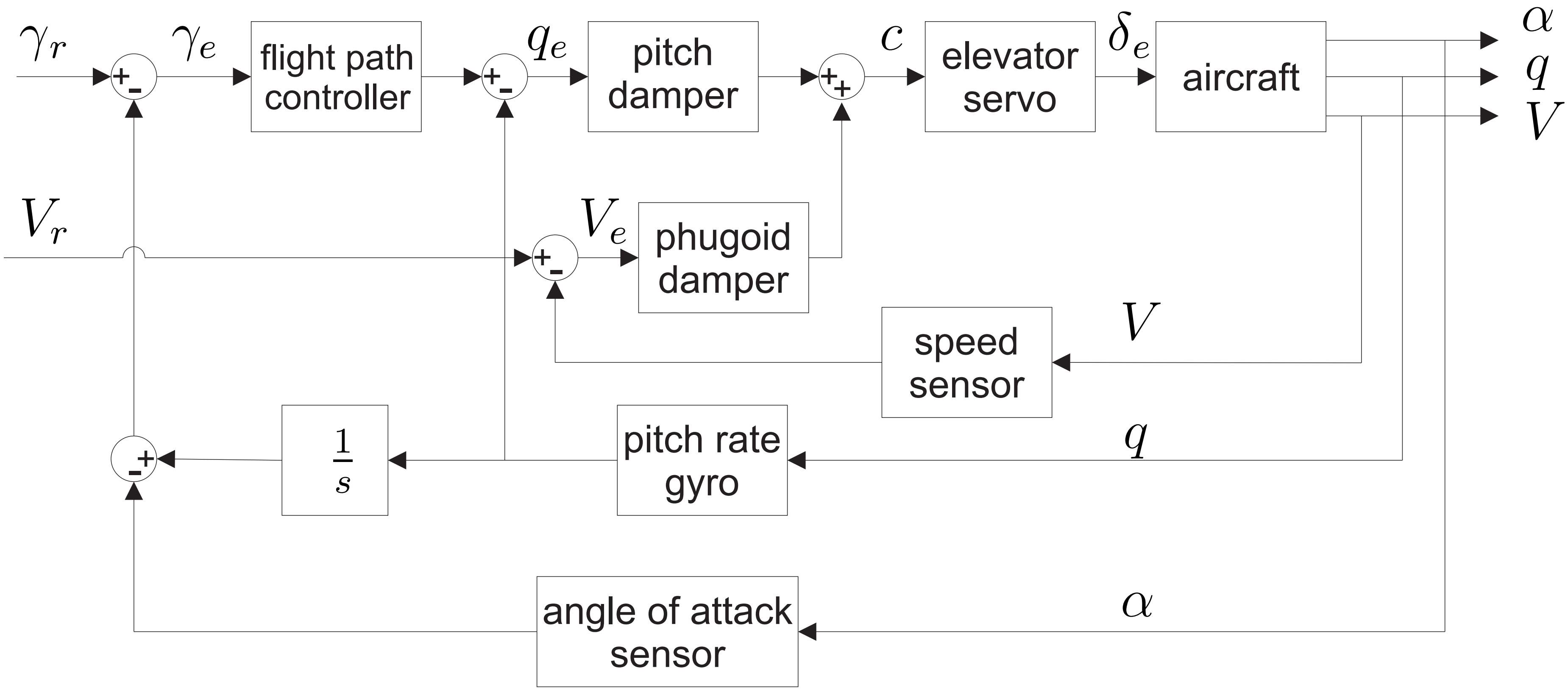
Flight path angle hold mode

Simplified block diagram of a flight path angle hold loop:



Flight path angle hold mode

Detailed block diagram of a flight path angle hold loop:



Flight path angle hold mode

The flight path angle hold mode controller incorporates three types of action:

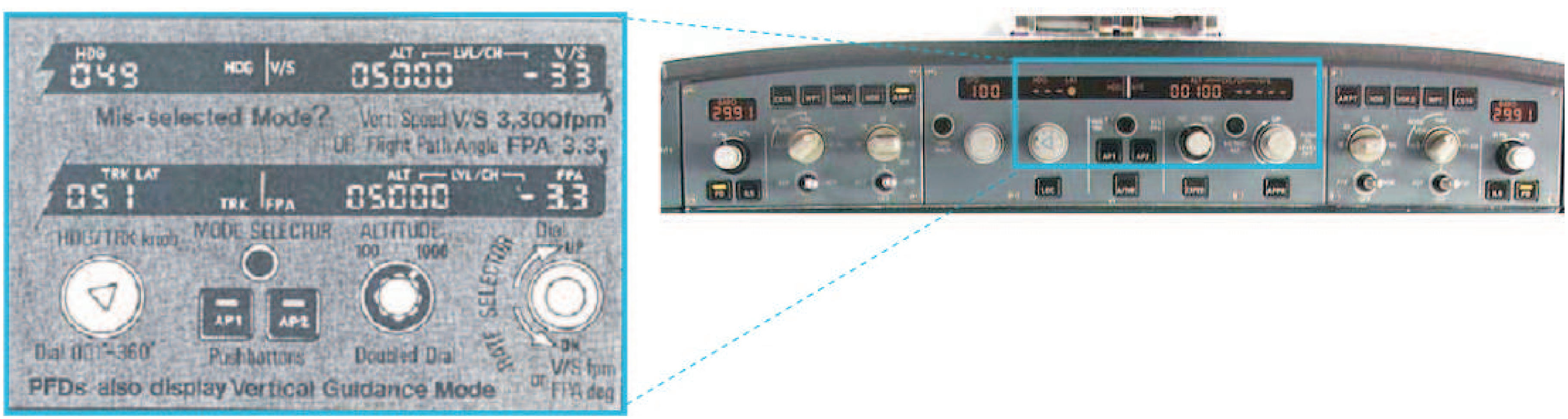
- Proportional: reduces rise time.
- Integrating: eliminates steady state error.
- Derivative: improves transient response(damping).

$$K(s) = K_P + \frac{K_I}{s} + K_D s = \frac{K_D s^2 + K_P s + K_I}{s}$$

The input reference flight path angle γ_r is set by the pilot on the Mode Control Panel.

Flight path angle hold mode

- Air Inter Flight 148, 20 january 1992, crashed into a mountain on approach to Strasbourg, 87 fatalities.
- Flight crew inadvertently selected 3300 fpm descent rate on approach instead of the desired 3.3 degrees flight path angle.



Summary

- For each control/hold mode there are 2 important questions:
  - Which type of control device?
  - Which type of feedback loop structure(sensor)?
- Keep controller as simple as possible:
  - Proportional only if acceptable
  - Add derivative control to improve damping.
  - Add integration control to reduce steady state errors.
- The closed loop system must comply with the airworthiness requirements.

Summary

- Take into account the inner loop, when constructing the outer loop.
- Keep in mind the possible effects on the closed loop system of:
  - Servo break frequency.
  - Flight condition (airspeed, altitude).
- Sometimes the input/output channels need to be extended to obtain the signals that are required in the feedback loops.

What's next?

- Lateral autopilot modes.
- Navigational autopilot systems.
  - Longitudinal
  - Lateral

AE4-301 Automatic Flight Control System Design

Ewoud Smeur
Delft University of Technology - AE - C&S

October 8, 2020

AE4-301 Automatic Flight Control System Design

Ewoud Smeur
Delft University of Technology - AE - C&S

October 11, 2020

Course structure

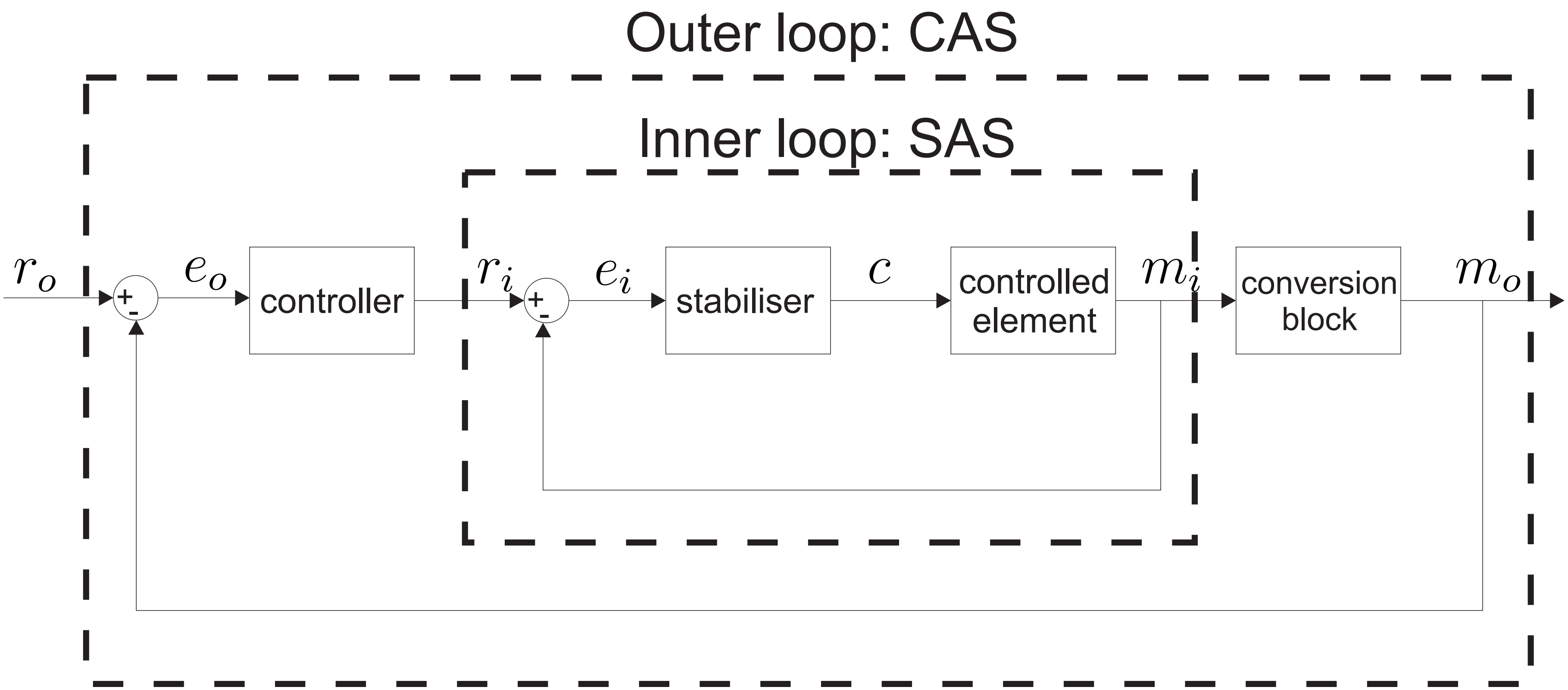
- Introduction automated flight control and recap flight dynamics (L1).
- Control theory (L2-L8).
- Performance and handling qualities (L9).
- Static/dynamic stability augmentation (L10-L11).
- Longitudinal autopilot modes (L12).
- **Lateral autopilot modes (today).**
- **Navigational autopilot modes (today/Friday).**

Lecture 13 structure

- Introduction
- Lateral control/hold functions
  - Roll angle hold mode
  - Coordinated roll angle hold mode
  - Heading angle hold mode
- Summary

Introduction

Position of the Control Augmentation System (CAS) in the global Flight Control System.



Introduction

Autopilot system use:

- For safety reasons.
- Lowers pilot workload (on long flights).

Autopilot system categories:

- Control or hold certain flight parameters.
- Perform navigational tasks.

Introduction

Lateral control/hold functions:

- Roll angle hold mode
- Coordinated roll angle hold mode
- Heading angle hold mode

Roll angle hold mode

Use: prevents pilot having to constantly control roll angle. Roll angle is set to zero (wing leveler).

Autopilot mode analysis/design, 2 important questions:

- Which type of control device do we use?

Roll angle hold mode

Use: prevents pilot having to constantly control roll angle. Roll angle is set to zero (wing leveler).

Autopilot mode analysis/design, 2 important questions:

- Which type of control device do we use? $\longrightarrow$**Ailerons**
- What type of feedback loop structure (sensor)?

Roll angle hold mode

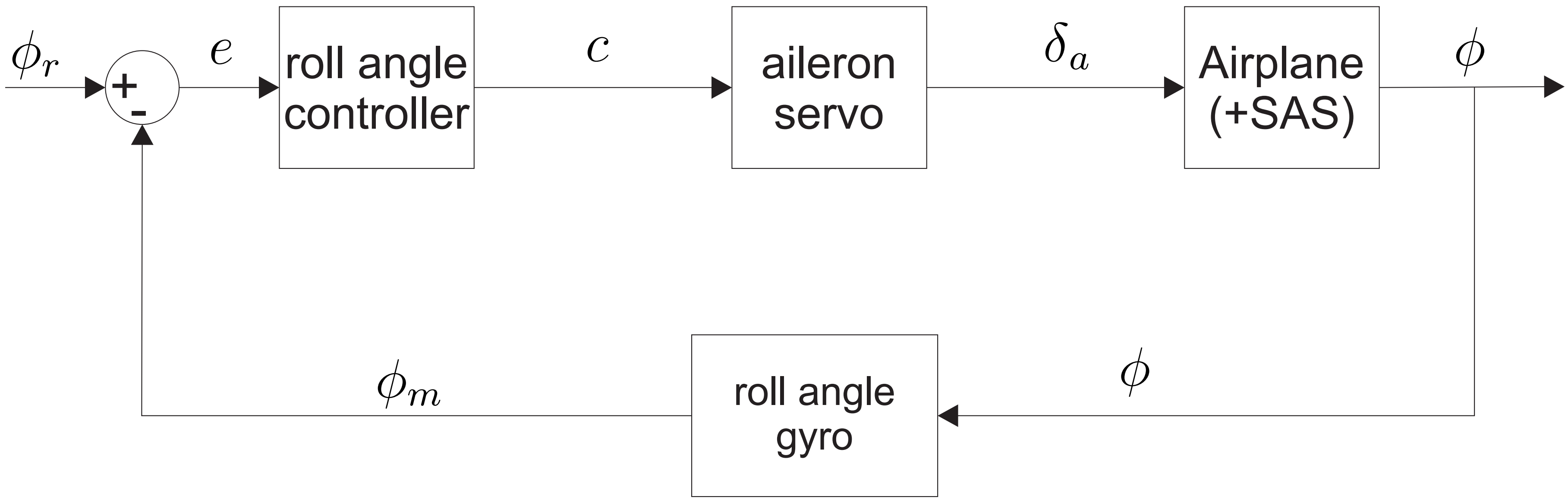
Use: prevents pilot having to constantly control roll angle. Roll angle is set to zero (wing leveler).

Autopilot mode analysis/design, 2 important questions:

- Which type of control device do we use? $\longrightarrow$**Ailerons**
- What type of feedback loop structure (sensor)? $\longrightarrow$**Roll angle gyro**

Roll angle hold mode

Block diagram of a roll angle hold loop:

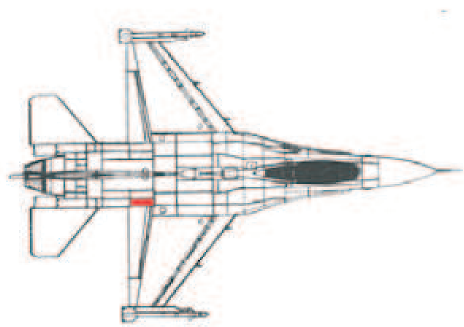


Roll angle hold mode

Picture of aileron tab servo of the BAe RJ85:

Roll angle hold mode

Picture of flaperon
(combined flap-aileron)
actuator of the F-16



Roll angle hold mode

Gyro and servo model representations(as before):

- Roll angle gyro: $H_{gyro} \approx 1$.
- Aileron servo: $H_{servo} \approx \frac{1}{\tau_{servo}s+1}$.

The roll angle controller incorporates three types of action:

- Proportional: reduces rise time.
- Integrating: eliminates steady state error.
- Derivative: improves transient response(damping).

$$K(s) = K_P + \frac{K_I}{s} + K_D s = \frac{K_D s^2 + K_P s + K_I}{s}$$

Roll angle hold mode

What about the reference input signal ϕ_r?

This roll angle is indirectly defined on the mode control panel by the pilot. In practice: limited ϕ established for $\Delta\psi$.

Assumption: only important degree of freedom is roll.
Reduced transfer function:

$$\frac{\phi(s)}{\delta_a(s)} = \frac{L_{\delta_a}}{s(s - L_p)}$$

- What's the sign of L_{δ_a}?

Roll angle hold mode

What about the reference input signal ϕ_r?

This roll angle is indirectly defined on the mode control panel by the pilot. In practice: limited ϕ established for $\Delta\psi$.

Assumption: only important degree of freedom is roll.
Reduced transfer function:

$$\frac{\phi(s)}{\delta_a(s)} = \frac{L_{\delta_a}}{s(s - L_p)}$$

- What's the sign of L_{δ_a}? negative, since a positive aileron deflection results in a negative rolling moment.

Roll angle hold mode

Cessna C182

Dimensions:

-wing area: 16.2 m2
-mean chord: 1.5 m
-wingspan 10.97 m

Flight condition: cruise

-altitude: 1500 m
-speed: 241.5 km/h
-weight: 12 kN

Roll angle hold mode

Cessna 182 complete aileron to roll angle transfer function:

```
        -75.05 s^2 - 97.57 s - 603.3
--------------------------------------------
s^4 + 14.37 s^3 + 28.23 s^2 + 137.5 s + 2.45
```

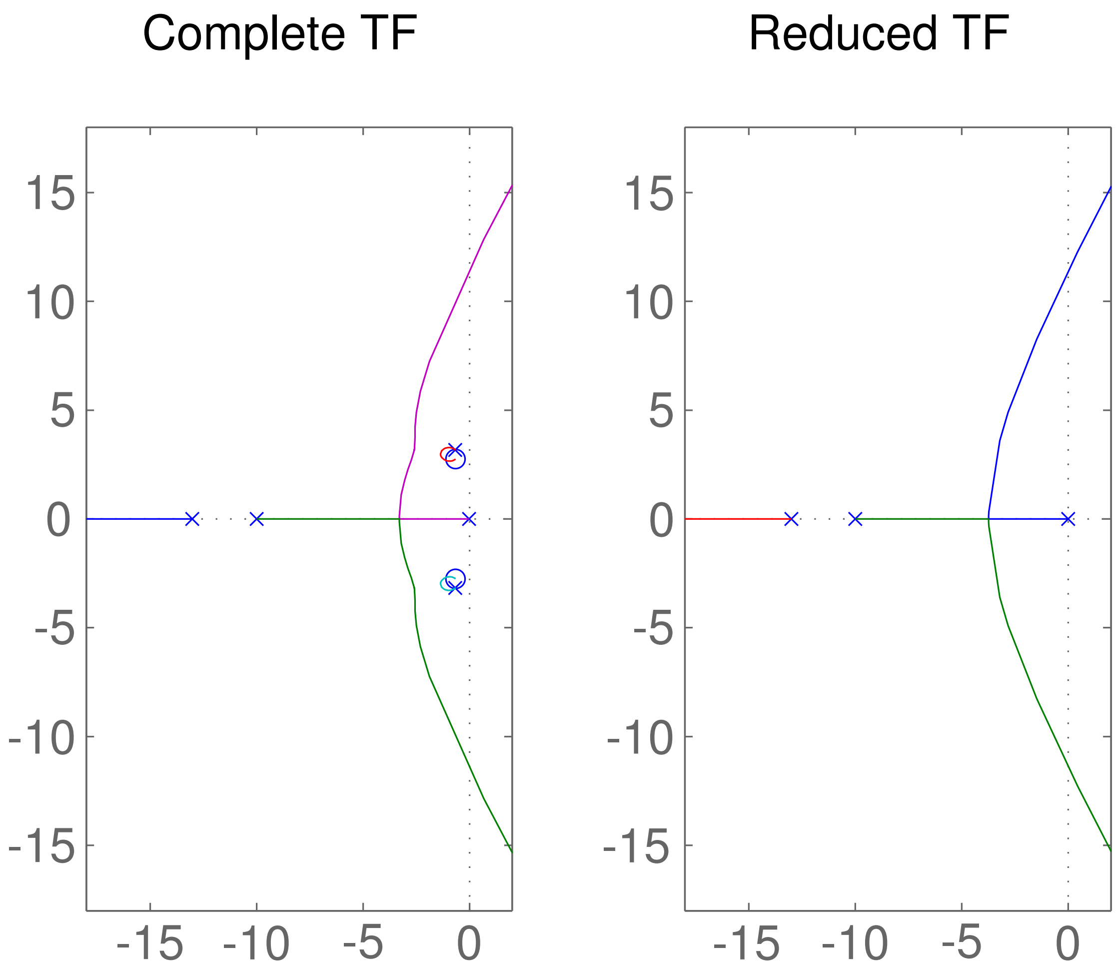
Reduced aileron to roll angle transfer function ($\frac{\phi(s)}{\delta_a(s)} = \frac{L_{\delta_a}}{s(s-L_p)}$):

```
   -75.1
----------
s^2 + 13 s
```

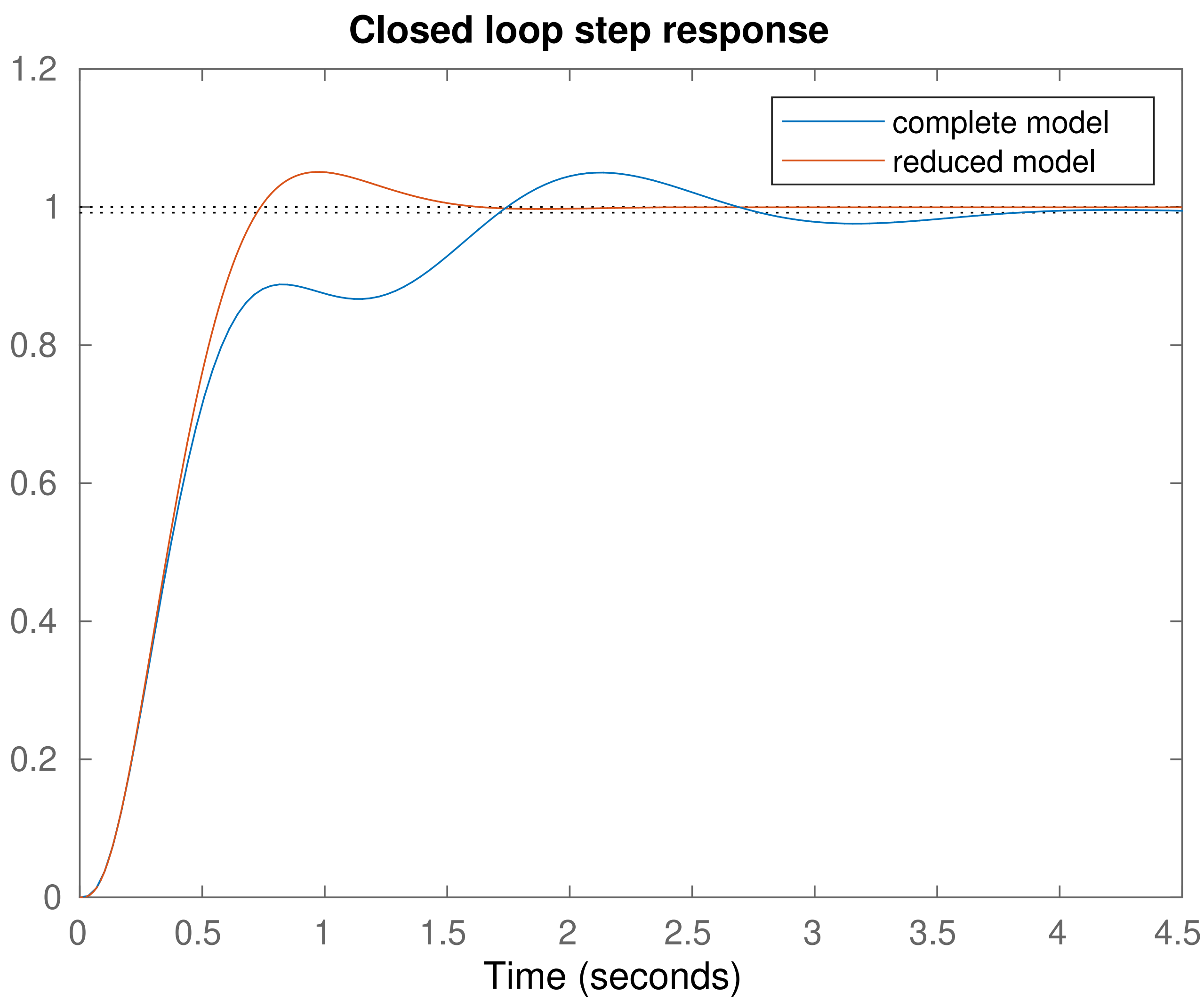
Roll angle hold mode

- **Simulink**: open loop response.
- **Sisotool**: Design roll angle feedback gain. Trade-off between rise time and damping.
- **Simulink**: Closed loop response.
- Compare complete TF and reduced TF.

Roll angle hold mode

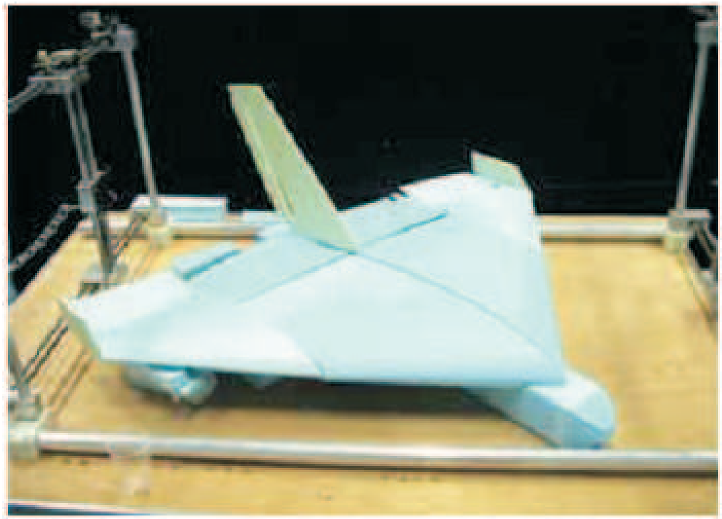


Roll angle hold mode



Roll angle hold mode

Example roll angle hold mode Bunting UAV.



Reduced aileron to roll angle transfer function:

$$\frac{\phi\left(s\right)}{\delta_a\left(s\right)} = \frac{-655.6}{s^2 + 52.52s}$$

Roll angle hold mode

Bunting UAV roll angle hold mode example:

- **Simulink**: open loop response.
- **Sisotool**: Design roll angle feedback gain. This time aim for damping of 1 (critically damped).
- **Simulink**: Closed loop response.

Roll angle hold mode

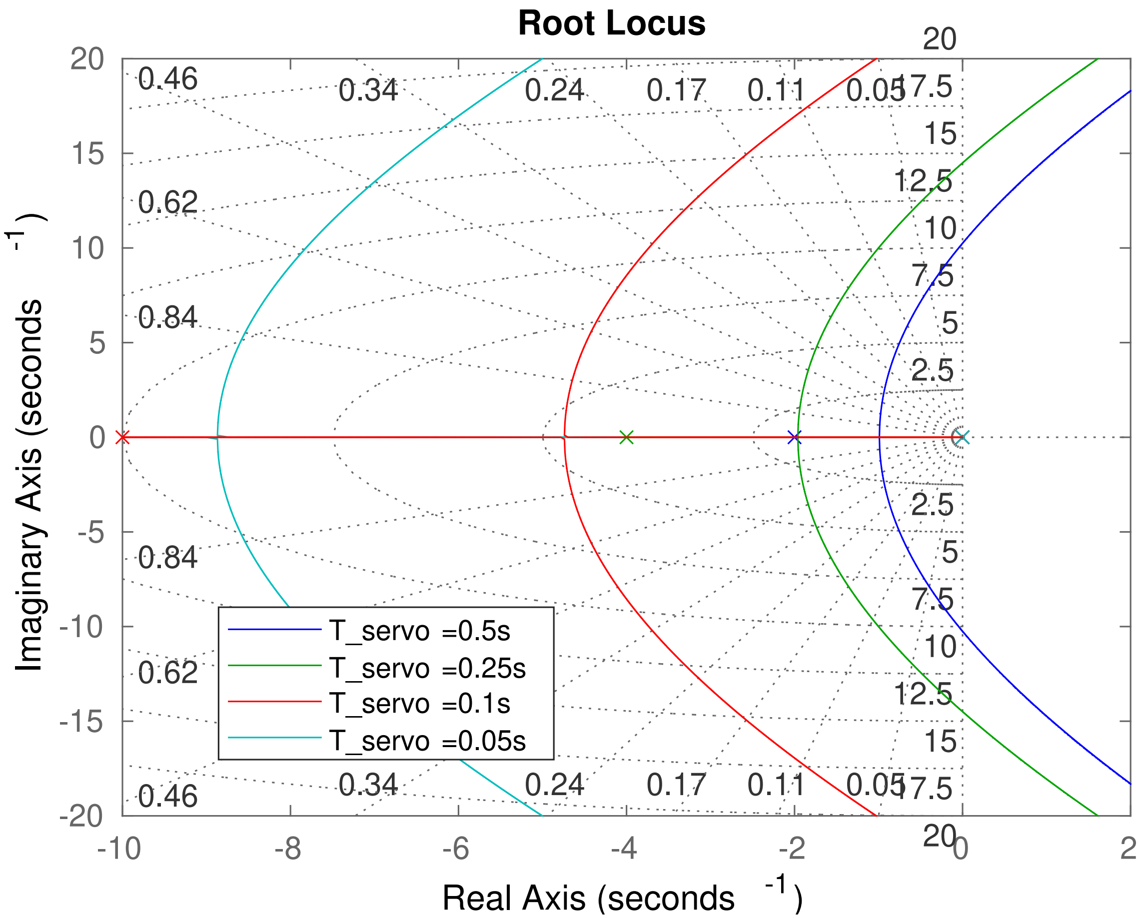
Bunting UAV roll angle hold mode example:

- **Simulink**: open loop response.
- **Sisotool**: Design roll angle feedback gain. This time aim for damping of 1 (critically damped).
- **Simulink**: Closed loop response.

What is the effect of the servo time constant on the root locus?

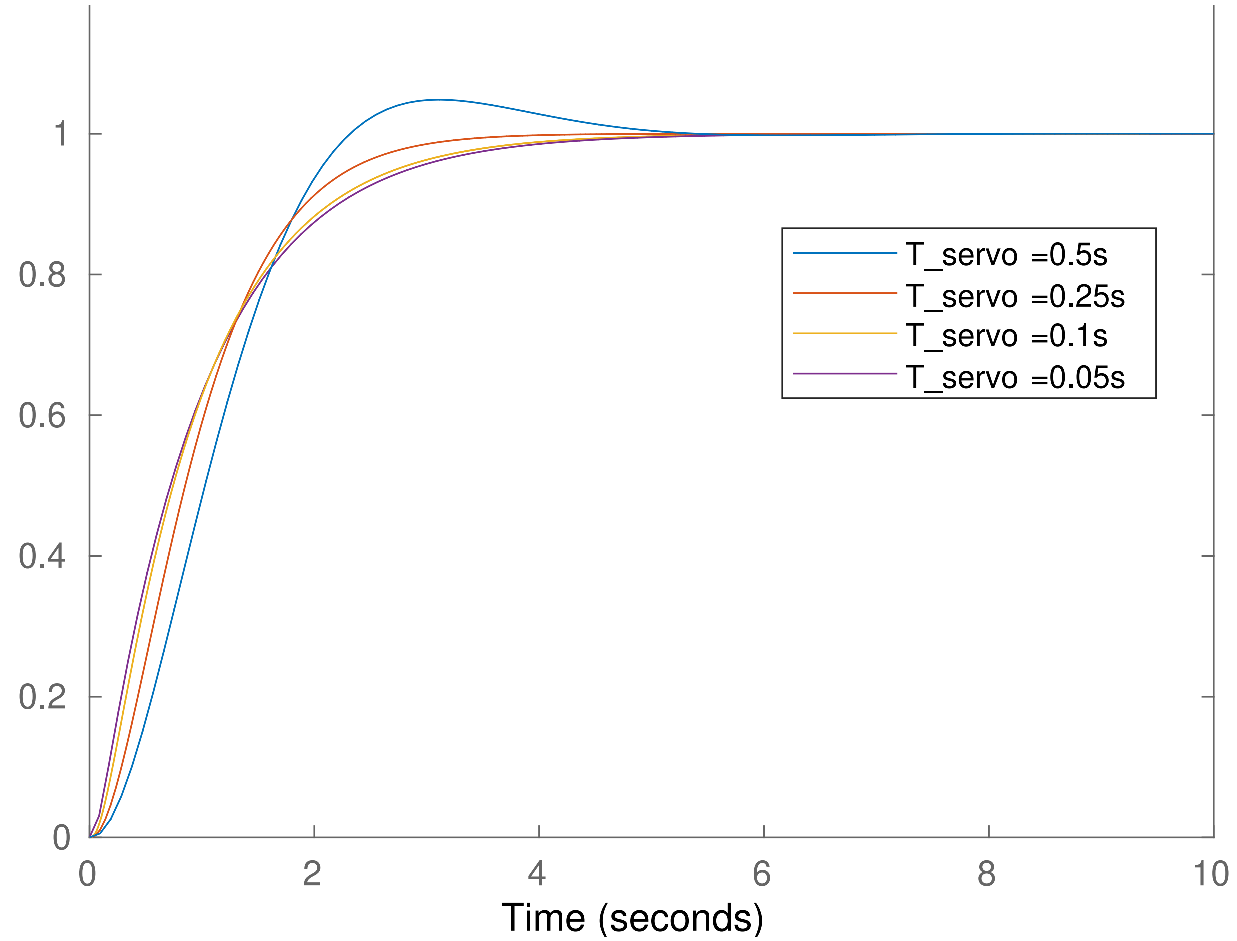
Roll angle hold mode

Effect of servo time constant



Roll angle hold mode

Step response



Roll angle hold mode

The value of the roll hold control gain in cruise $K_\phi = -0.08$, does not guarantee compliance with airworthiness regulations in other flight conditions. In fact the roll hold control gain is different in each flight condition.

Flight condition	K
launch	-0.439
climb	-0.336
cruise	-0.08
patrol	-0.481
descent	-0.336
recovery	-0.6

Gain Scheduling is required.

Coordinated roll angle hold mode

Extension of roll angle hold mode which reduces sideslip angle β to zero.
Use:

- Align aircraft with streamlines: reduced drag.
- No lateral accelerations: more comfort for crew and pax.

Autopilot mode analysis/design, 2 important questions:

- Which type of control device do we use?

Coordinated roll angle hold mode

Extension of roll angle hold mode which reduces sideslip angle β to zero.
Use:

- Align aircraft with streamlines: reduced drag.
- No lateral accelerations: more comfort for crew and pax.

Autopilot mode analysis/design, 2 important questions:

- Which type of control device do we use? $\longrightarrow$**Rudder**
- What type of feedback loop structure (sensor)?

Coordinated roll angle hold mode

Extension of roll angle hold mode which reduces sideslip angle β to zero.
Use:

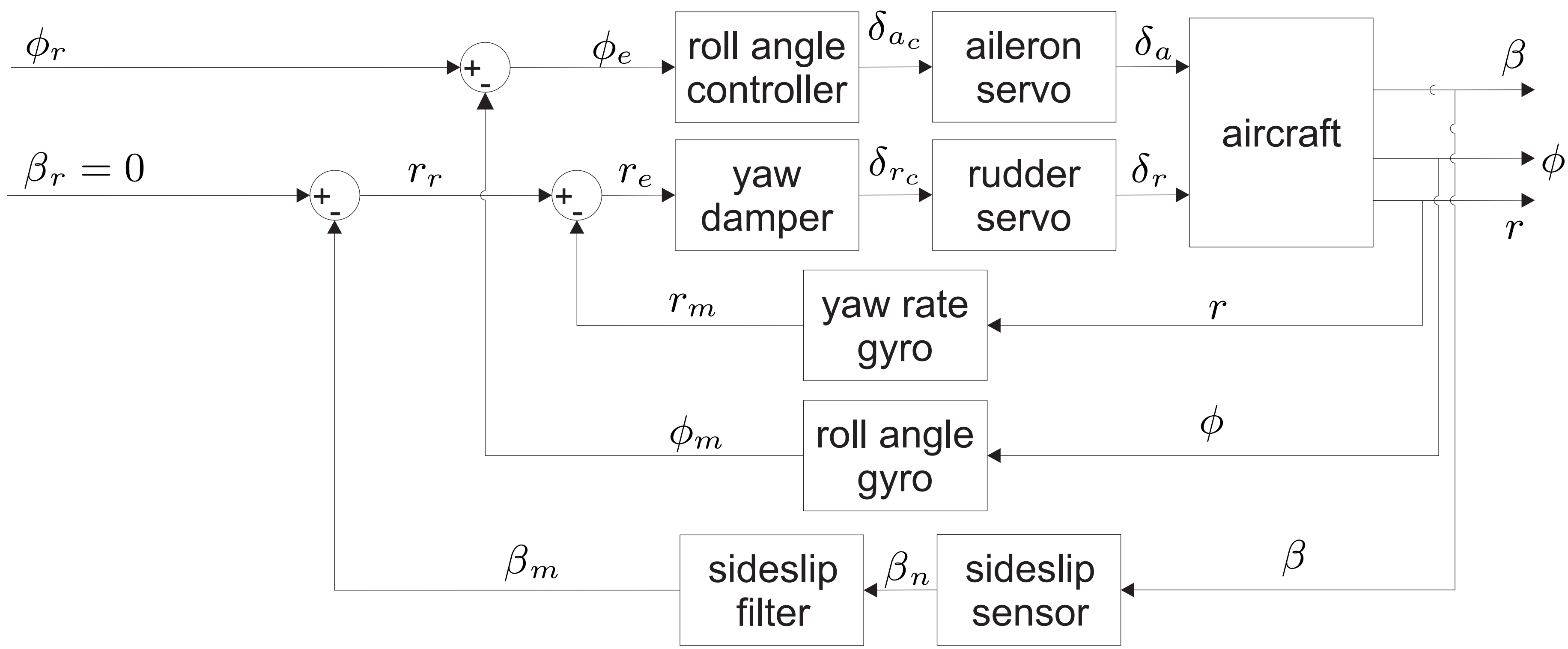
- Align aircraft with streamlines: reduced drag.
- No lateral accelerations: more comfort for crew and pax.

Autopilot mode analysis/design, 2 important questions:

- Which type of control device do we use? $\longrightarrow$**Rudder**
- What type of feedback loop structure (sensor)? $\longrightarrow$**Sideslip sensor**

Coordinated roll angle hold mode

Detailed block diagram of a coordinated roll angle hold mode.



Heading angle hold mode

Use: Prevents the pilot from constantly having to control heading angle during flight.

Use roll angle control as inner loop for heading control. Maintain roll angle until required heading change is obtained.
Autopilot mode analysis/design, 2 important questions:

- Which type of control device do we use?

Heading angle hold mode

Use: Prevents the pilot from constantly having to control heading angle during flight.

Use roll angle control as inner loop for heading control. Maintain roll angle until required heading change is obtained.
Autopilot mode analysis/design, 2 important questions:

- Which type of control device do we use? $\longrightarrow$**Aileron**
- What type of feedback loop structure (sensor)?

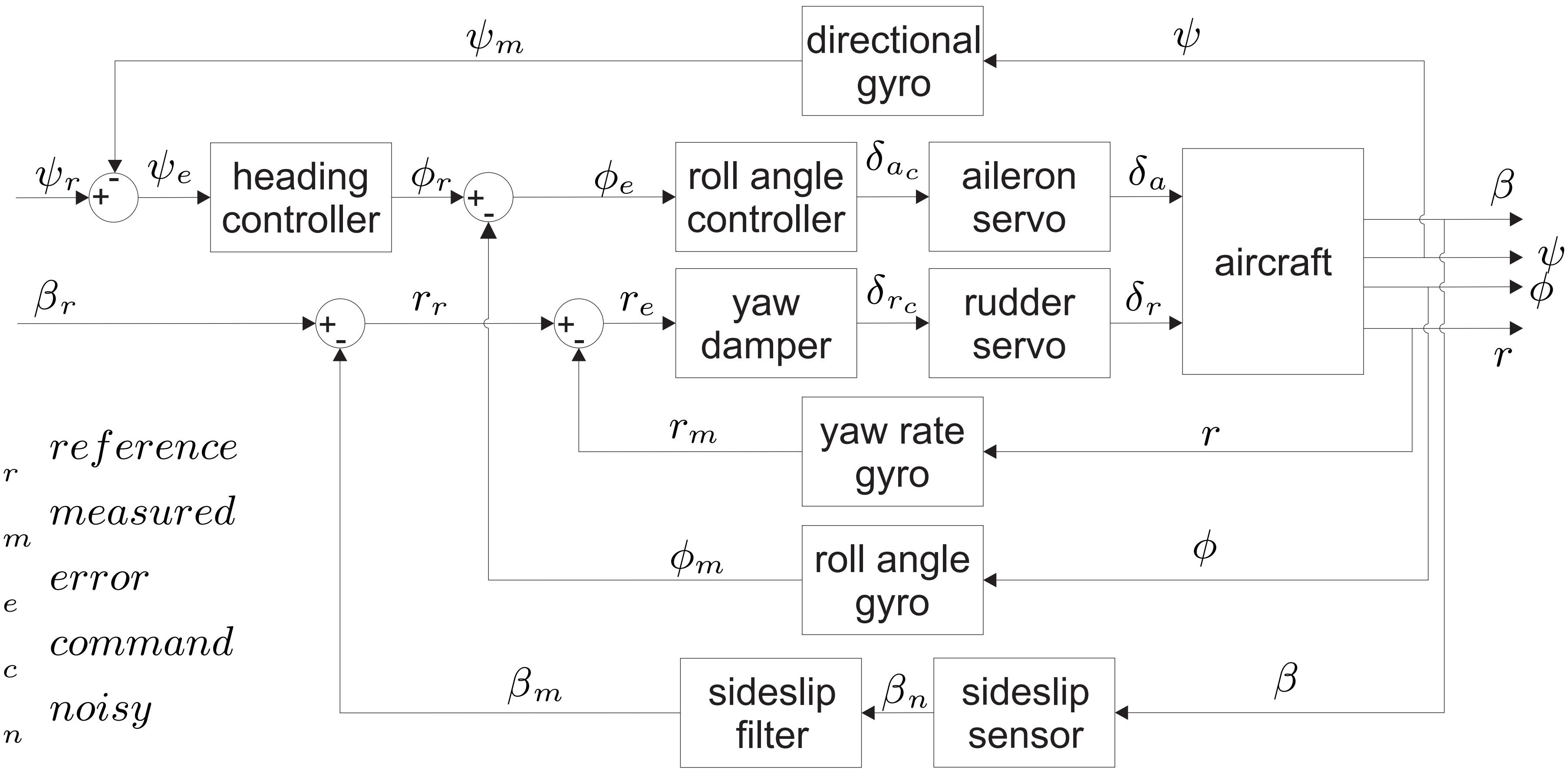
Heading angle hold mode

Use: Prevents the pilot from constantly having to control heading angle during flight.

Use roll angle control as inner loop for heading control. Maintain roll angle until required heading change is obtained.
Autopilot mode analysis/design, 2 important questions:

- Which type of control device do we use? $\longrightarrow$**Aileron**
- What type of feedback loop structure (sensor)? $\longrightarrow$**Directional gyro**

Heading angle hold mode



$\square_r$ *reference*

$\square_m$ *measured*

$\square_e$ *error*

$\square_c$ *command*

$\square_n$ *noisy*

Heading angle hold mode

Gyro representations(as before):

- Directional gyro: $H_{gyro} \approx 1$.

The heading angle controller incorporates three types of action:

- Proportional: reduces rise time.
- Integrating: eliminates steady state error.
- Derivative: improves transient response(damping).

$$K\left(s\right) = K_P + \frac{K_I}{s} + K_D s = \frac{K_D s^2 + K_P s + K_I}{s}$$

Heading angle hold mode

What about the reference input signal ψ_r?
The heading angle is defined on the mode control panel by the pilot.

- The value of the heading ψ is not directly available in our model, we only have β, ϕ, p, r.
- Recall the state space form:

$$\begin{bmatrix} \dot{\beta} \\ \dot{\phi} \\ \frac{\dot{p}b}{2V} \\ \frac{\dot{r}b}{2V} \end{bmatrix} = \begin{bmatrix} \hat{A}_{11} & \hat{A}_{12} & \hat{A}_{13} & \hat{A}_{14} \\ \hat{A}_{21} & \hat{A}_{22} & \hat{A}_{23} & \hat{A}_{24} \\ \hat{A}_{31} & \hat{A}_{32} & \hat{A}_{33} & \hat{A}_{34} \\ \hat{A}_{41} & \hat{A}_{42} & \hat{A}_{43} & \hat{A}_{44} \end{bmatrix} \begin{bmatrix} \beta \\ \phi \\ \frac{pb}{2V} \\ \frac{rb}{2V} \end{bmatrix} + \begin{bmatrix} \hat{B}_{11} & \hat{B}_{12} \\ \hat{B}_{21} & \hat{B}_{22} \\ \hat{B}_{31} & \hat{B}_{32} \\ \hat{B}_{41} & \hat{B}_{42} \end{bmatrix} \begin{bmatrix} \delta_a \\ \delta_r \end{bmatrix}$$

Heading angle hold mode

The heading angle can be obtained by two methods:

1. By using the kinematic relationship between ψ and r.
2. By looking at the force equations of an aircraft in a turn.

1. Kinematic approach:

$$\dot{\psi} = q\frac{\sin\phi}{\cos\theta} + r\frac{\cos\phi}{\cos\theta}$$

Assume $q = 0$, then

$$\dot{\psi} = r\frac{\cos\phi}{\cos\theta}$$

Heading angle hold mode

1. Kinematic approach:

Assume θ is constant and ϕ is small, then

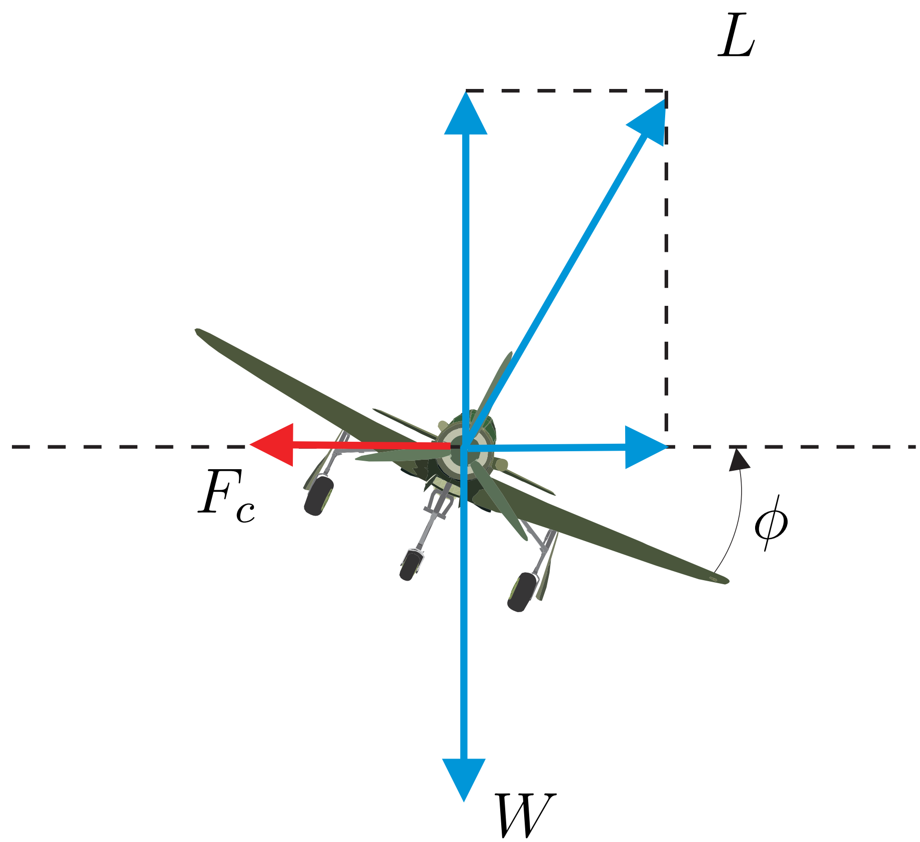
$$\psi = \frac{1}{\cos\theta} \int_{t_0}^{t_0+\Delta t} r \cdot dt$$

or in the frequency domain:

$$\psi = \frac{r}{s\cos\theta}$$

2. Force balance approach:

Heading angle hold mode



$$F_c = L \sin\phi = mR_t\dot{\psi}^2 = mU\dot{\psi}$$

Heading angle hold mode

$$L = \frac{mg}{\cos\phi} \rightarrow L\sin\phi = \frac{mg\sin\phi}{\cos\phi} = mU\dot{\psi} \rightarrow \frac{g\sin\phi}{\cos\phi} = U\dot{\psi}$$

$$g\tan\phi = U\dot{\psi} \rightarrow \dot{\psi} = \frac{g}{U}\tan\phi$$

For small ϕ:

$$\dot{\psi} = \frac{g}{U}\phi \rightarrow \psi = \int_{t_0}^{t_0+\Delta t} \frac{g}{U}\phi dt$$

Which gives in the frequency domain: $\psi = \frac{g}{Us}\phi$

Heading angle hold mode

Summary. How to calculate heading ψ:

1. Use yaw rate:

$$\psi = \frac{r}{s\cos\theta}$$

2. Use roll angle ϕ:

$$\psi = \frac{g}{Us}\phi$$

The second option is preferred, since there are fewer assumptions.

Summary

- For each control/hold mode there are 2 important questions:
  - Which type of control device?
  - Which type of feedback loop structure(sensor)?
- Keep controller as simple as possible:
  - Proportional only if acceptable
  - Add derivative control to improve damping.
  - Add integration control to reduce steady state errors.
- The closed loop system must comply with the airworthiness requirements.

Summary

- Take into account the inner loop, when constructing the outer loop.
- Keep in mind the possible effects on the closed loop system of:
  - Servo break frequency.
  - Flight condition (airspeed, altitude). **Gain scheduling.**
- Sometimes the input/output channels need to be extended to obtain the signals that are required in the feedback loops.

AE4-301 Automatic Flight Control System Design

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October 11, 2020

AE4-301 Automatic Flight Control System Design

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Delft University of Technology - AE - C&S

October 15, 2020

Course structure

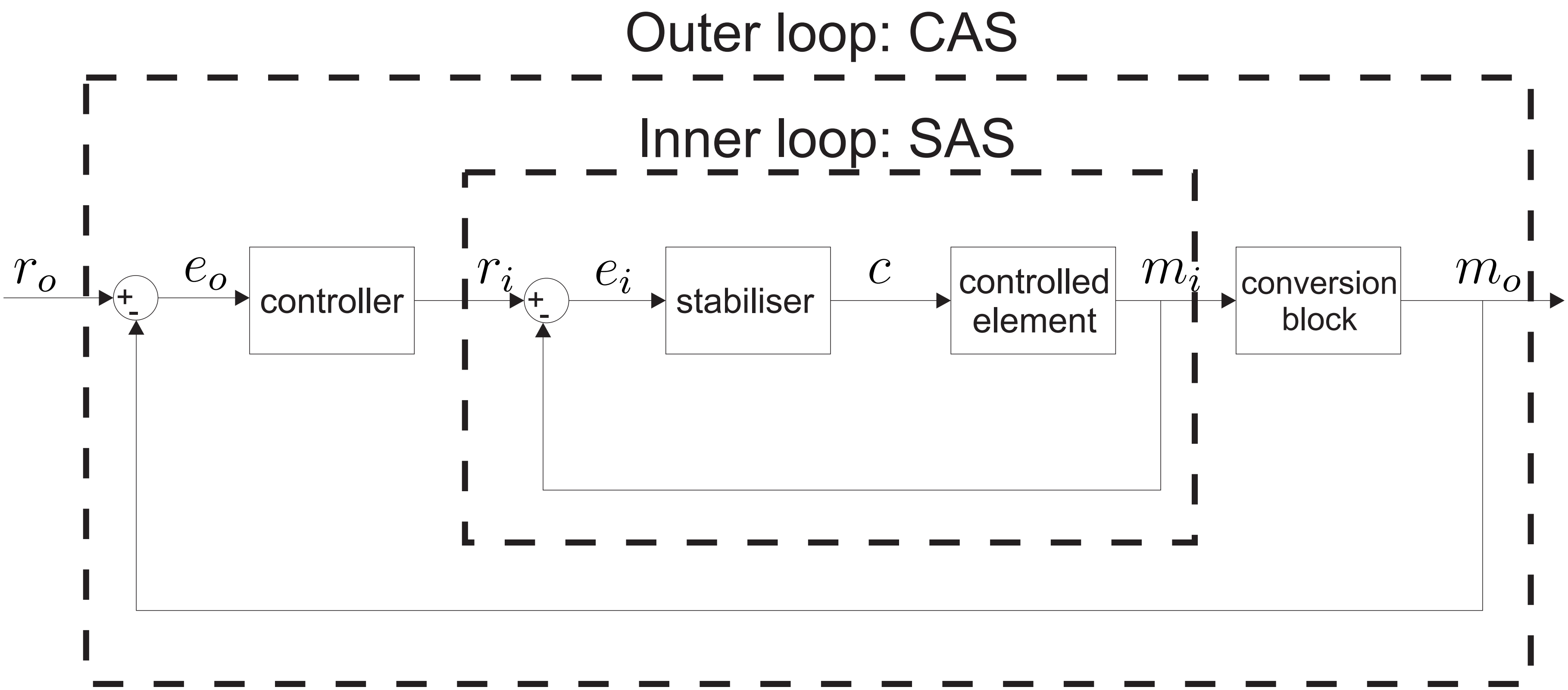
- Introduction automated flight control and recap flight dynamics.
- Control theory.
- Performance and handling qualities.
- Static/dynamic stability augmentation.
- Longitudinal autopilot modes.
- Lateral autopilot modes.
- **Navigational autopilot systems**

Lecture 14 structure

- Introduction
- Longitudinal navigational functions
  - Glideslope hold mode
  - Automatic flare mode
- Lateral navigational functions
  - Localizer hold mode
  - VOR hold mode
- Summary

Introduction

Position of the Control Augmentation System (CAS) in the global Flight Control System.



Introduction

Autopilot system use:

- For safety reasons.
- Lowers pilot workload (on long flights).

Navigational mode: **Adding an extra loop.**

Autopilot systems performing navigational tasks:

- Civil airliners
- Military transport aircraft
- Business jets, etc

Introduction

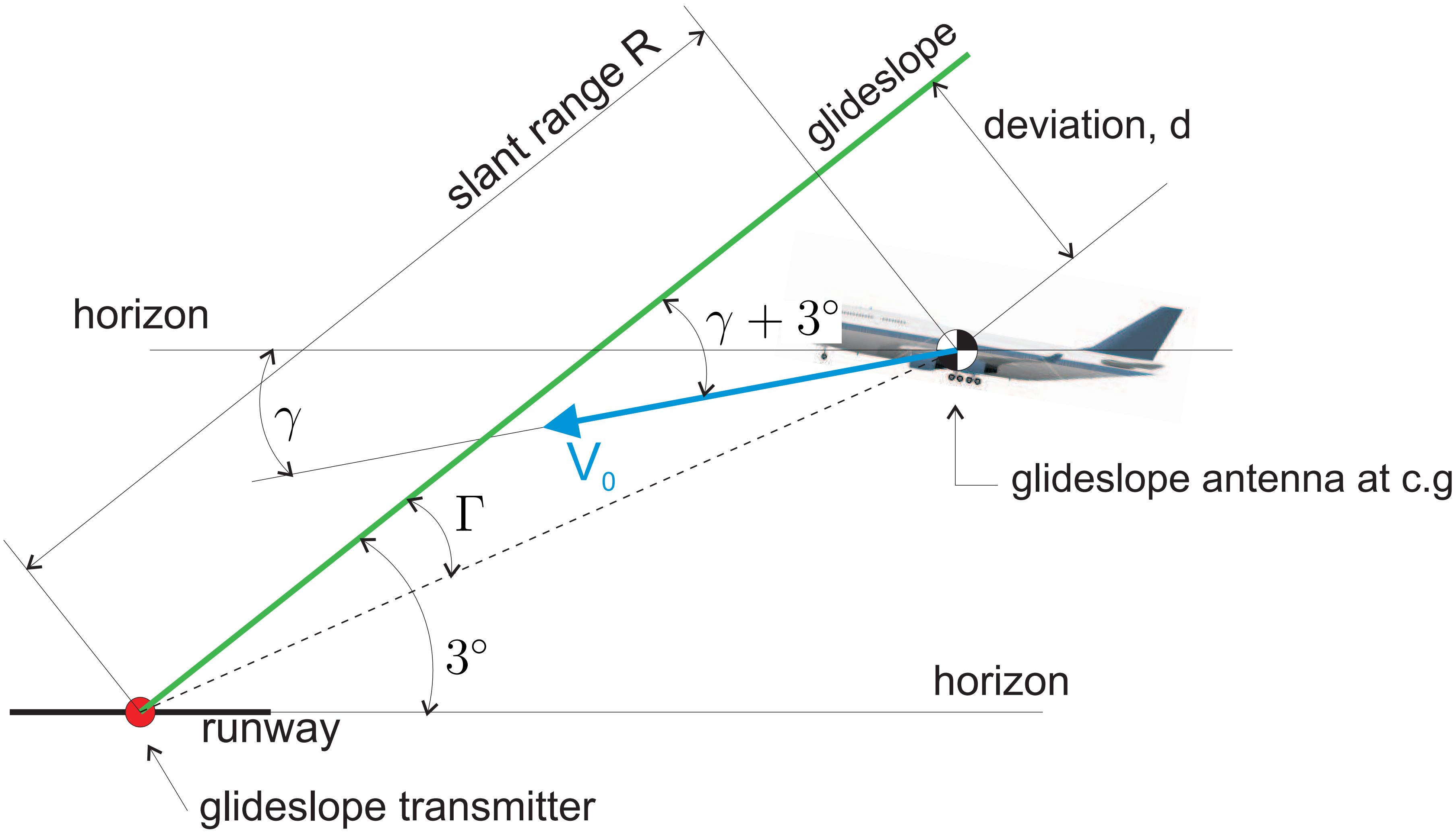
Longitudinal navigational functions:

- Glideslope hold mode.
- Automatic flare mode.

Glide slope

- The glideslope intercept and hold mode is part of an automatic Instrument Landing System (ILS) approach.
- It is more accurate to automatically follow the glideslope than the pilot performing this task manually.
- Assumptions:
  - Pitch attitude control system is already present (**Lecture 12**).
  - Speed control system is already present (**Lecture 12**).

Glide slope



slant range R
glideslope
deviation, d
horizon
$\gamma + 3°$
γ
V_0
glideslope antenna at c.g.
Γ
3°
horizon
runway
glideslope transmitter

Glide slope

Assumptions:

- Airplane glide slope antenna coincides with c.g.
- c.g. is driven along the glideslope.
- Glideslope error angle Γ is sensed by on-board glideslope receiver.
- Airplane is kept on glideslope by pitch attitude command.
- Autothrottle is used for speed control.

Glide slope

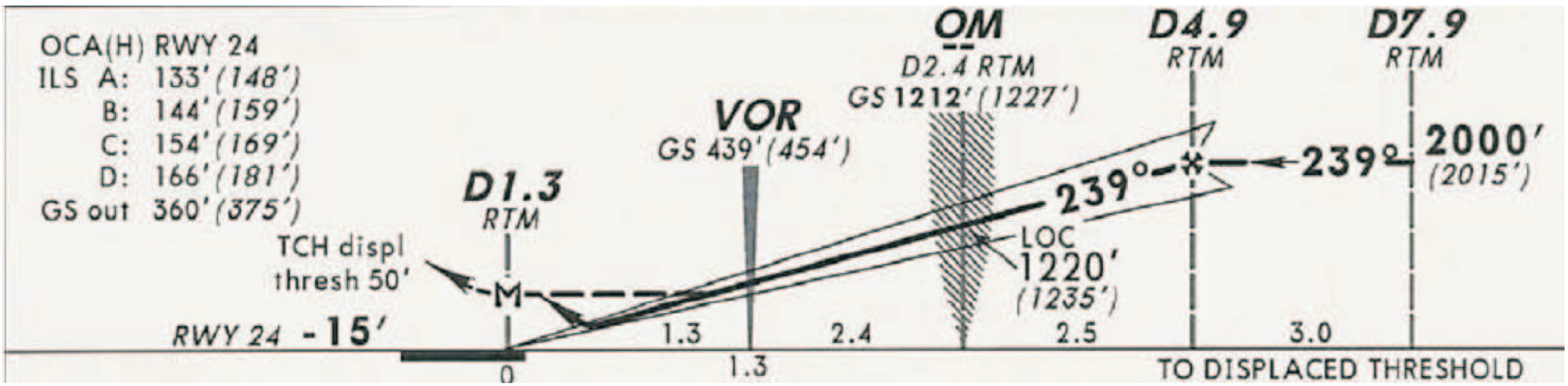
Glideslope antenna

Glide slope

Picture of glideslope antennas on the BAe RJ 85:

Glide slope

Picture of glideslope markings on civil aviation charts:



Glide slope

Quantity to be controlled (reduced): glideslope error angle Γ

- Deviation velocity:

$$\dot{d} = V_0 \sin\left(\left(\gamma + 3^\circ\right)\frac{\pi}{180}\right) \approx V_0\left(\gamma + 3^\circ\right)\frac{\pi}{180}$$

- Deviation distance:

$$d \approx \int V_0\left(\gamma + 3^\circ\right)\frac{\pi}{180}dt$$

$$d\left(s\right) \approx \frac{V_0}{s}\frac{\pi}{180}\left(\gamma + 3^\circ\right)$$

Glide slope

Deviation distance:

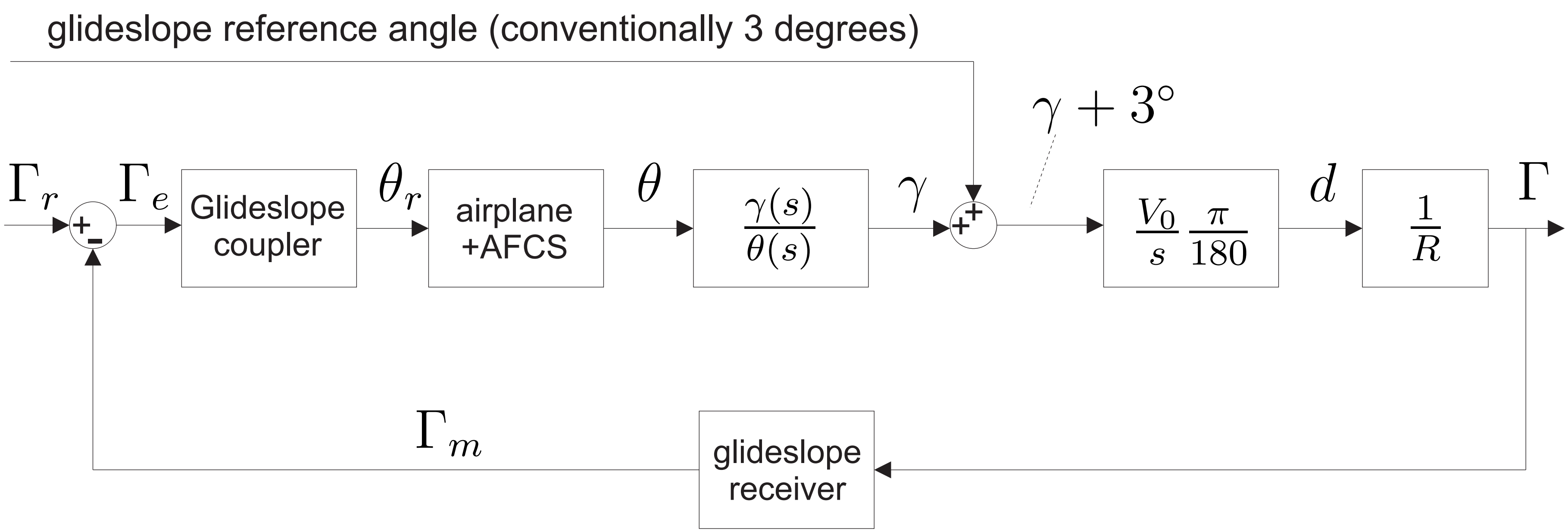
$$d\left(s\right) \approx \frac{V_0}{s}\frac{\pi}{180}\left(\gamma + 3^\circ\right)$$

Glideslope error angle:

$$\Gamma = \mathrm{atan}\left(\frac{d}{R}\right) \approx \frac{d}{R}$$

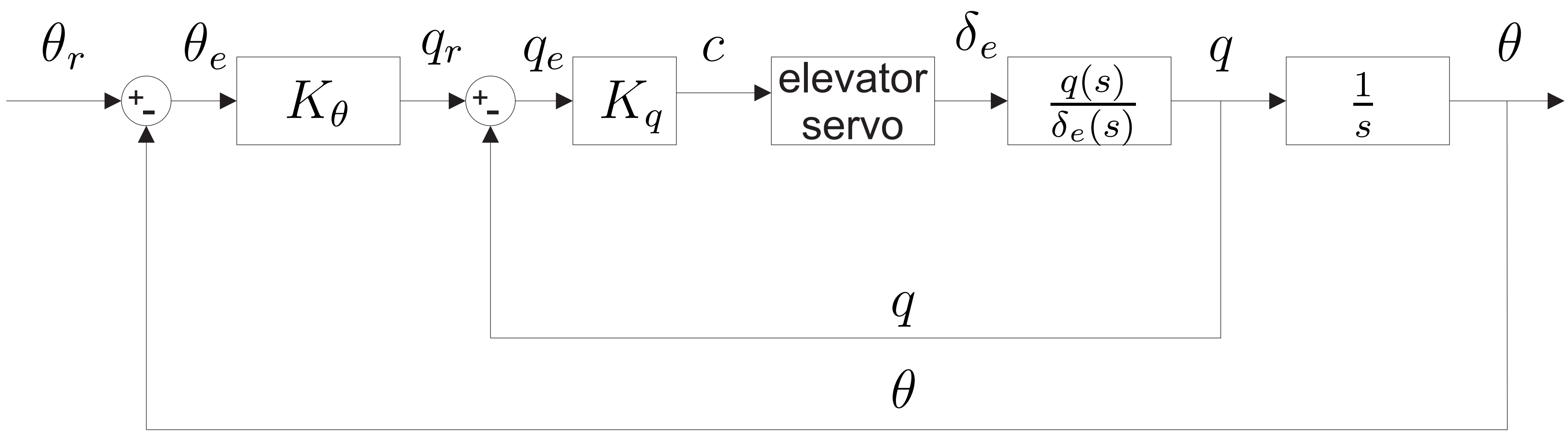
Glide slope

Block diagram of a glide slope hold loop:



Glide slope

Contents of the "Airplane+AFCS" block from the previous diagram:



Glide slope

Block diagrams:

- Glideslope coupler: couples error signal to autopilot.
- $\frac{\gamma(s)}{\theta(s)}$: conversion from pitch attitude to flight path angle.
- $\frac{V_0}{s}\frac{\pi}{180}$: conversion from flight path to deviation.
- $\frac{1}{R}$: conversion from deviation to glideslope error.
- Glideslope coupler: $H_{coupler} = K_c\left(1 + \frac{W_1}{s}\right)$, with K_c the coupler gain for acceptable closed loop behaviour, and W_1 the weighting constant to cope with turbulance. Usually $W_1 = 0.1$.

Glide slope

- Conversion from pitch attitude to flight path angle:

$$\frac{\gamma(s)}{\theta(s)} = \frac{\theta(s) - \alpha(s)}{\theta(s)} = 1 - \frac{\alpha(s)}{\theta(s)} = 1 - \frac{\alpha(s)/\delta_e(s)}{\theta(s)/\delta_e(s)} = 1 - \frac{N_\alpha(s)}{N_\theta(s)}$$

- Glideslope receiver:

$$H_{receiver} \approx 1$$

- What about the reference input signal Γ_{ref}?

Glide slope

- Conversion from pitch attitude to flight path angle:

$$\frac{\gamma(s)}{\theta(s)} = \frac{\theta(s) - \alpha(s)}{\theta(s)} = 1 - \frac{\alpha(s)}{\theta(s)} = 1 - \frac{\alpha(s)/\delta_e(s)}{\theta(s)/\delta_e(s)} = 1 - \frac{N_\alpha(s)}{N_\theta(s)}$$

- Glideslope receiver:

$$H_{receiver} \approx 1$$

- What about the reference input signal Γ_{ref}?
  This is the reference glideslope error and this is zero in all circumstances.

Glide slope

Remark:

- The only degree of freedom to control the closed loop system is the coupler gain K_c.
- The glideslope coupler replaces the 'controller' which can usually be found at this position in the loop and which contains the three types of action:
  - Proportional
  - Derivative
  - Integral

Automatic flare mode

Allowable vertical touchdown velocity on runway is determined by several factors:

- Passenger and crew comfort.
- Landing gear and attachment structure load limits.

Passenger comfort:

- Hard landings, $\dot{h} \geq 6ft/s$: Not acceptable everyday.
- Firm landings, $2ft/s \geq \dot{h} \geq 3ft/s$: Desirable.
- Egg-landings, $\dot{h} \approx 0ft/s$: Undesirable ("floatation" and lack of control over touchdown point).

Automatic flare mode

Landing gear and attachment structure load limits:

- Civil transports: $\dot{h} \leq 10 ft/s$.
- Carrier based airplanes: $\dot{h} \leq 25 ft/s$.

Relationship between forward airspeed, glideslope angle, and vertical touchdown speed?

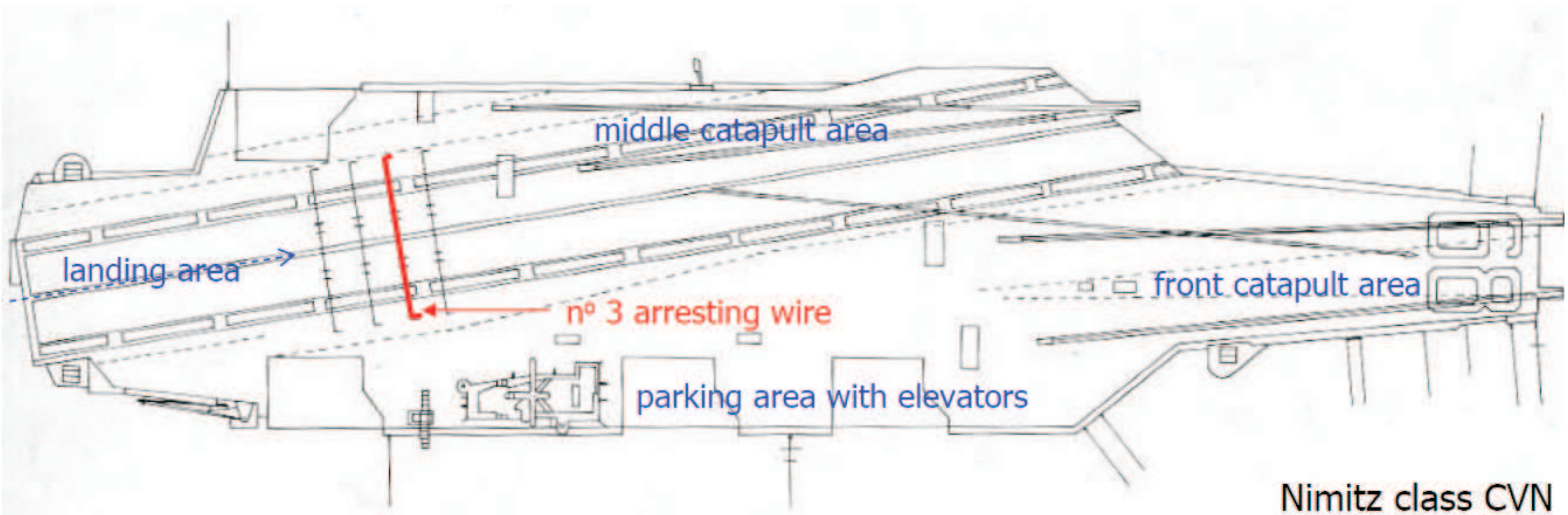
Automatic flare mode

Relationship between forward airspeed, glideslope angle, and vertical touchdown speed:

Forward velocity		Vertical touchdown rate on a 3° degree glideslope
knots	ft/s	ft/s
10	17	0.9
20	34	1.8 "Soft landings"
40	68	3.6
80	135	7.1
120	203	10.6 "Hard landings"
160	270	14.2 Flare manoeuvre required

Automatic flare mode

Carrier based airplanes: no flare,
driven straight into the deck



Automatic flare mode

Which one is the carrier based aircraft?

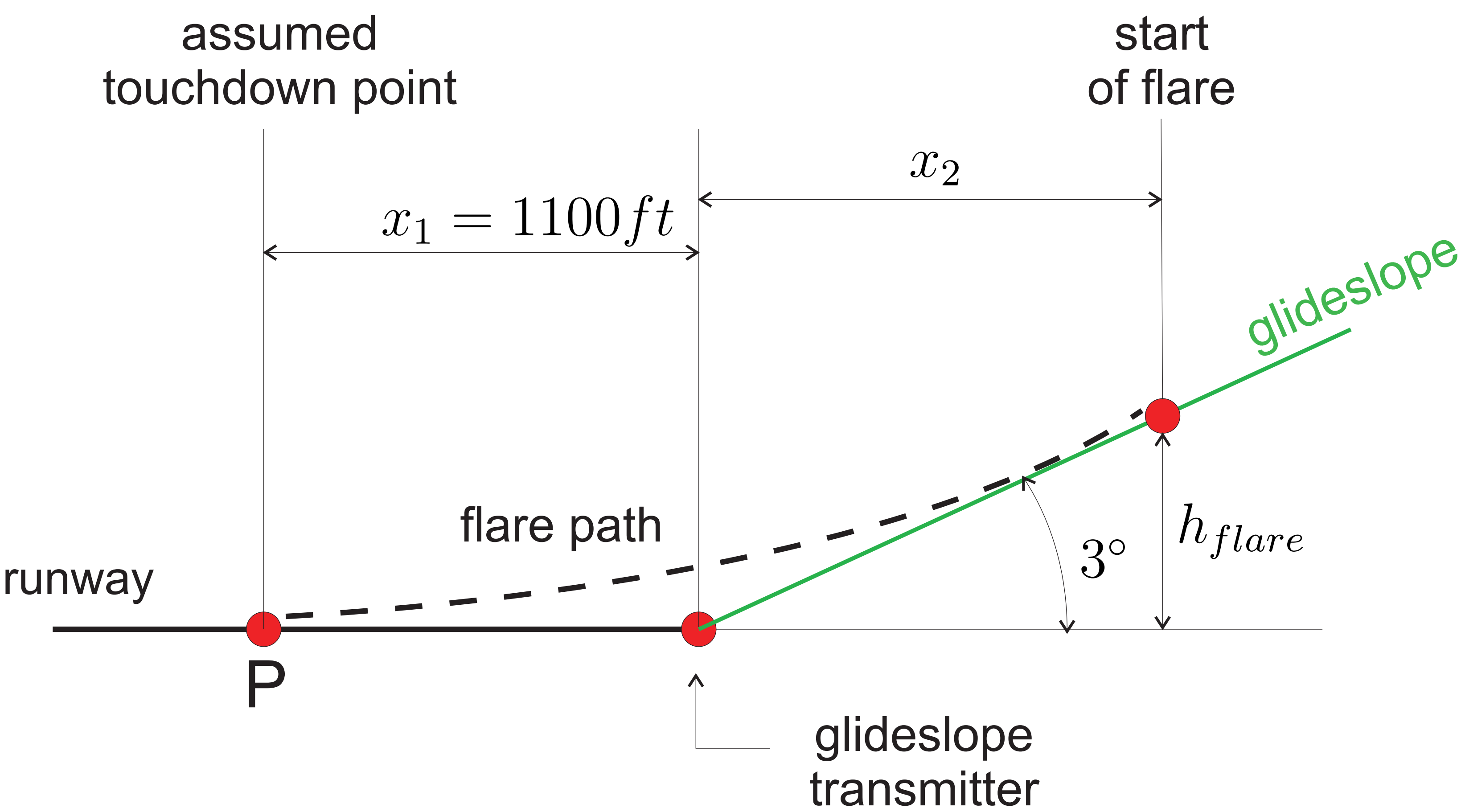
Automatic flare mode

Which one is the carrier based aircraft?

Left: F-18 carrier based. Right: F-15 land based.

Automatic flare mode

Geometry of the flare path during automatic landing:



Automatic flare mode

Assumptions:

- Airplane kept on flare path by pitch attitude command.
- Flare path starts at height of h_{flare}.
- Intended touchdown point is $1100 ft$ from the glideslope transmitter.

For the flare control law we need an approximation of the flare path.

- Approximation of the flare path:

$$h = h_{flare} e^{-\frac{t}{\tau}}$$

Automatic flare mode

Example B747-200

- Configuration: Landing.
- Airspeed: $V_0 = 221$ ft/s $= 242.5$ km/h.
- Descent rate at the start of the manoeuvre:

$$\dot{h}_{h_{flare}} = V_0 \sin\left(-\frac{3}{180}\pi\right) = -11.57\mathrm{ft/s}$$

- Differentiation of flare path:

$$h = h_{flare}e^{-\frac{t}{\tau}} \rightarrow \dot{h} = -\frac{h_{flare}}{\tau}e^{-\frac{t}{\tau}}$$

$$\dot{h}(0) = -\frac{h_{flare}}{\tau} = -11.57\mathrm{ft/s}$$

Automatic flare mode

Example B747-200

Assume: airplane touchdown in $t = 4\tau$

$$x_1 + x_2 = 1100 + x_2 = 4\tau V_0 = 884\tau \rightarrow x_2 = 884\tau - 1100$$

However, from flare geometry:

$$x_2 = \frac{h_{flare}}{\tan\left(3\frac{\pi}{180}\right)}$$

Using: $-\frac{h_{flare}}{\tau} = -11.57\text{ft/s}$

$$x_2 = \frac{11.57\tau}{\tan\left(3\frac{\pi}{180}\right)} \approx 221\tau$$

Automatic flare mode

Example B747-200

Resulting system:

$$884\tau - 1100 = 221\tau \rightarrow \tau = 1.66s$$

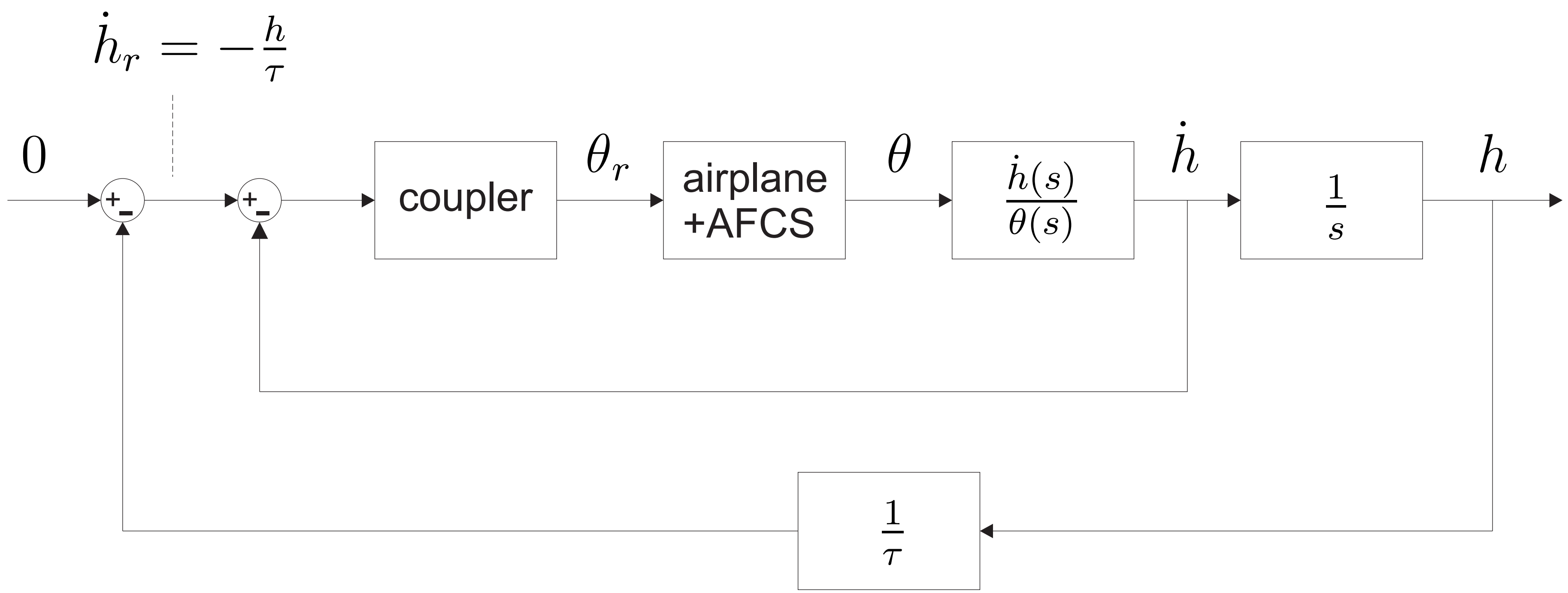
Resulting flare altitude:

$$h_{flare} = -\dot{h}(t=0)\tau = 19.2ft = 5.85m$$

Resulting flare control law:

$$\dot{h} = -\frac{h}{\tau} = -0.6h$$

Automatic flare mode



Automatic flare mode

Coupler representation: Try same representation as glideslope coupler.

$H_{coupler} = K_c \left(1 + \frac{W_1}{s}\right)$

with K_c the coupler gain for acceptable closed loop behaviour, and W_1 the weighting constant to cope with turbulence. Usually $W_1 = 0.1$.

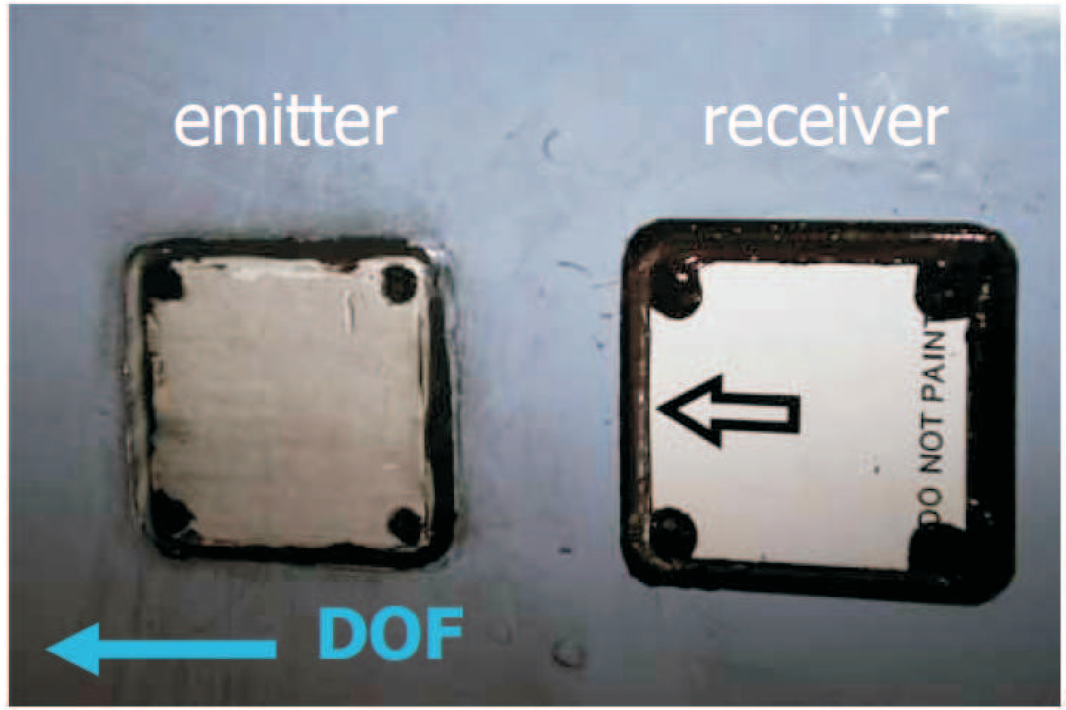
Pitch angle to vertical speed transfer function:

$$\dot{h} \approx V_0 \gamma \rightarrow \frac{\dot{h}(s)}{\theta(s)} = V_0 \frac{\gamma(s)}{\theta(s)}$$

Automatic flare mode

Altitude measurement during approach and flare is critical:

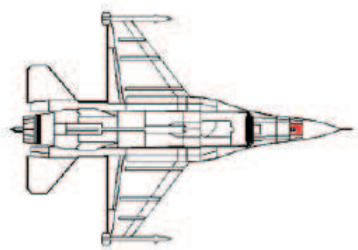
Radar altimeter
of Bae Rj85:



Automatic flare mode

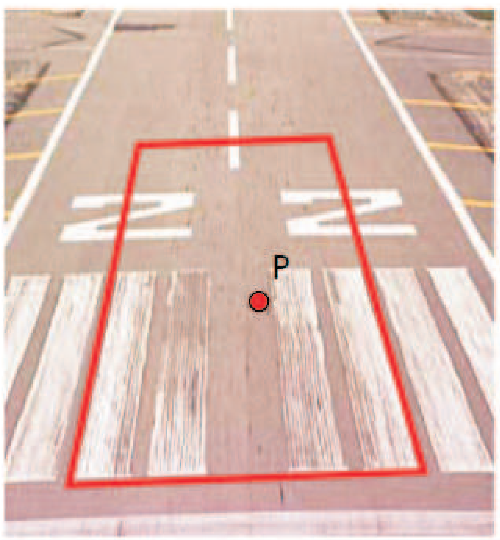
Altitude measurement during approach and flare is critical:

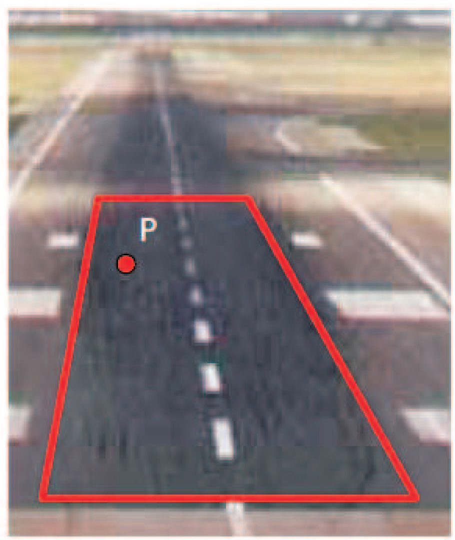
Radar altimeter
of F-16 and
Cheetah

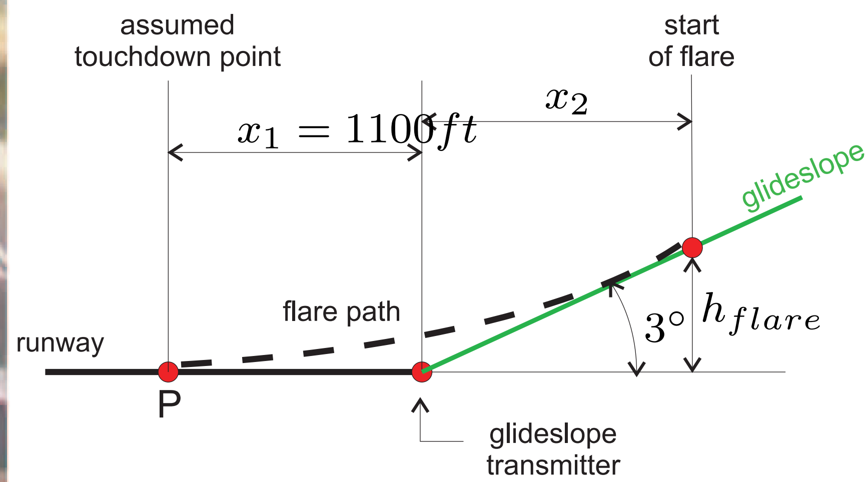


Automatic flare mode

Principle of Monte Carlo scheme: Variation of the actual touchdown point P inside a box of acceptable touchdown points by means of a normalized random number sampling method.







Lateral

Introduction

Lateral navigational functions:

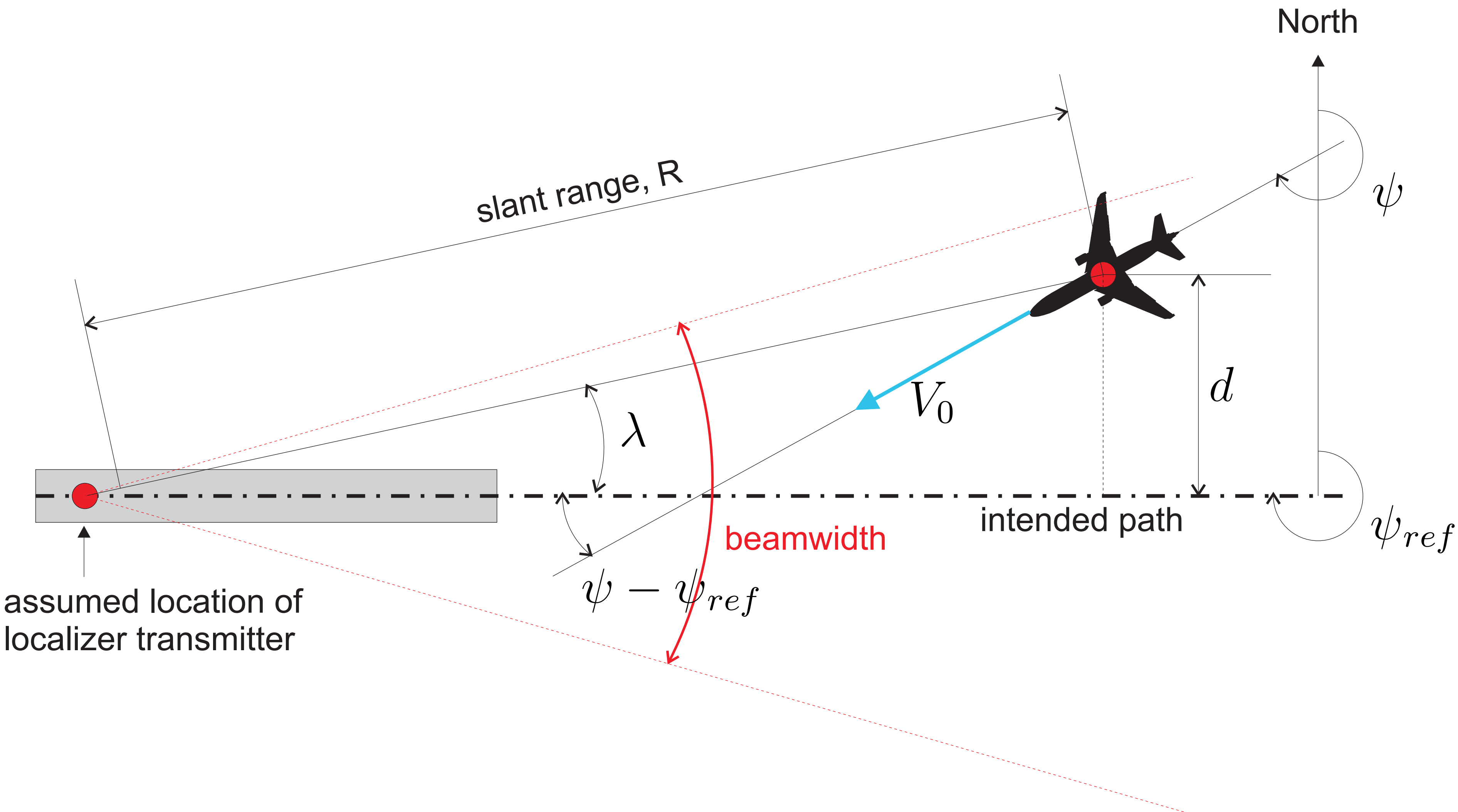
- Localizer hold mode.
- VOR hold mode.

Localizer hold mode

Localizer hold mode:

- It is more accurate to follow the localizer automatically than the pilot performing this task manually.
- Localizer intercept and hold mode is part of an automatic ILS approach.
- Assumption: Heading angle control system already present (**Lecture 13**).
- Localizer geometry?

Localizer hold mode



Localizer hold mode

Assumptions:

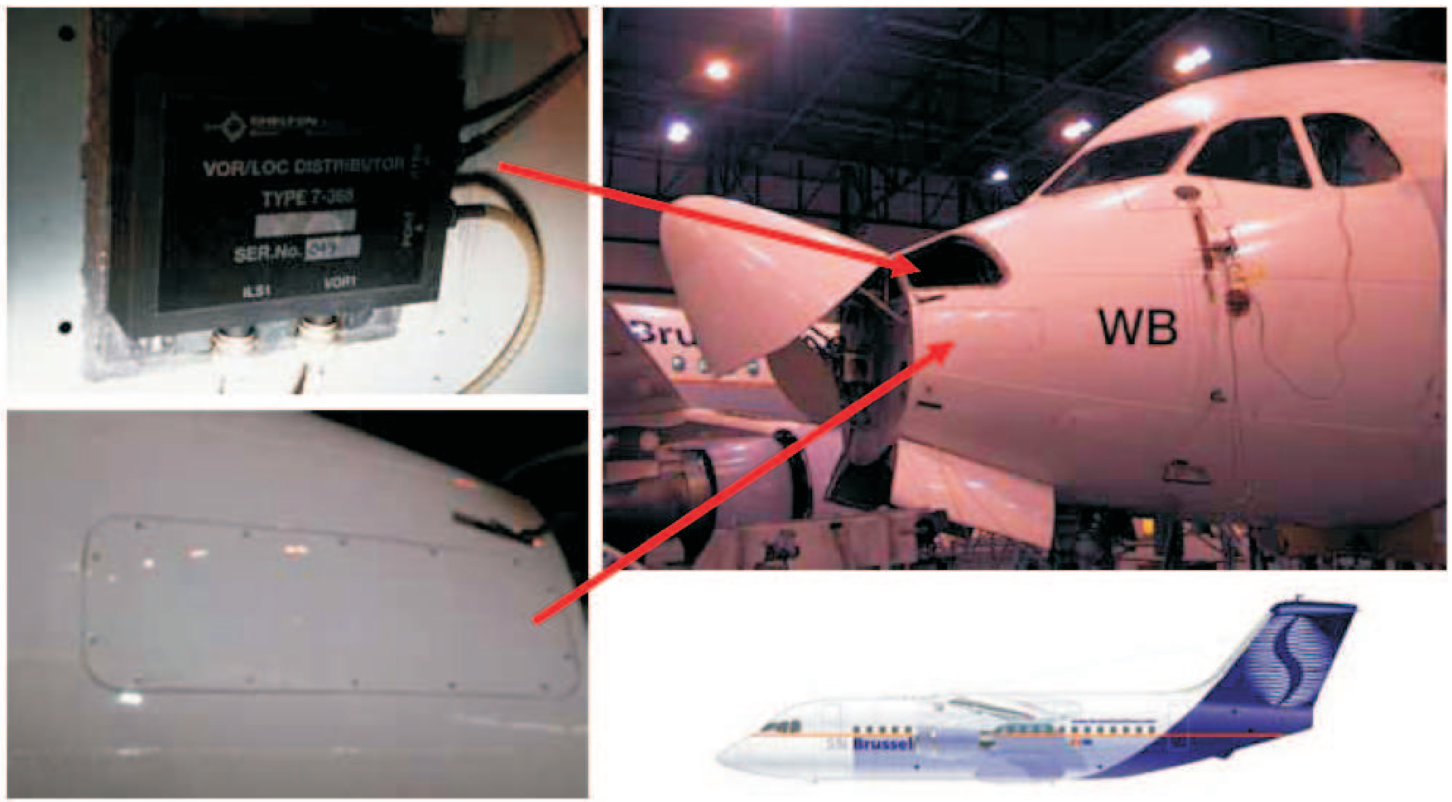
- Airplane localizer antenna coincides with cg.
- cg driven along localizer beam centerline: intended path.
- Localizer error angle λ is sensed by localizer receiver.
- Airplane is kept on centerline by heading angle command.
- Localizer beam width is $5°$.
- Any speed/lift changes due to rolling are automatically compensated.
- Localizer guidance and control is independent of longitudinal control.

Localizer hold mode

Picture of ILS-localizer antenna at an airport.

Localizer hold mode

Picture of localizer receiver and antenna on the BAe RJ85:

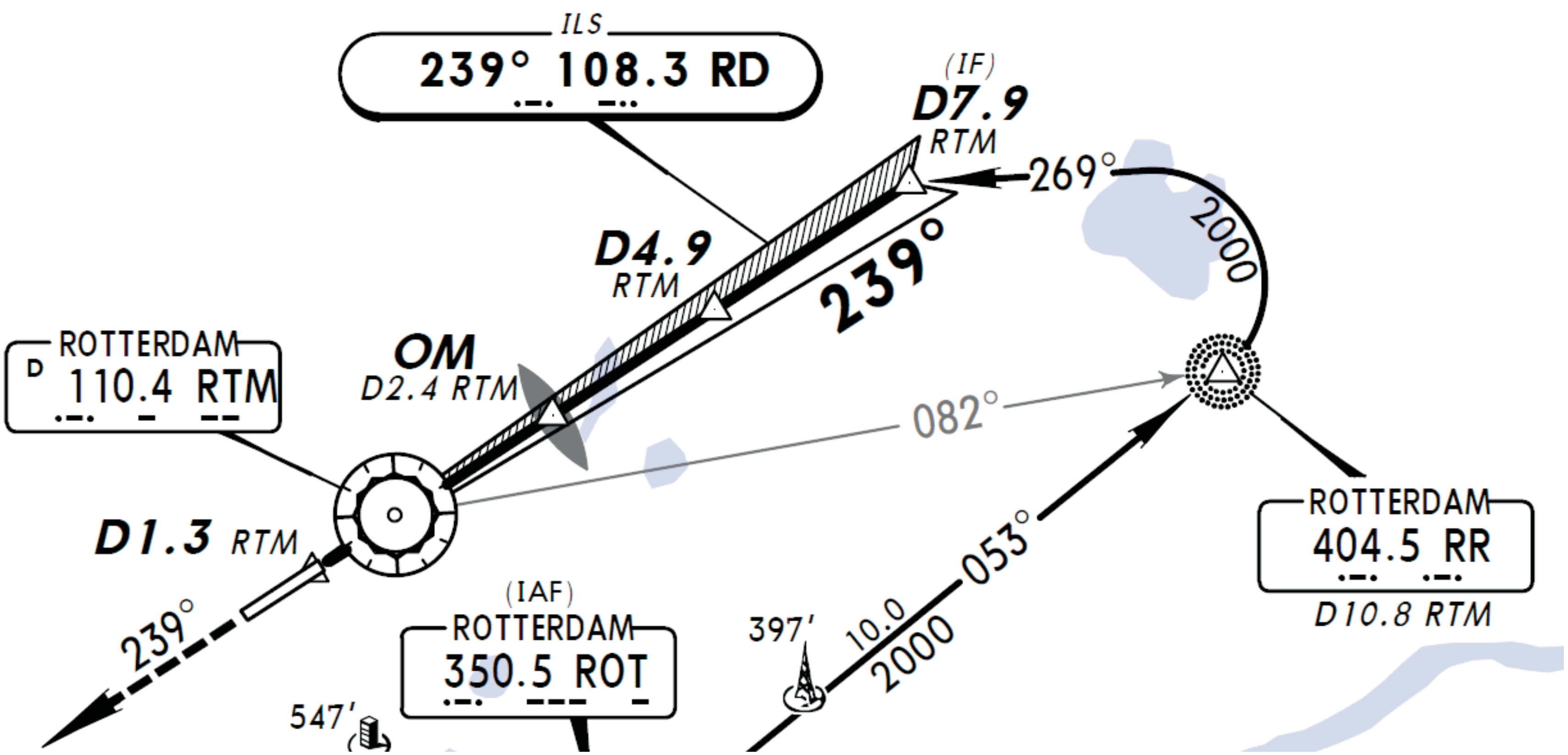


Localizer hold mode

Picture of localizer antenna on the Alpha Jet:

Localizer hold mode

Indication of ILS-localizer on aviation charts:



Localizer hold mode

Quantity to be controlled: localizer error angle λ:

- Localizer error angle λ:

$$\lambda = \text{asin}\left(\frac{d}{R}\right) \approx \frac{d}{R}$$

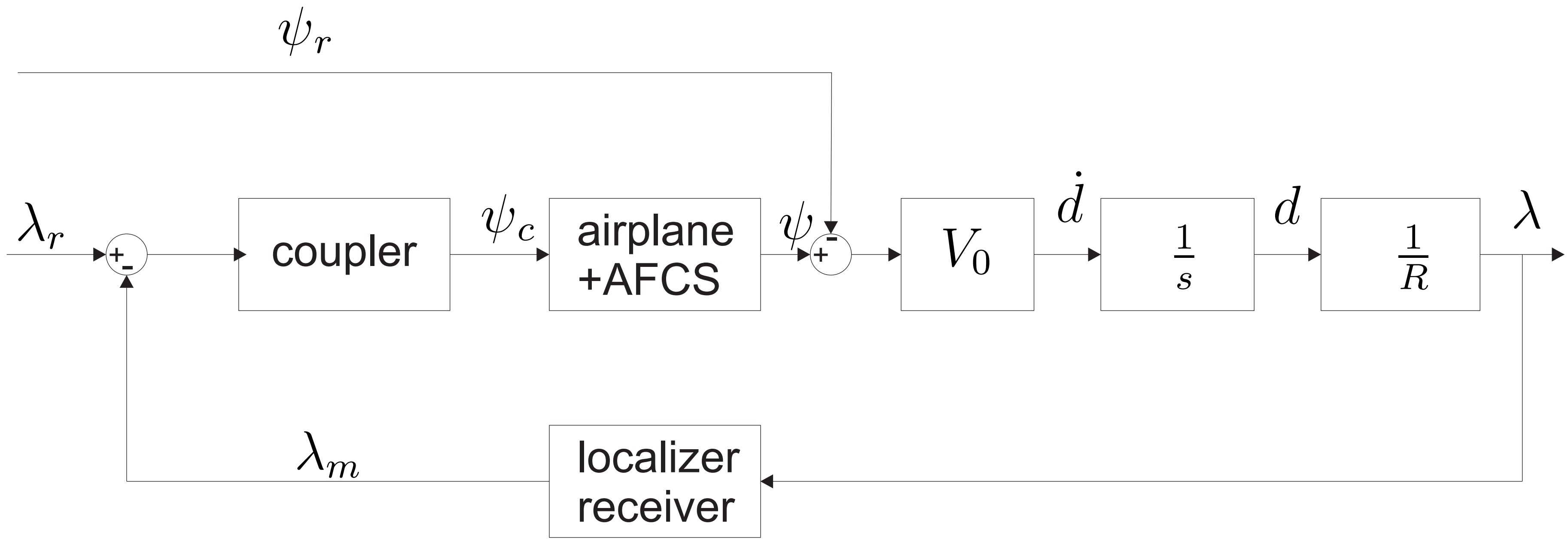
- Deviation distance:

$$d(s) = \frac{1}{s}\dot{d}(s)$$

$$\dot{d}(s) = V_0 \sin\left(\psi(s) - \psi_{ref}(s)\right) \approx V_0\left(\psi(s) - \psi_{ref}(s)\right)$$

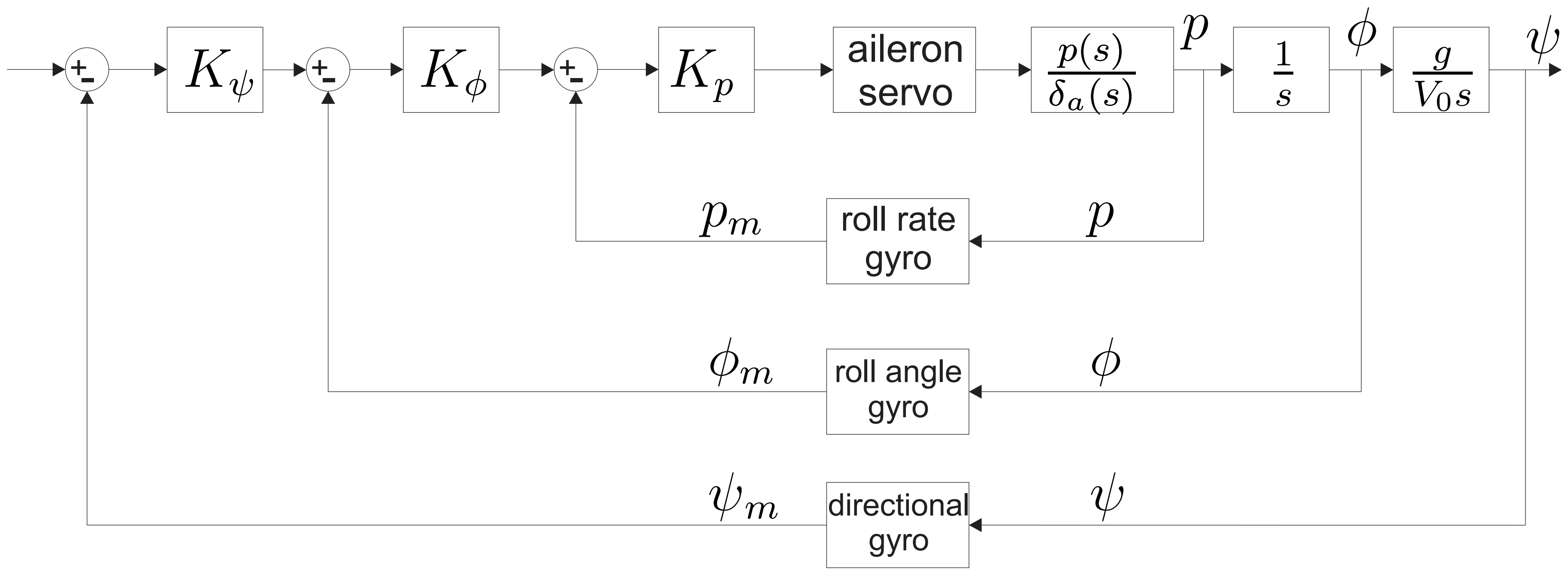
Localizer hold mode

Block diagram of a localizer hold loop:



Localizer hold mode

Inner loops:



Localizer hold mode

The coupler block couples the localizer error signal to the previously designed autopilot.

- $$H_{coupler} = K_c \left(1 + \frac{W_1}{s}\right)$$
- where K_c is the couple gain for acceptable closed loop behaviour and W_1 is a weighting constant to cope with turbulence, usually $W_1 = 0.1$.

Localizer hold mode

- Localizer receiver is modelled as a perfect sensor: $H_{receiver} \approx 1$.
- What about the localizer reference input signal λ_r?

Localizer hold mode

- Localizer receiver is modelled as a perfect sensor: $H_{receiver} \approx 1$.
- What about the localizer reference input signal λ_r?
- The localizer reference input signal λ_r is always 0.
- The only degree of freedom in the closed loop system is the coupler gain K_c, which replaces the controller that is usually found at this position in the loop (PID-controller).

Localizer hold mode

Boeing 747-200 example.

Dimensions:
-wing area: 511 m^2
-mean chord: 8.3 m
-wingspan: 59.7m
Flight condition: approach
-altitude: 0 m
-speed: 242.5 km/h
-weight: 2.888MN

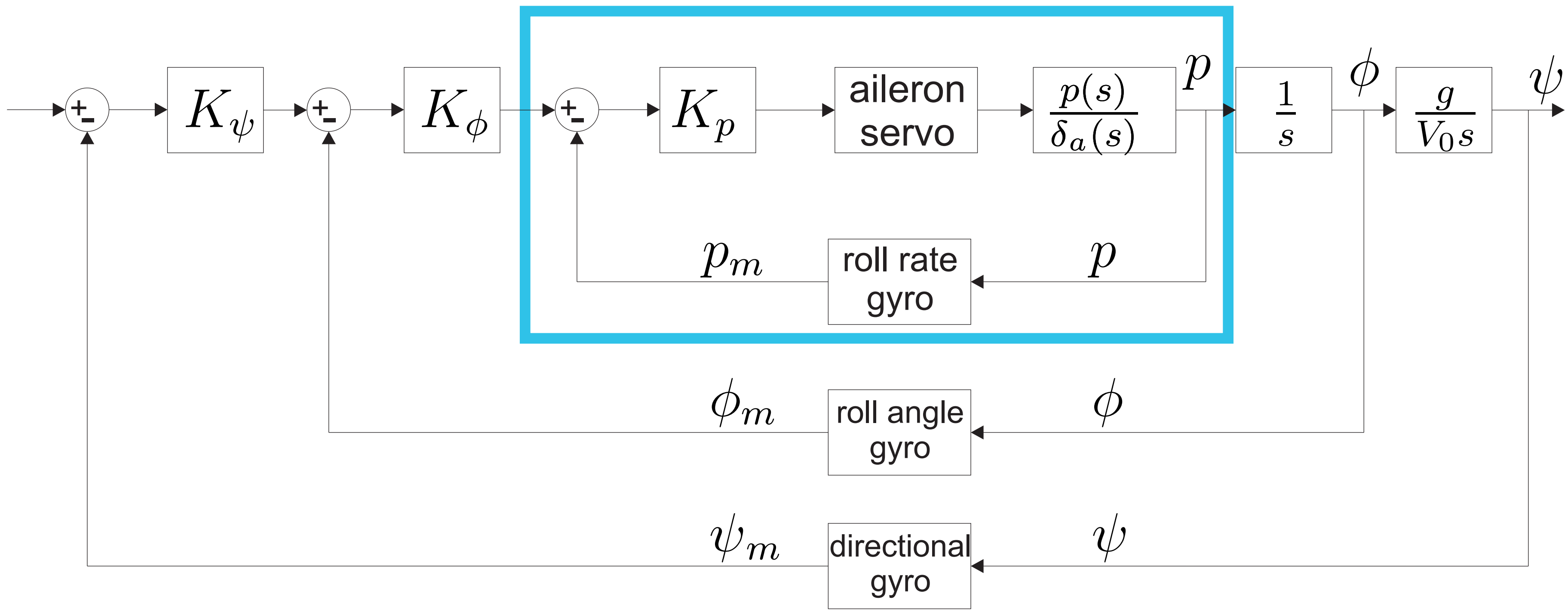
Localizer hold mode

Boeing 747-200 example.

- Start with design of inner loops:
  - Roll rate control: K_p.
  - Roll angle control: K_ϕ.
  - Heading angle control: K_ψ.
- Design outer loop:
  - Coupler gain for various slant ranges: K_c.

Localizer hold mode

Boeing 747-200 example: Roll rate loop



Localizer hold mode

Boeing 747-200 example: Roll rate control

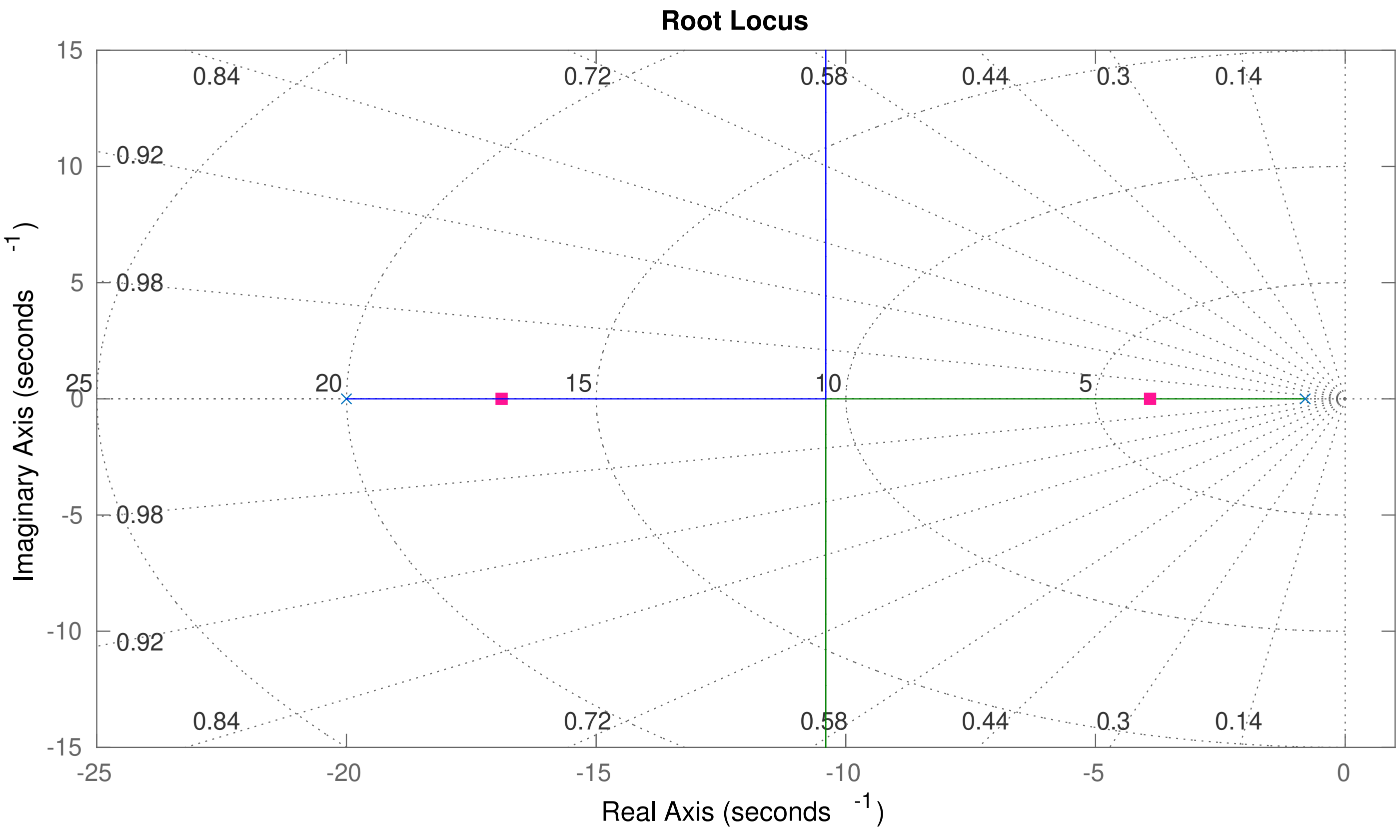
- Start with reduced transfer function from aileron to roll rate:

$$\frac{p(s)}{\delta_a(s)} = \frac{L_{\delta_a}}{s - L_p}$$

- Aileron servo dynamics as before $H_{servo} = \frac{20}{s+20}$.
- Matlab/Sisotool demo.
- A feedback gain $K_p = -10$ is selected.
- Examine step response.

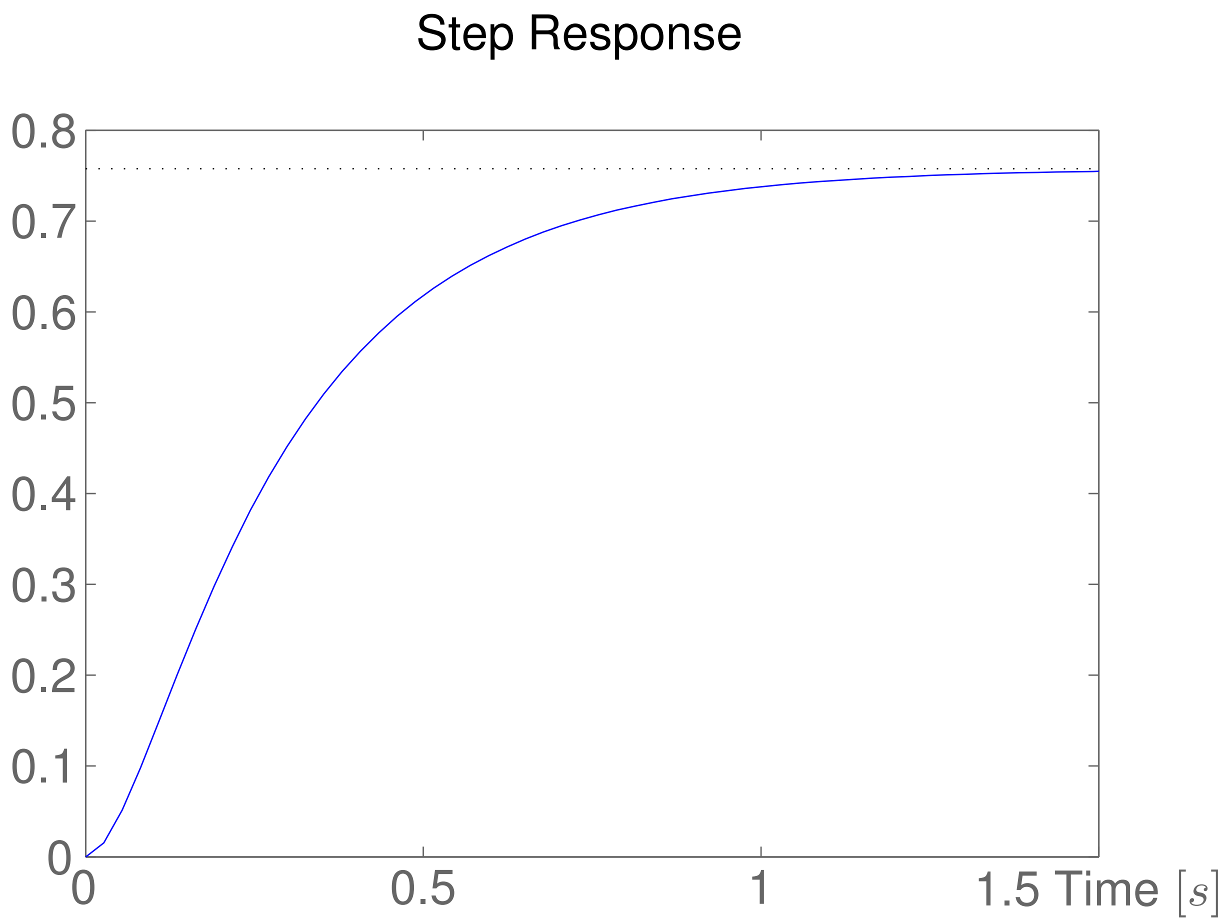
Localizer hold mode

Boeing 747-200 example: Roll rate Root Locus



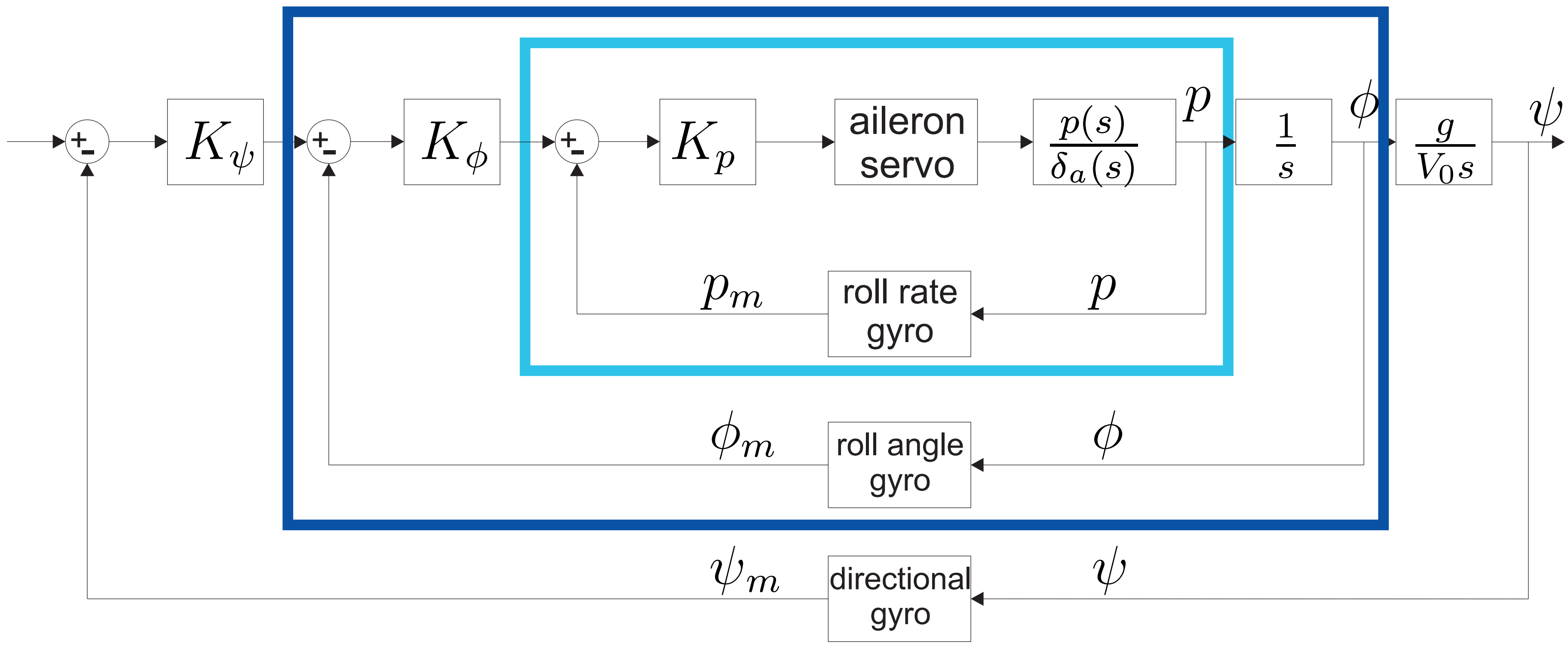
Localizer hold mode

Boeing 747-200 example: Roll rate step



Localizer hold mode

Boeing 747-200 example: Roll angle loop



Localizer hold mode

Boeing 747-200 example: Roll angle control

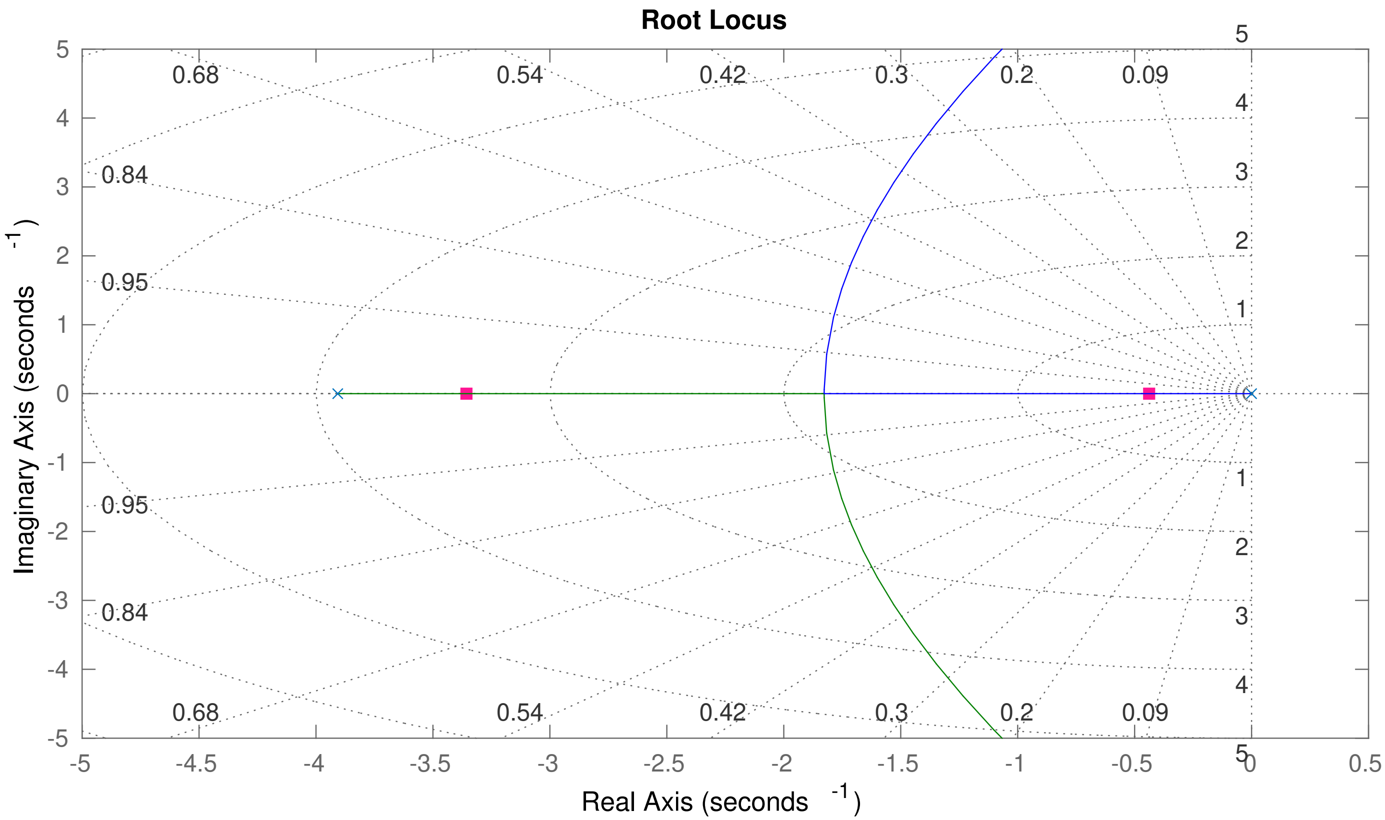
- Start with closed loop from previous step:

$$\frac{p(s)}{p_c(s)} = \frac{K_p H_{servo} \frac{p(s)}{\delta_a(s)}}{1 + K_p H_{servo} \frac{p(s)}{\delta_a(s)}}$$

- Integrate output p to get ϕ.
- Matlab/Sisotool demo.
- A feedback gain $K_\phi = 0.5$ is selected.
- Examine step response.

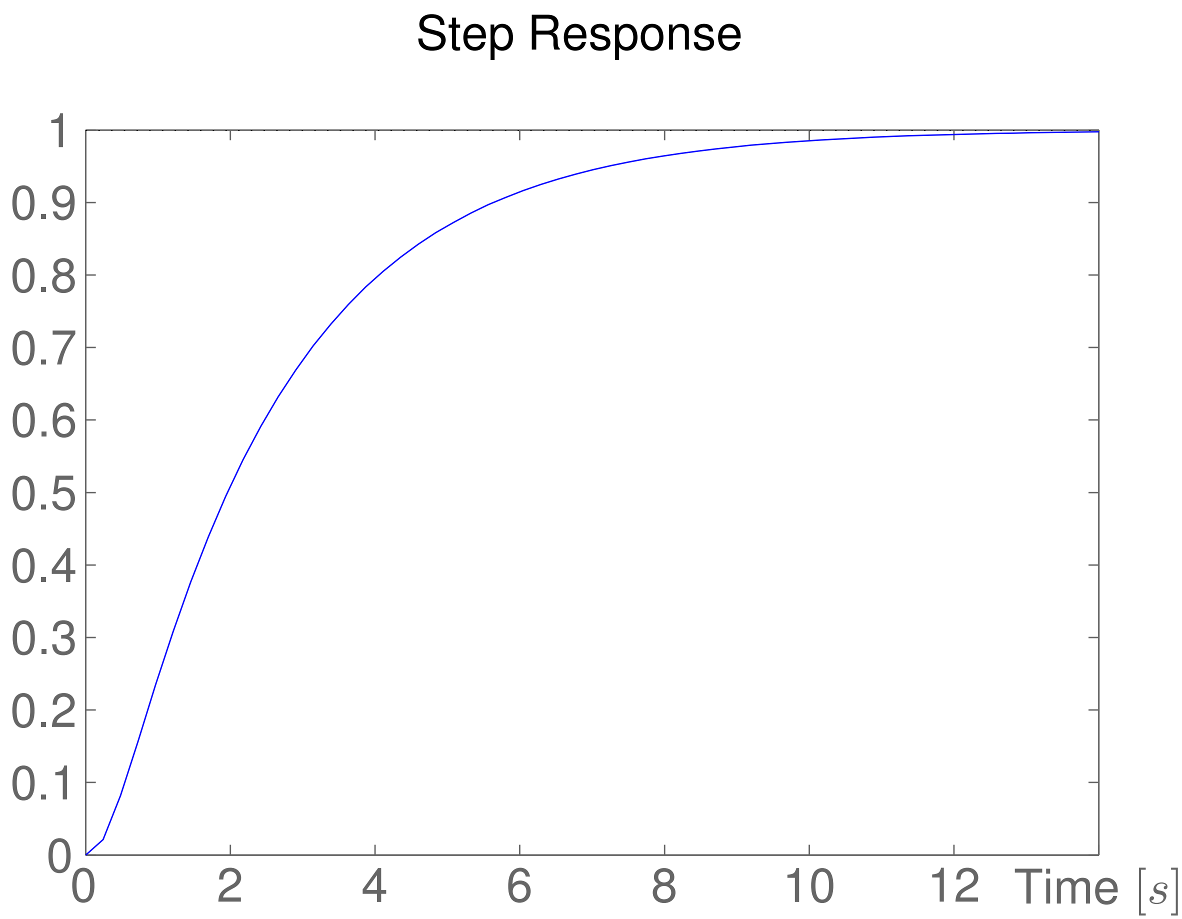
Localizer hold mode

Boeing 747-200 example: Roll angle Root Locus



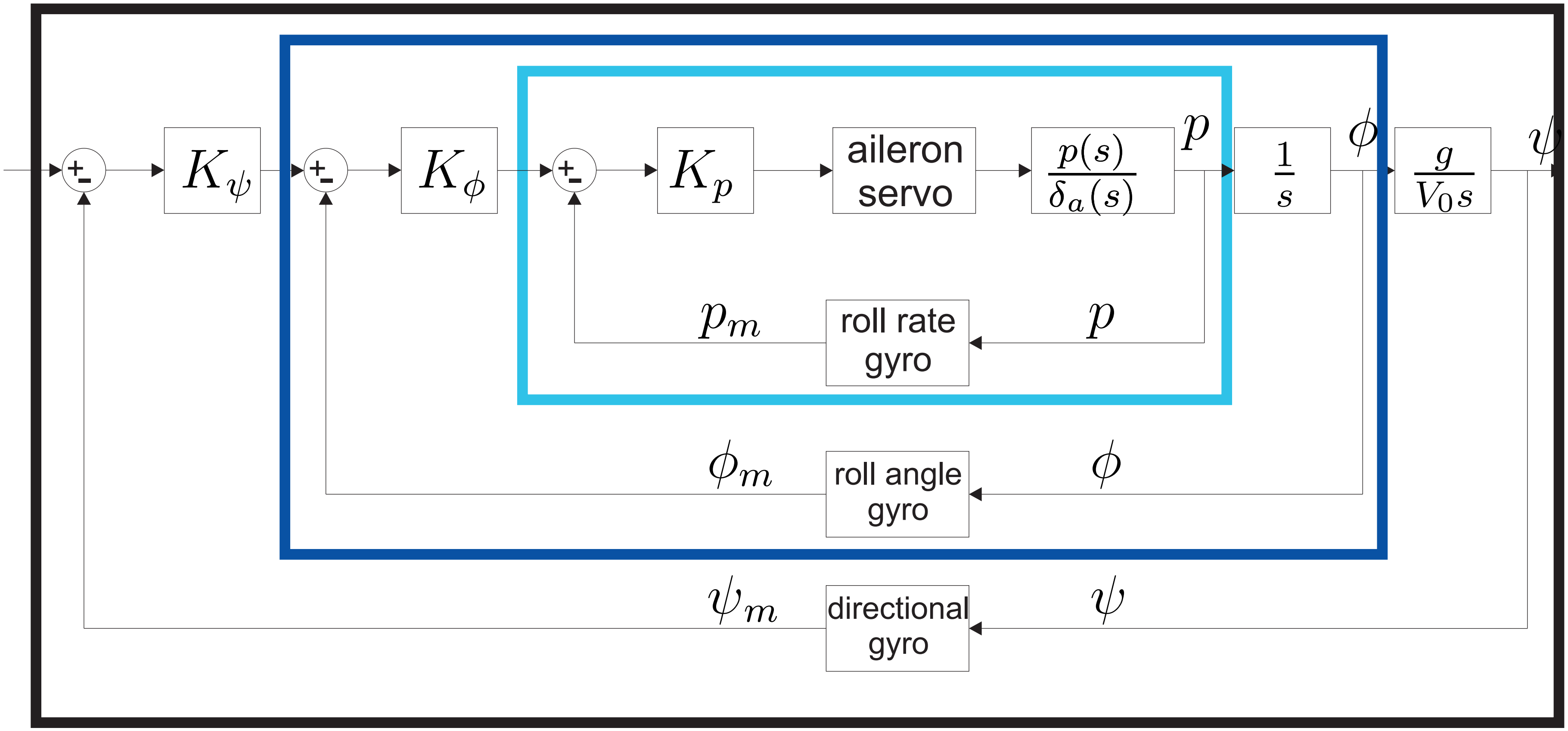
Localizer hold mode

Boeing 747-200 example: Roll angle step



Localizer hold mode

Boeing 747-200 example: Heading angle loop



Localizer hold mode

Boeing 747-200 example: Heading angle control

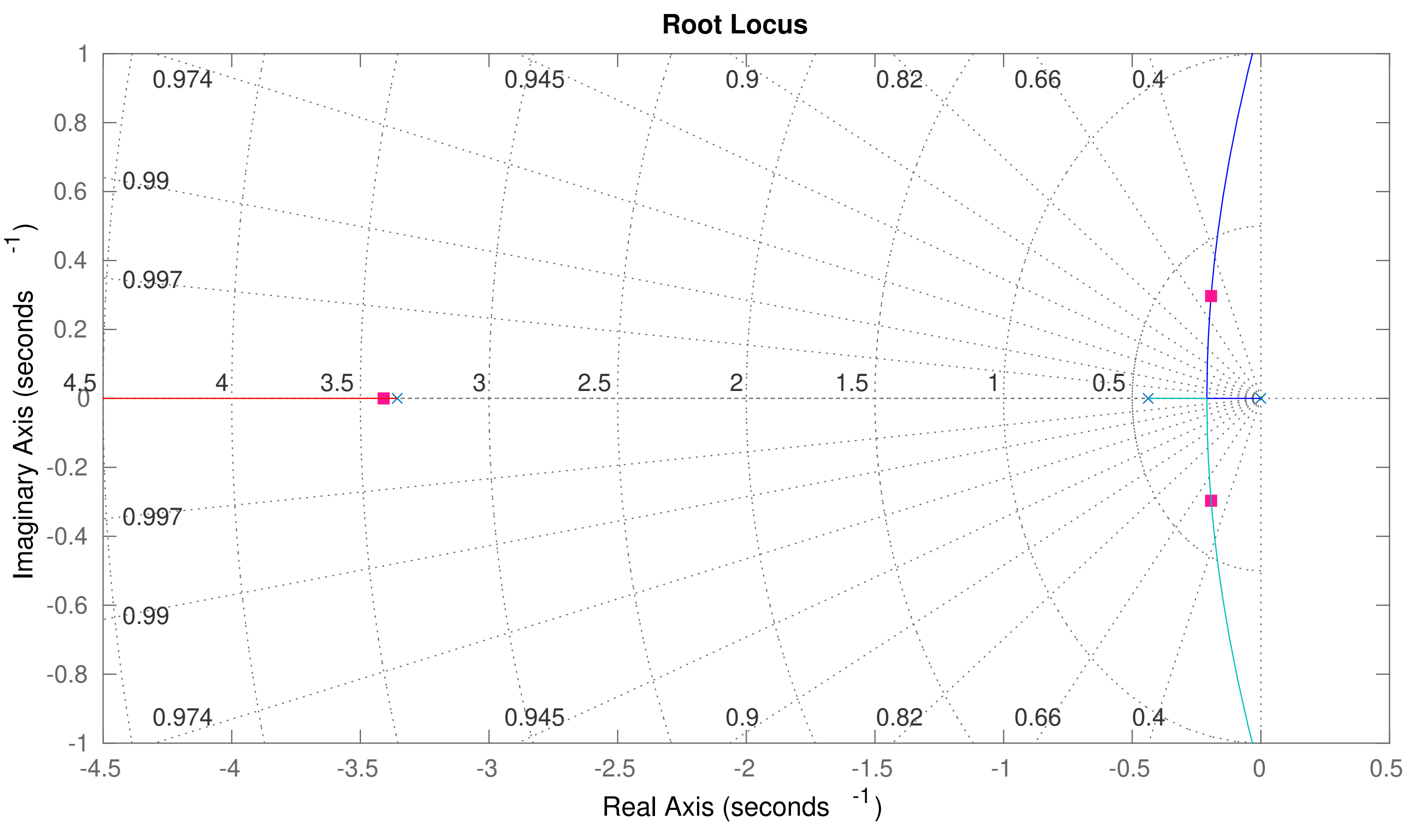
- Start with closed loop from previous step:

$$\frac{\phi\left(s\right)}{\phi_c\left(s\right)} = \frac{K_\phi \frac{p(s)}{p_c(s)} \frac{1}{s}}{1 + K_\phi \frac{p(s)}{p_c(s)} \frac{1}{s}}$$

- Multiply output ϕ with $\frac{g}{V_0 s}$ to get ψ (**Lecture 13**).
- Matlab/Sisotool demo.
- A feedback gain $K_\psi = 2$ is selected.
- Examine step response.

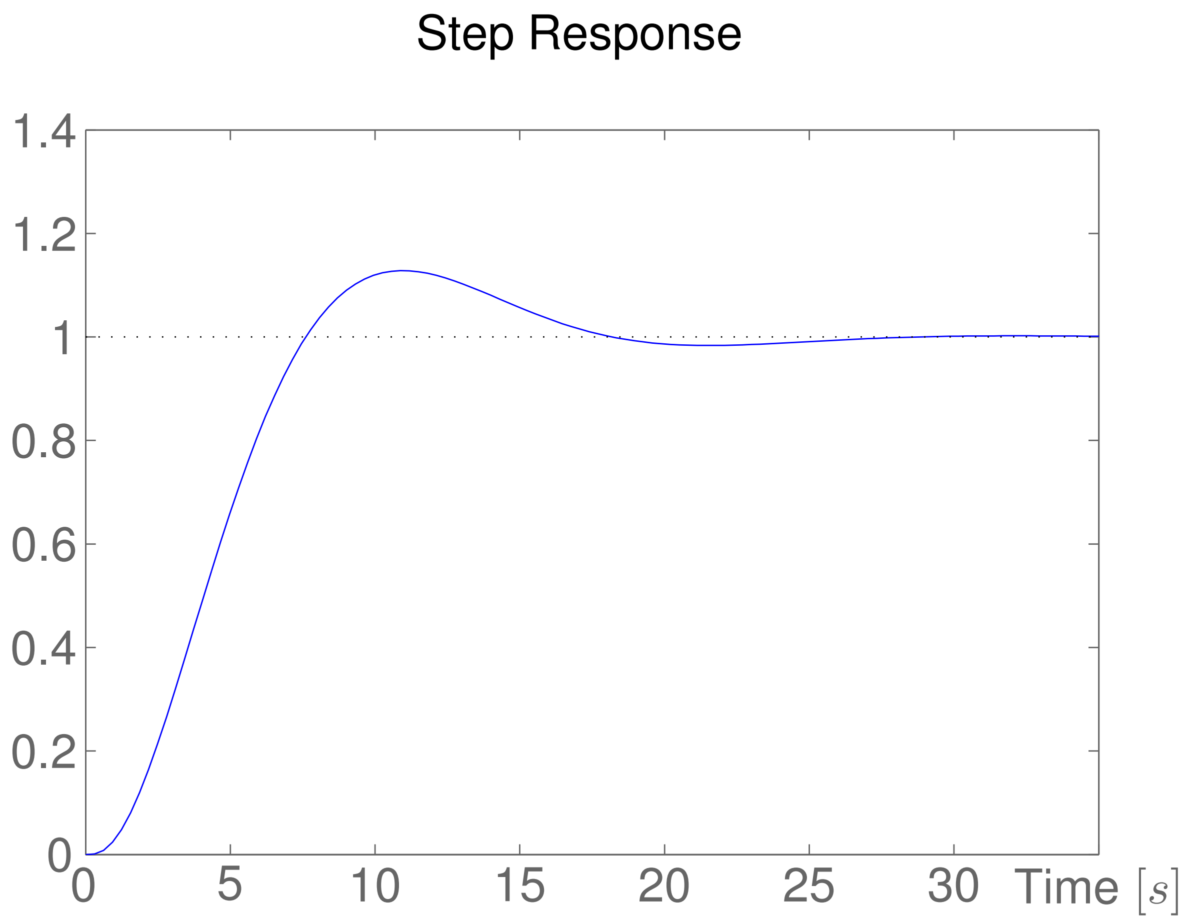
Localizer hold mode

Boeing 747-200 example: Heading angle Root Locus



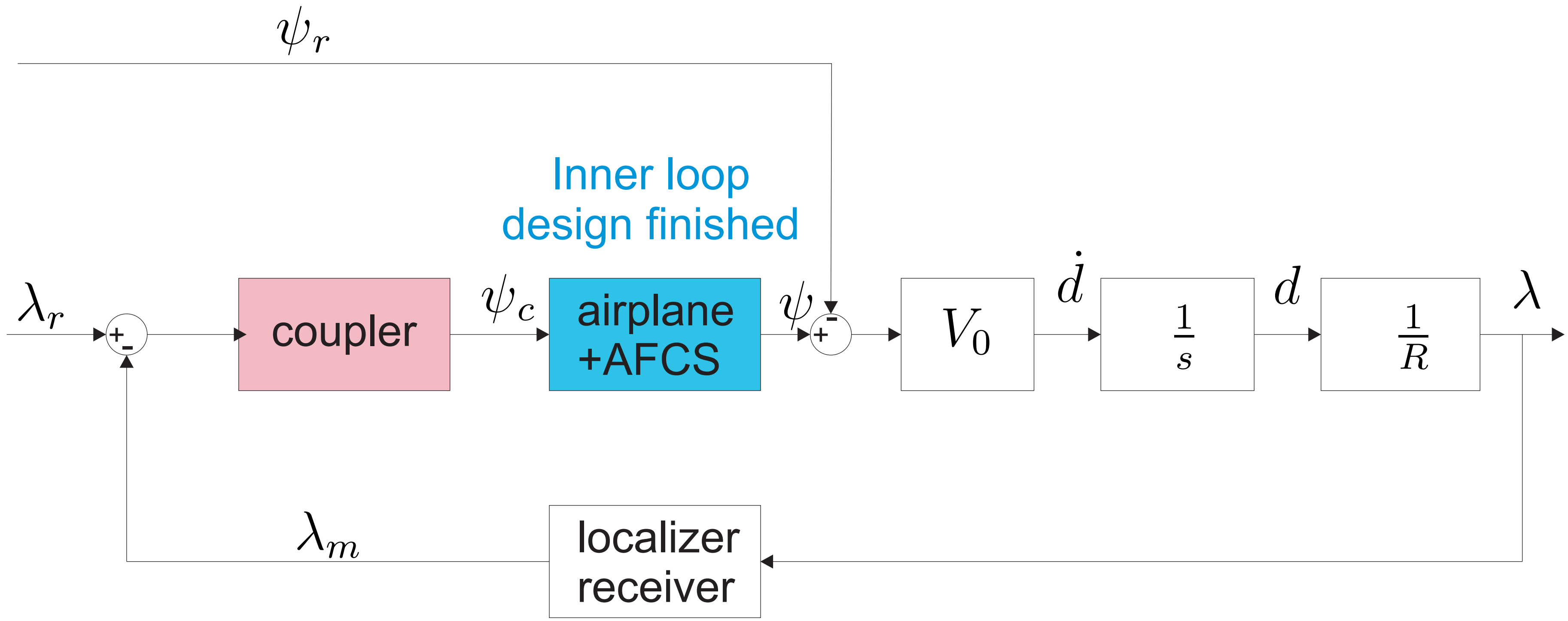
Localizer hold mode

Boeing 747-200 example: Heading angle step



Localizer hold mode

Block diagram of a localizer hold loop:



Localizer hold mode

Boeing 747-200 example: Localizer control

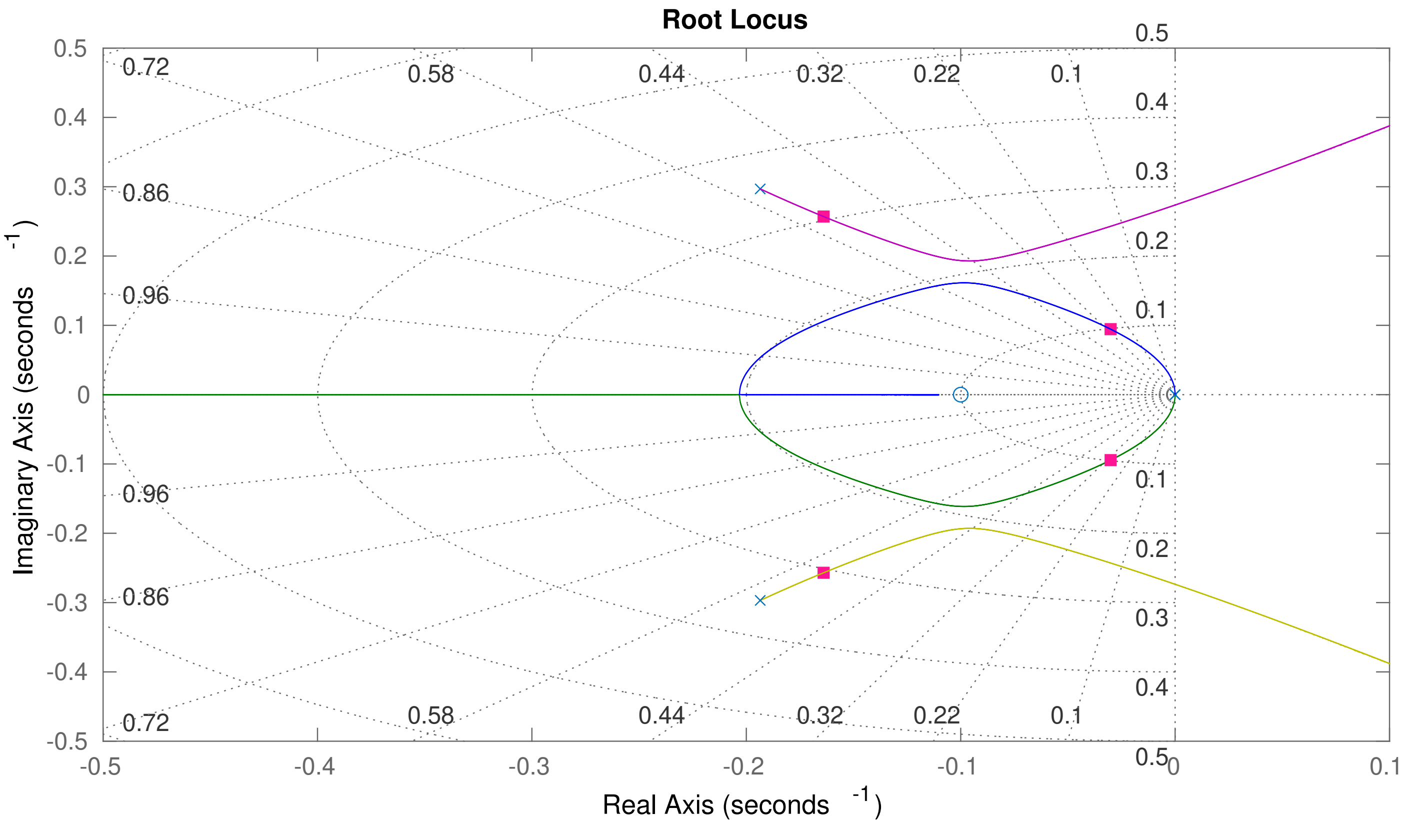
- Simplification: runway 36, so $\psi_{ref} = 0$
- Start with closed loop from previous step:

$$\frac{\psi(s)}{\psi_c(s)} = \frac{K_\psi \frac{\phi(s)}{\phi_c(s)} \frac{g}{V_0 s}}{1 + K_\psi \frac{\phi(s)}{\phi_c(s)} \frac{g}{V_0 s}}$$

- Multiply output ψ with $\frac{V_0}{Rs}$ to get λ ($R = 5nm$).
- Matlab/Sisotool demo. A coupler gain $K_c = 10$ is selected.
- Examine step response.

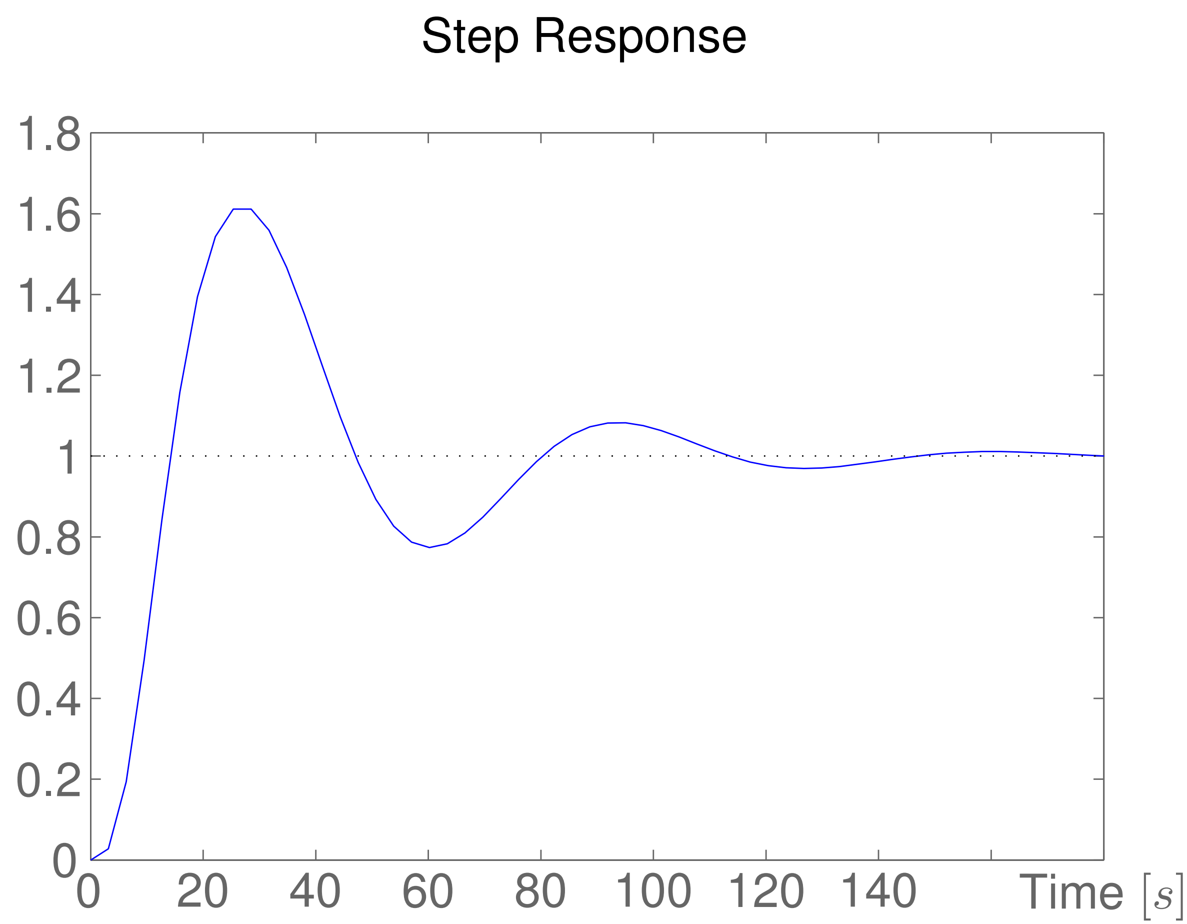
Localizer hold mode

Boeing 747-200 example: Localizer Root Locus



Localizer hold mode

Boeing 747-200 example: Localizer step for slant range of 5 nm.

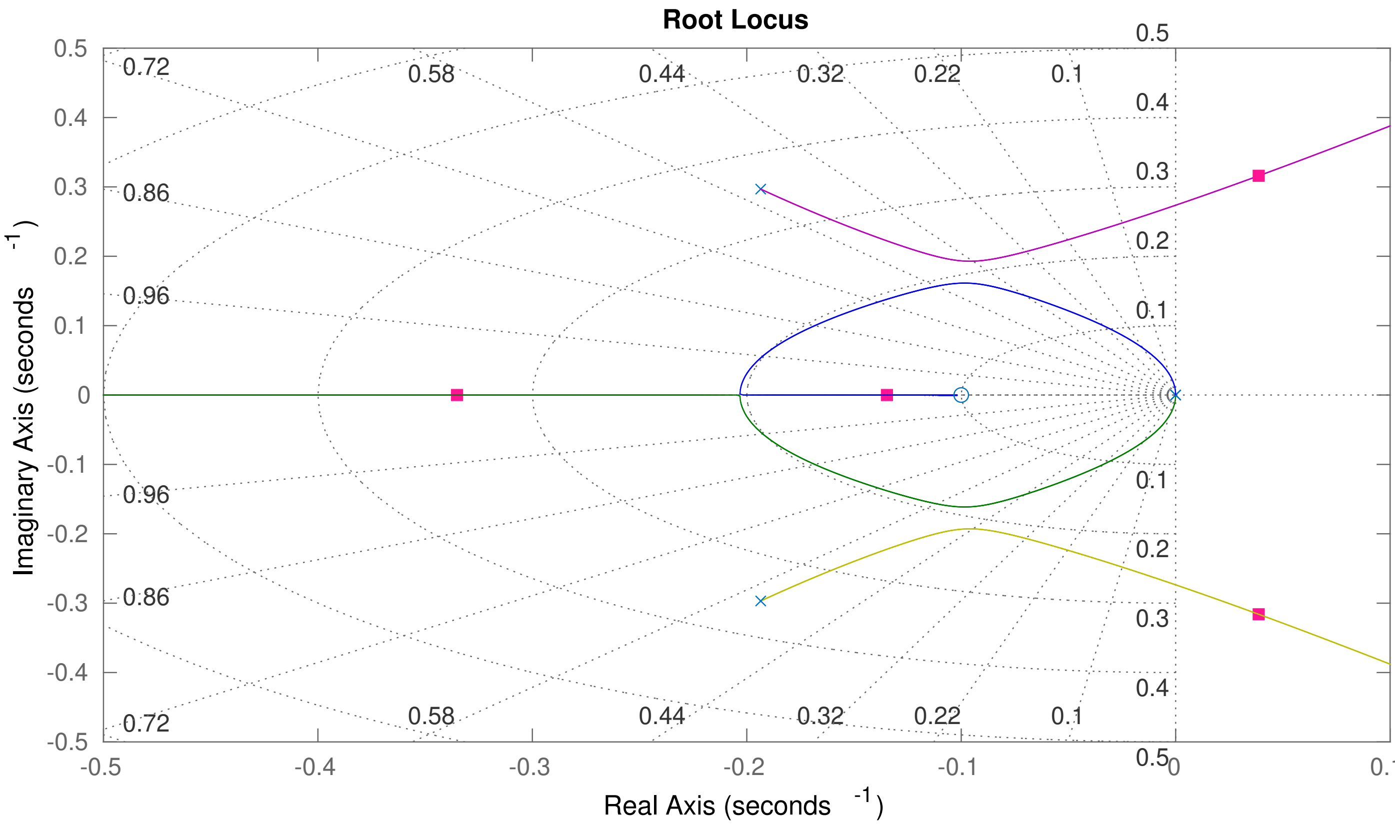


Localizer hold mode

- What about the effect of the slant range R on the performance of the localizer hold mode?
- The coupler gain $K_c = 10$ for $R = 5nm$ is not good for other slant ranges.
- Example: $K_c = 10$ for $R = 1nm$.

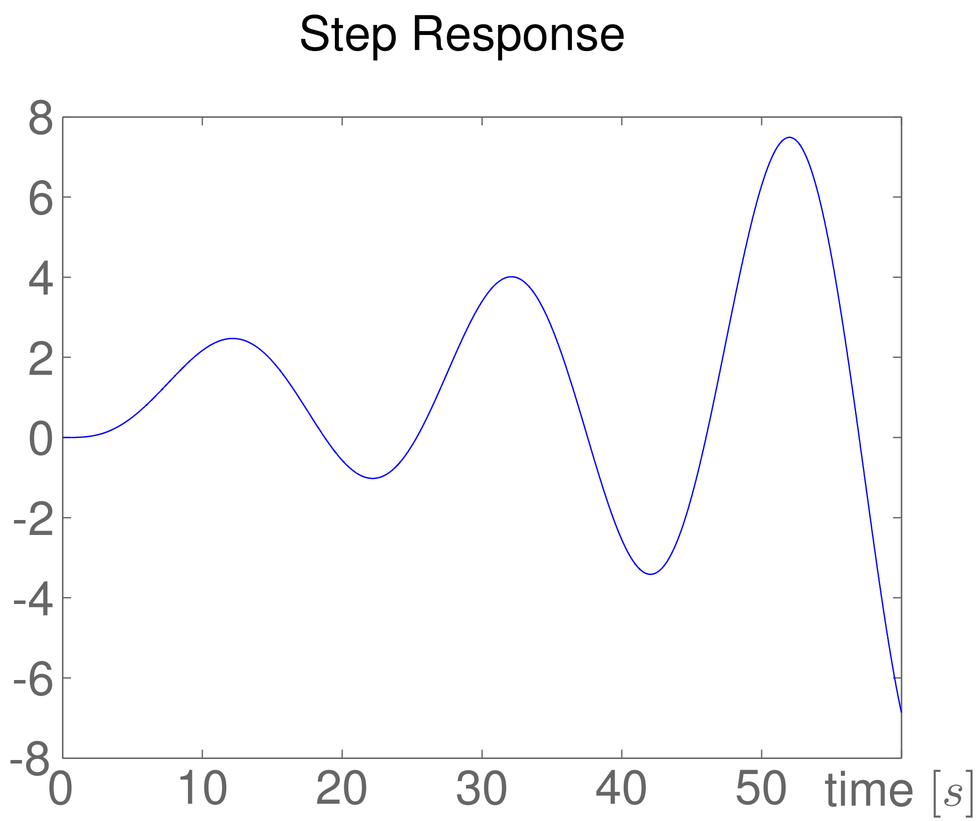
Localizer hold mode

Boeing 747-200 example: Localizer Root Locus, $R = 1nm$



Localizer hold mode

Boeing 747-200 example: Localizer step for slant range of 1 nm.



Localizer hold mode

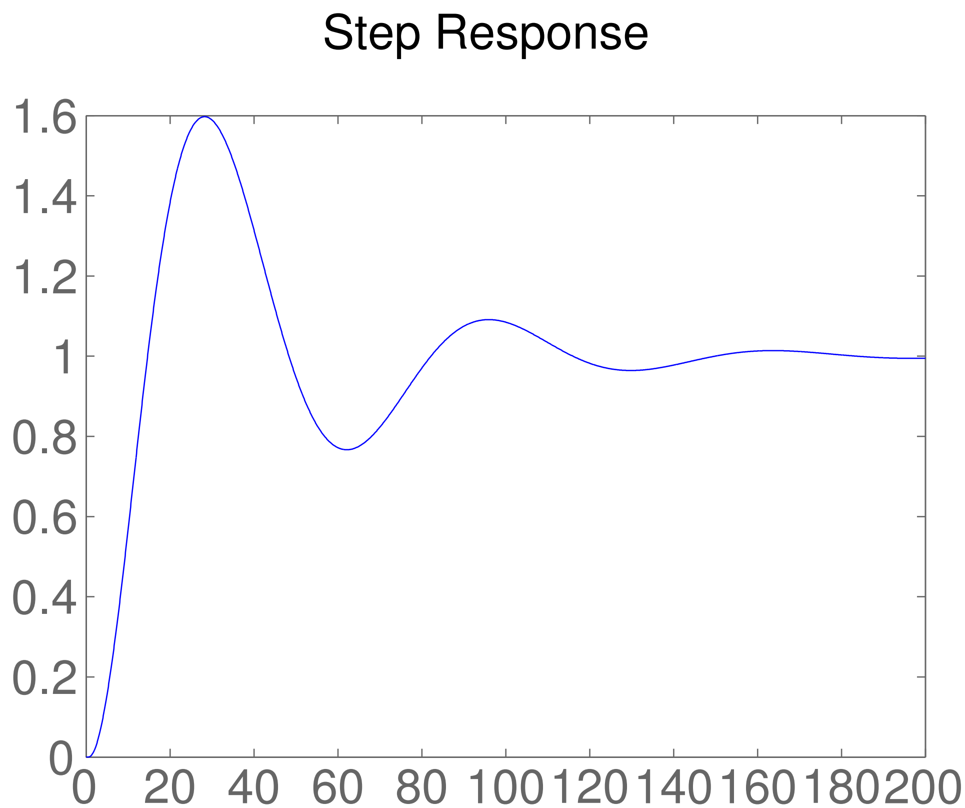
- The coupler gain $K_c = 10$ for $R = 5nm$ is not good for other slant ranges.
- Example: $K_c = 10$ for $R = 1nm$: unstable response for shorter slant ranges.
- Solution?
  - Add compensator to coupler:

$$H_{compensator} = \frac{5\left(s^2 + 0.4s + 0.2\right)}{s^2 + 2s + 1}$$

  - Gain scheduling for K_c with varying R.

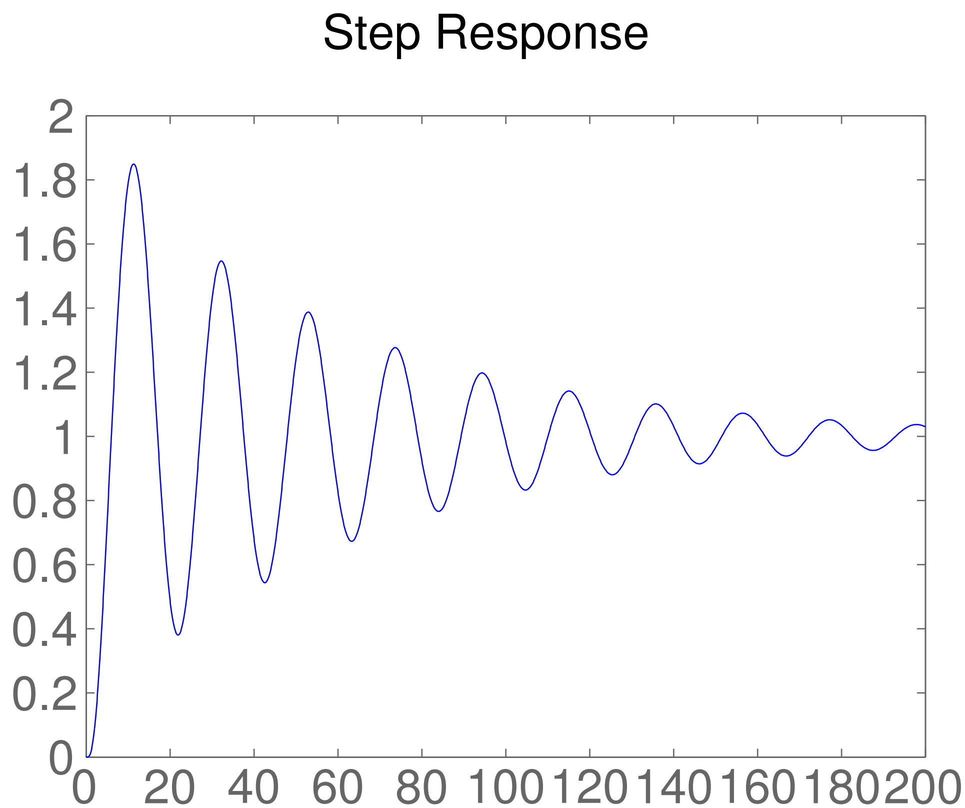
Localizer hold mode

Boeing 747-200 example: Localizer step for $K_c = 10, R = 5nm$, with compensator



Localizer hold mode

Boeing 747-200 example: Localizer step for $K_c = 10, R = 1nm$, with compensator



Localizer hold mode

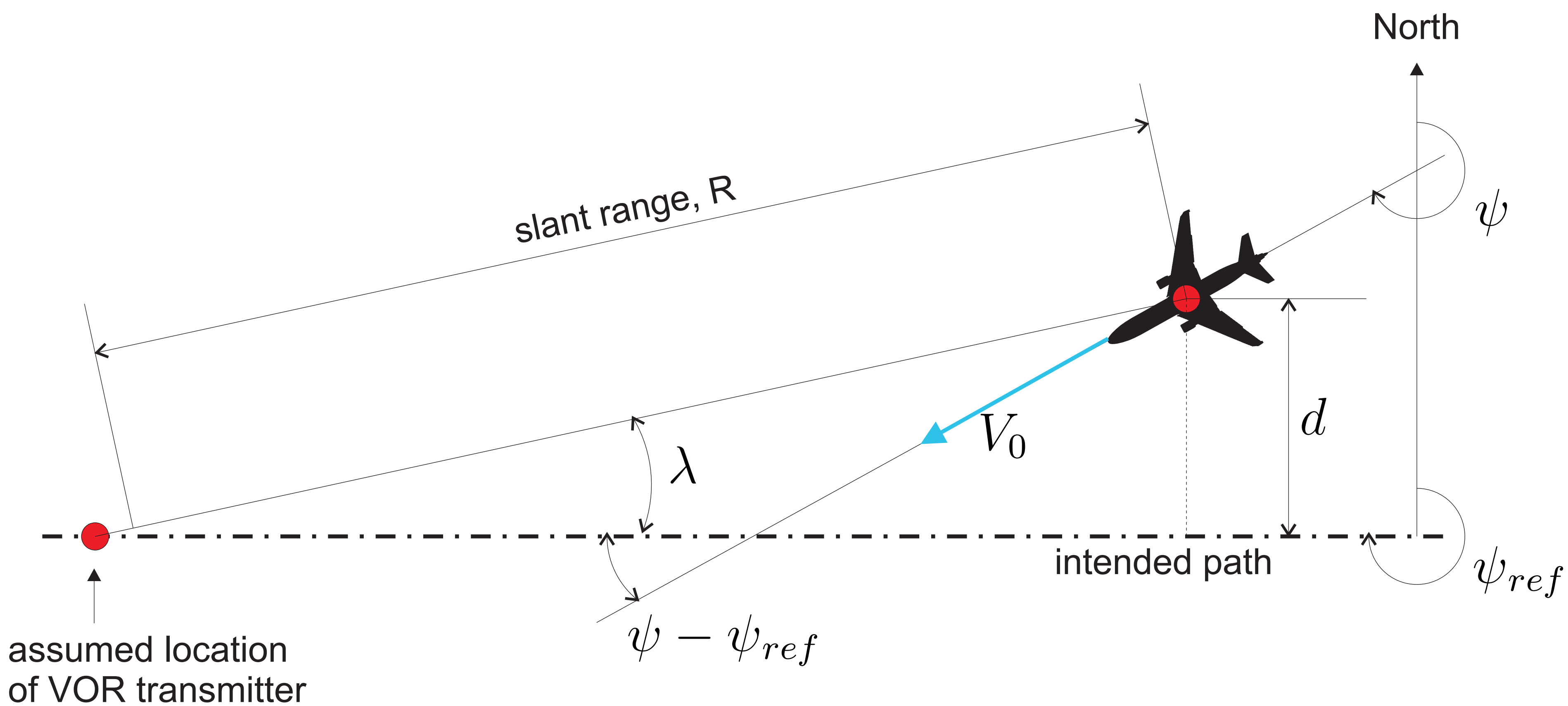
Remarks:

- Localizer is placed at the end of the runway, so the airplane has already touched down at $R = 1nm$: short slant range performance not critical.
- If constant performance (damping, frequency) is required, the coupler gain must be varied as a function of R, which requires distance measuring equipment (DME).

VOR hold mode

- VOR (VHF Omnidirectional Range) hold mode.
- Prevents pilot from manually having to follow a VOR bearing during the flight.
- Assumption: A heading angle control system is already present.
- VOR mode is similar to the localizer mode, but there are some differences (presented later on).
- Geometry of the VOR hold mode:

VOR hold mode



VOR hold mode

Assumptions:

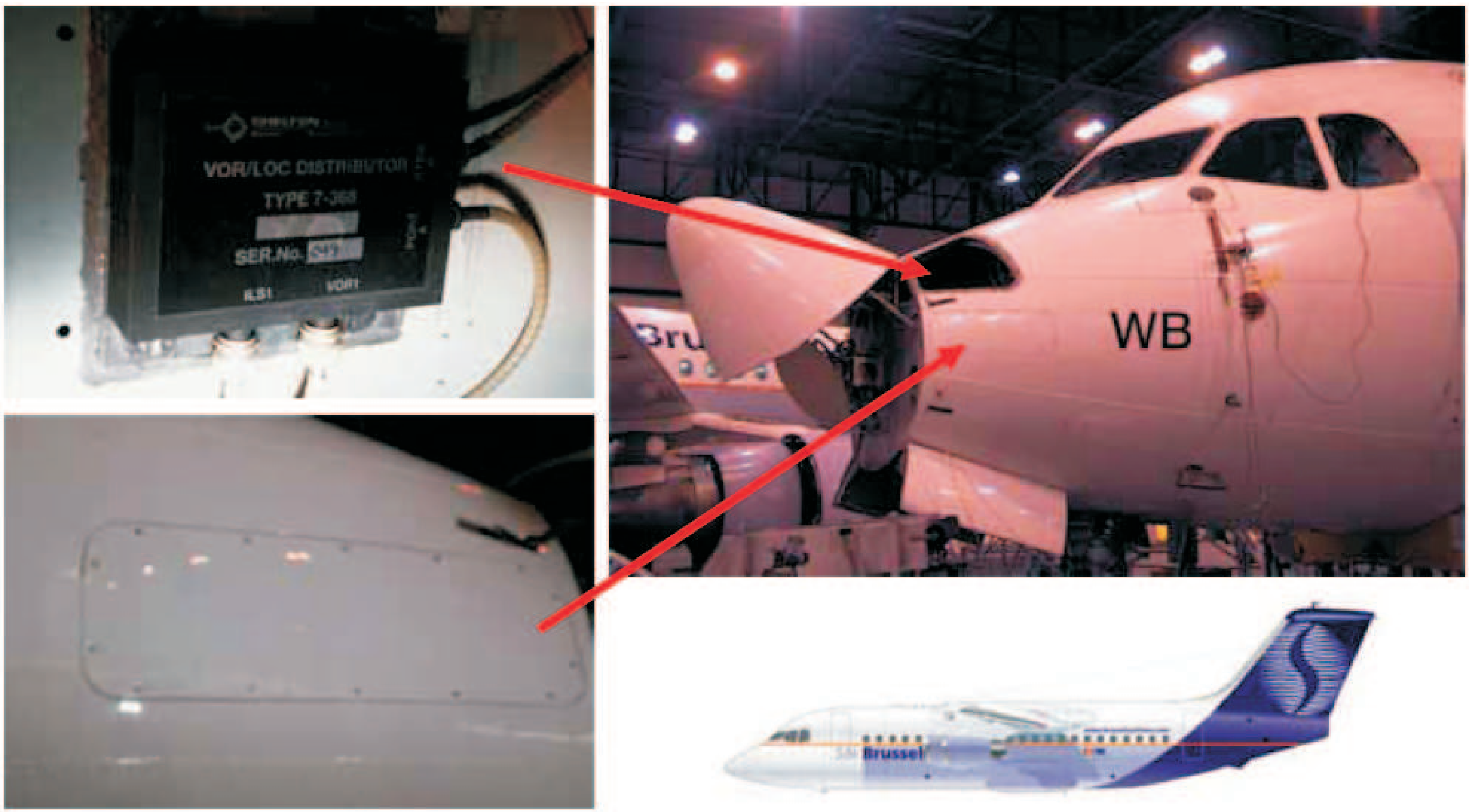
- Airplane VOR antenna coincides with cg.
- cg driven along VOR beam centerline: intended path.
- VOR error angle λ is sensed by VOR receiver.
- Airplane is kept on centerline by heading angle command.
- VOR beam width is $360°$.
- Any speed/lift changes due to rolling are automatically compensated.
- VOR guidance and control is independent of longitudinal control.

VOR hold mode

Pictures of VOR antennas

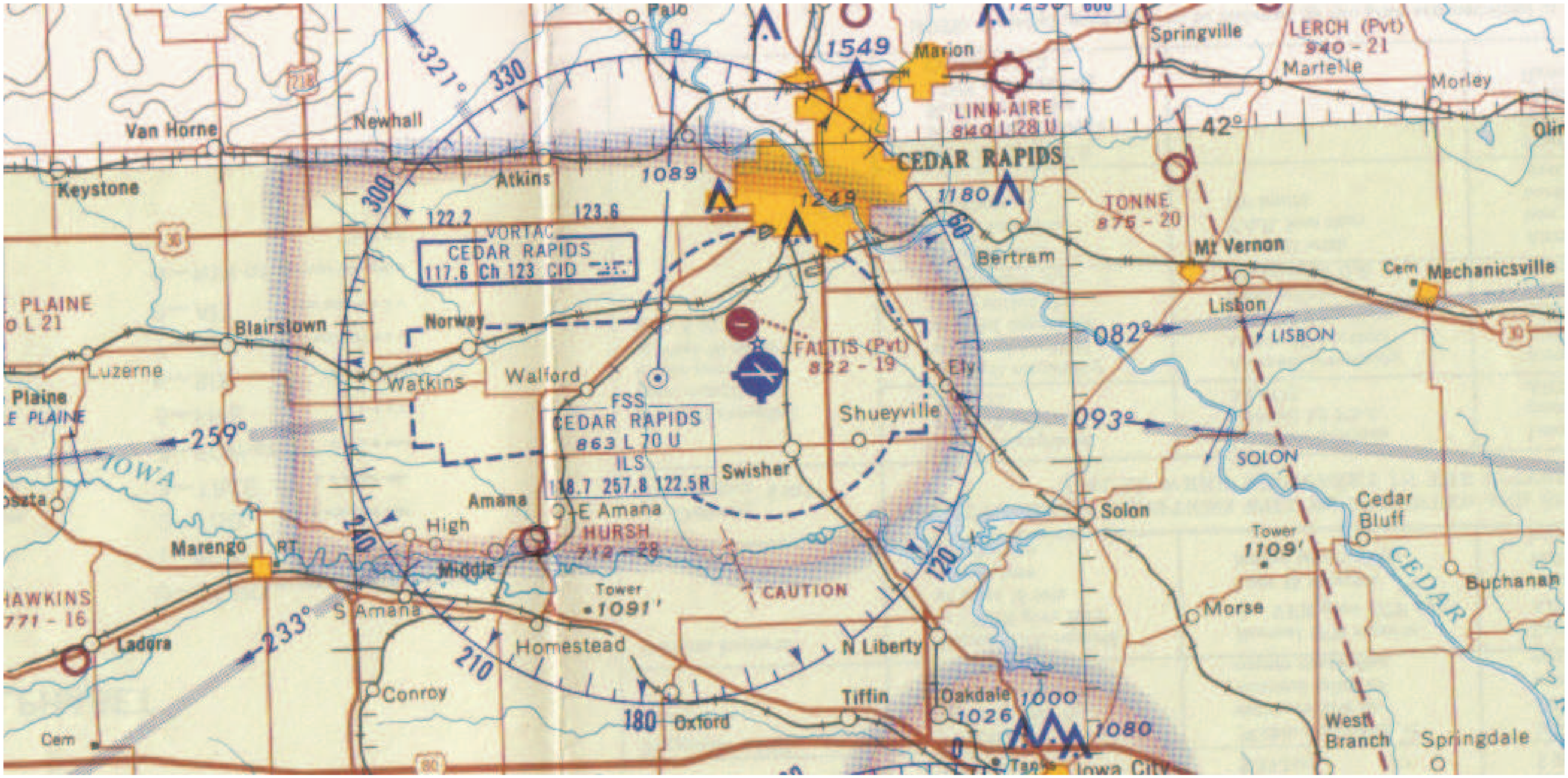
VOR hold mode

Picture of VOR antenna on BAe RJ 85:



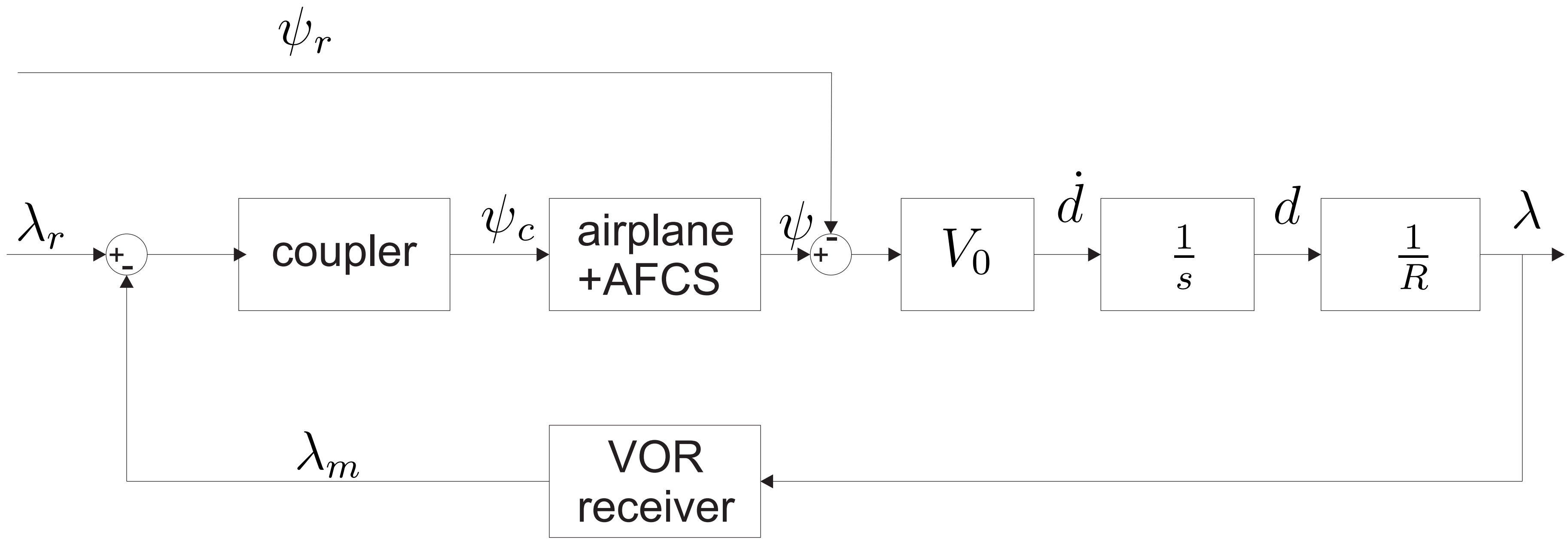
VOR hold mode

Indication of VOR beacons on aviation charts:



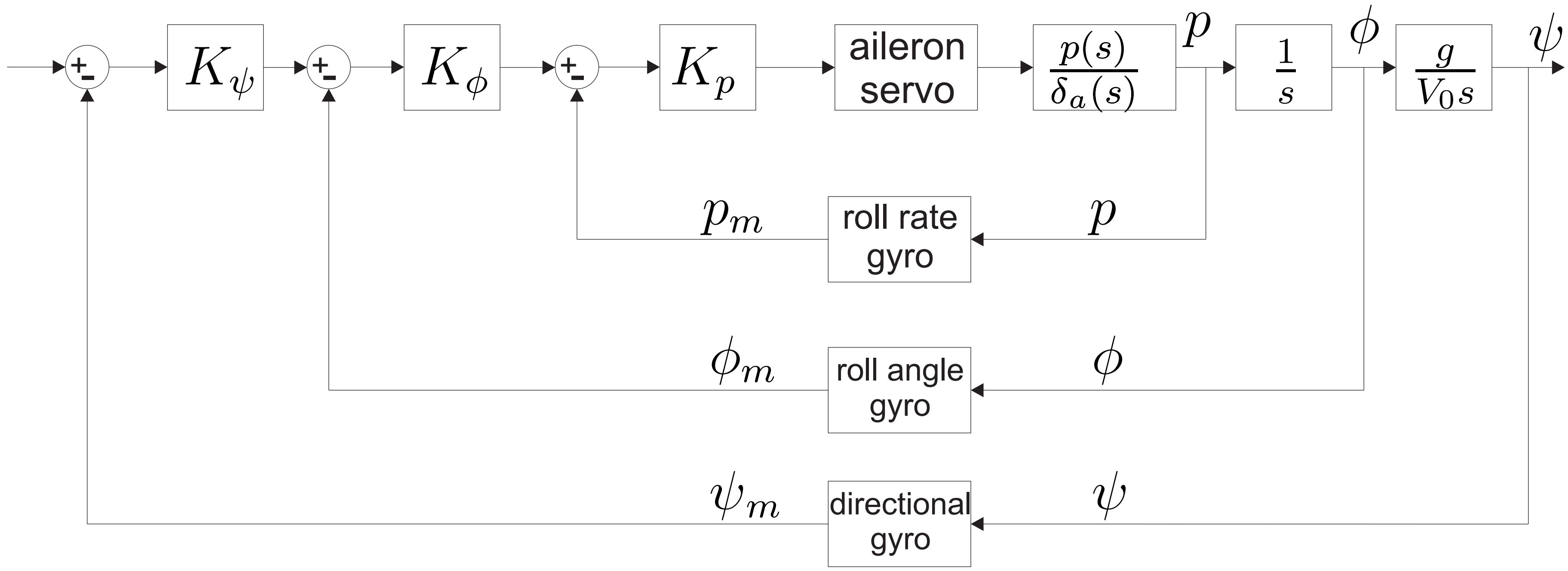
VOR hold mode

Block diagram of a VOR hold loop:



VOR hold mode

Inner loops:



VOR hold mode

Differences between VOR and localizer:

- VOR transmitter beamwidth is $360°$, while the localizer beamwidth is only $5°$.
- The range of the VOR mode is much larger than the localizer (130nm vs 20nm): larger variation in slant range.
- There is no signal directly above the VOR beacon. This problem is not encountered with the localizer, since it is located at the end of the runway.

VOR hold mode

Cessna 182 example.

Cessna C182

Dimensions:

- -wing area: 16.2 m2
- -mean chord: 1.5 m
- -wingspan 10.97 m

Flight condition: cruise

- -altitude: 1500 m
- -speed: 241.5 km/h
- -weight: 12 kN

VOR hold mode

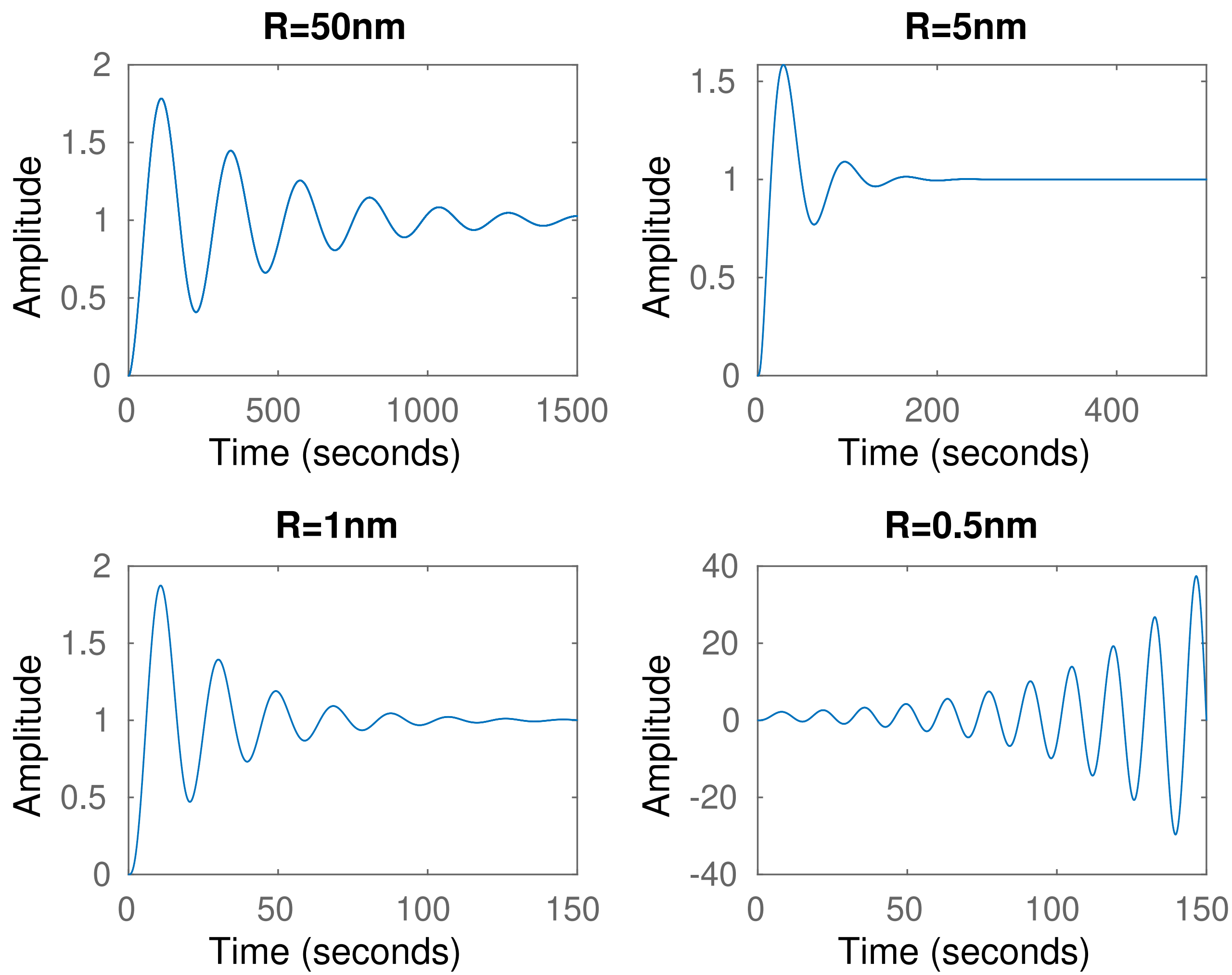
Cessna 182 example.
Results for inner loop designs:

- $K_p = -1$
- $K_\phi = 1$
- $K_\psi = 2$

Test outer loop performance with fixed coupler gain $K_c = 10$, for 4 different slant ranges:

- $R = 50nm$, $R = 5nm$, $R = 1nm$, $R = 0.5nm$
- Effect of slant range on damping?

VOR hold mode



VOR hold mode

Cessna 182 example, observations:

- System works for a wide range of slant ranges without coupler gain adjustment.
- For short slant ranges damping diminishes as the aircraft approaches the transmitter (typical behaviour for constant gain loops)
- Gain scheduling of the coupler gain can be used to obtain constant performance with respect to slant range (up to a minimum slant range value).
- Relationship between change in VOR angle error and slant range.

Summary

- Longitudinal navigational tasks:
  - Glideslope hold mode, sensor: glideslope receiver.
  - Automatic flare, sensor: radar altimeter.
- Lateral navigational tasks:
  - Localizer hold mode, sensor: localizer receiver.
  - VOR hold mode, sensor: VOR receiver.
- Coupler includes integrating action to compensate for turbulence.
- Compensators or gain scheduling methods can be used to maintain performance for varying slant ranges.

Course overview

1. Introduction, evolution of AFCS and flight dynamics.
2. Laplace domain, transfer functions.
3. 1st+2nd order systems, poles and zeros.
4. Root locus method.
5. State space formulation, pole placement, Ackerman, LQR.
6. Basic controllers: P,PI,PID. Ziegler Nichols tuning.
7. Frequency response and Bode plots.
8. Polar plot of frequency response, lead-lag compensators.

Course overview

9. Performance and handling qualities.
10. Dynamic SAS: yaw, pitch, phugoid and dutch roll dampers.
11. Static SAS: Angle of attack, load factor and sideslip feedback.
12. Longitudinal autopilot modes: pitch, altitude, airspeed, climb/descent rate.
13. Lateral autopilot modes: (coordinated) roll angle, heading modes.
14. Navigational autopilot modes: glideslope, flare, localizer, VOR.

Course examination

AE4-301 Exam:

- Online proctored exam via Möbius.
- Closed book: no notes, no formula sheets.
- Exam is scheduled for Friday November 5th, 13:30-15:30.
- Check Brightspace for updates!
- Old exams on Brightspace.

AE4-301 Automatic Flight Control System Design

Ewoud Smeur
Delft University of Technology - AE - C&S

October 15, 2020