Stability augmentation systems

Stability augmentation systems make the aircraft more stable. There are SASs for both the dynamic stability (whether the eigenmotions don't diverge) and the static stability (whether the equilibrium position itself is stable). First, we'll look at the dynamic stability: how can we effect the eigenmotion properties? Second, we'll examine the static stability: how do we make sure an aircraft stays in a steady flight?

1 Dampers – Acquiring dynamic stability

An airplane has several eigenmotions. When the properties of these eigenmotions don't comply with the requirements, we need an SAS. The SAS is mostly used to damp the eigenmotions. Therefore, we will now examine how various eigenmotions are damped.

1.1 The yaw damper: modelling important systems

When an aircraft has a low speed at a high altitude, the Dutch roll properties of the aircraft deteriorate. To prevent this, a **yaw damper** is used. An overview of this system can be seen in figure 1. The yaw damper gets its input (feedback) from the **yaw rate gyro**. It then sends a signal to the **rudder servo**. The rudder is then moved in such a way that the Dutch roll is damped much more quickly than usual. As a designer, we can only influence the yaw damper. However, we do need to know how the other systems work as well. For this reason, we model those systems. We usually do assume that the model of the aircraft is known. (Or we use the one that is derived in the Flight Dynamics course.) So, we only examine the other systems.

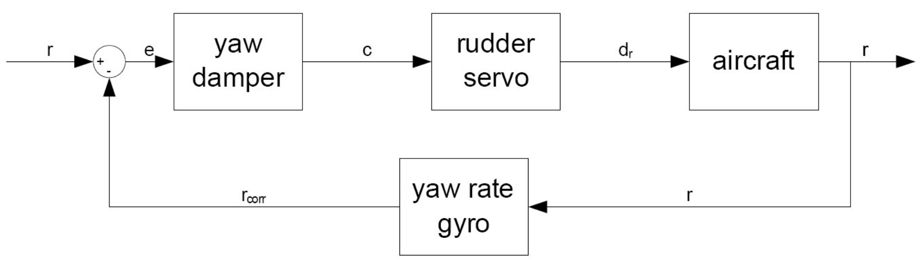

Figure 1: An overview of the yaw damper system.

First, let's look at the gyro. Gyros are generally very accurate in low frequency measurements, but not so good in high frequency regions. So, we can model our gyro as a low pass filter, being

$$H_{gyro}(s) = \frac{1}{s + \omega_{br}}. \tag{1.1}$$

The **gyro break frequency** ω_{br} (above which the performance starts to decrease) is quite high. In fact, it usually is higher than any of the important frequencies of the aircraft. Therefore, the gyro can often also be simply modelled as $H(s) = 1$. In other words, it can be assumed that the gyro is sufficiently accurate.

Now let's examine the rudder servo actuator. Actuators are always a bit slow too respond: they lag behind the input. So, we model the rudder as a lag transfer function, like

$$H_{servo}(s) = \frac{K_{servo}}{1 + T_{servo}s}. \tag{1.2}$$

The time constant T_{servo} depends on the type of actuator. For slow electric actuators, $T_{servo} \approx 0.25$. However, for fast hydraulic actuators, $T_{servo} \approx 0.05$ to 0.1. This time constant (or equivalently, the servo break frequency $\omega_{br_{servo}}$) can be very important. If it turns out to be different than expected, the results can also be very different. So, it is often worth while to investigate what happens if T_{servo} varies a bit.

1.2 The yaw damper: determining the transfer function

Now we'll turn our focus to the yaw damper. We know that the yaw damper has to reduce the yaw rate. But it shouldn't always try to keep the yaw rate at zero. In this case, the pilot will have a hard time to change the heading of the aircraft. Thus, a **reference yaw rate** r is also supplied to the system. This yaw rate can be calculated from the desired heading rate $\dot{\psi}$ by using

$$r = \dot{\psi} \cos\theta \cos\phi. \tag{1.3}$$

In this equation, θ is the pitch angle and ϕ is the roll angle. Both of them thus need to be known. Alternatively, we can also assume that the aircraft is in a horizontal steady turn. In this case, we have

$$L \sin\phi = \frac{mg}{\cos\phi} \sin\phi = mU\dot{\psi} \qquad \Rightarrow \qquad \psi = \frac{g}{Us}\phi. \tag{1.4}$$

In this equation, U is the forward velocity of the aircraft. Also, note that we have assumed that ϕ is small (by using $\tan\phi \approx \phi$) and that we have transformed the equation to the frequency domain (by replacing $\dot{\psi}$ by $s\psi$).

But even if we don't know r, we can still get the system working. In this case, we can use a **washout circuit**, which is much less expensive. We then simply incorporate a washout term in the controller, being

$$H_{washout}(s) = \frac{\tau s}{\tau s + 1}. \tag{1.5}$$

This will cause the yaw damper to fight less when a yaw rate is continuously present. In other words, the system 'adjusts' itself to a new desired yaw rate. The time constant τ is quite important. For too high values, the pilot will still have to fight the yaw damper. But for too low values, the yaw damper itself doesn't work, because the washout circuit simply adjusts too quickly. A good compromise is often at $\tau = 4s$.

Finally, we look at the yaw damper transfer function. In this transfer function, we have proportional, integral and derivative action. If the rise time should be reduced, we use proportional action. If the steady state error needs to be reduced, we add an integral action. And if the transient response needs to be reduced (e.g. to reduce overshoot) we apply a derivative action. In this way, the right values of K_p, K_I and K_D can be chosen.

Sometimes, the optimal values of the gains K_p, K_I and K_D differ per flight phase. In this case **gain scheduling** can be applied. The gains then depend on certain relevant parameters, like the velocity V and the altitude h. In this way, every flight phase will have the right gains.

1.3 The pitch damper

When an aircraft flies at a low speed and a high altitude, the short period eigenmotion has a low damping. To compensate for this, a **pitch damper** is used. The pitch damper is in many ways similar to the yaw damper. Also the set-up is similar. Only this time, the elevators and a pitch rate gyro are used, instead of the rudder and a yaw rate gyro. These two parts are modelled by

$$H_{gyro}(s) \approx 1 \qquad \text{and} \qquad H_{servo}(s) \approx \frac{K_{servo}}{1 + T_{servo}s} \approx \frac{1}{0.25s + 1}. \tag{1.6}$$

Just like with the yaw damper, the **reference pitch rate** q needs to be calculated. This time, this can be done by using

$$L = nW = nmg = mg + mUq \qquad \Rightarrow \qquad q = \frac{g}{U}(n-1). \tag{1.7}$$

Alternatively, a washout circuit can again be used. This washout circuit again has the function given in equation (1.5). Also, a value of $\tau \approx 4$ is again a good compromise. Just like a yaw damper, also the pitch damper has proportional, integral and derivative actions.

1.4 The phugoid damper

To adjust the properties of the phugoid, we can use a **phugoid damper**. It is very similar to the previous two dampers we have seen. However, this damper uses the measured velocity U as input. Its output is sent to the elevator. The speed sensor and the elevator servo are modelled as

$$H_{V-sensor}(s) \approx 1 \qquad \text{and} \qquad H_{servo}(s) \approx \frac{K_{servo}}{1 + T_{servo}s} \approx \frac{1}{0.05s + 1}. \tag{1.8}$$

Note that for the servo now a break frequency of $\omega_{br} = 20$ Hz is assumed.

A **reference velocity** U is also needed by the system. This reference velocity is simply set by the pilot/autopilot. Alternatively, a washout circuit can be used. This washout circuit is the same as those of the yaw and pitch damper. And, just like the previous two dampers, again proportional, integral and derivative actions can be used.

When using a phugoid damper, one should also keep in mind the short period motion properties. Improving the phugoid often means that the short period properties become worse.

2 Feedback – Acquiring static stability

Before an aircraft can be dynamically stable, it should first be statically stable. In other words, we should have $C_{m_\alpha} < 0$ and $C_{n_\beta} > 0$. Normal aircraft already have this. But very manoeuvrable aircraft, like fighter aircraft, do not. (Remember: less stability generally means more manoeuvrability.) Then how do we make these aircraft statically stable?

2.1 Angle of attack feedback

To make an aircraft statically stable, feedback is applied. The most important part is the kind of feedback that is used. First, we'll examine **angle of attack feedback** for longitudinal control. In other words, the angle of attack α is used as a feedback parameter. First, we have to model the angle of attack sensor and the (canard) servo actuator. This is often done using

$$H_{\alpha-sensor}(s) \approx 1 \qquad \text{and} \qquad H_{servo}(s) \approx \frac{1}{0.025s + 1}. \tag{2.1}$$

So, now a break frequency $\omega_{br} = 40$ is used for the servo.

For angle of attack feedback, usually only a proportional gain K_α is used. By using the models of the sensor and actuator (and of course also the aircraft), a root locus plot can be made. With this root locus plot, a nice value of the gain K_α can be chosen. This gain is then used to determine the necessary canard deflection δ_{canard}. This is done using

$$\Delta\delta_{canard} = K_\alpha \cdot \Delta\alpha. \tag{2.2}$$

However, a check does need to be performed on whether the canard deflections can be achieved. If gust loads can cause a change in angle of attack of $\Delta\alpha = 1^\circ$ and the maximum canard deflection is 25°, then K_α should certainly not be bigger than 25, or even be close to it for that matter.

2.2 Load factor feedback

There is a downside with angle of attack feedback. It is often hard to measure α accurately. So instead, **load factor feedback** can be applied. Now the value of n is used as feedback. As models for the sensor and actuator, we again use

$$H_{n-sensor}(s) \approx 1 \qquad \text{and} \qquad H_{servo}(s) \approx \frac{1}{0.025s+1}. \tag{2.3}$$

We also need a model for the aircraft. Normally, we assume that such a model is known. However, the transfer function between the load factor n and the canard deflection δ_c is usually not part of the aircraft model. So, we simply derive it. For that, we first can use

$$\Delta n = \frac{\dot{w}}{g} = \frac{U \tan \dot{\gamma}}{g} \approx \frac{U\dot{\gamma}}{g} = \frac{U\gamma s}{g}. \tag{2.4}$$

We now divide the equation by δ_c. If we also use $\gamma = \theta - \alpha$, then we find that

$$\frac{n(s)}{\delta_c(s)} \approx \frac{Us}{g}\left(\frac{\theta(s)}{\delta_c(s)} - \frac{\alpha(s)}{\delta_c(s)}\right). \tag{2.5}$$

The transfer functions from δ_c to both θ and α usually are part of the aircraft model. So we assume that they are known. The transfer function between n and δ_c is thus now also known. All that is left for us to do is choose an appropriate gain K_n. And of course, again it needs to be checked whether this gain K_n doesn't result into too big canard deflections.

The load factor sensor also has a downside. It is often hard to distinguish important accelerations (like the ones caused by turbulence) from unimportant accelerations (like vibrations due to, for example, a firing gun). Good filters need to be used to make sure a useful signal is obtained.

2.3 Sideslip feedback

Previously we have considered longitudinal stability. For lateral stability, **sideslip feedback** can be used. (However, sideslip feedback is not yet applied in practice.) With sideslip feedback, the sideslip angle β is used as feedback parameter for the rudder. The β-sensor and the rudder are usually modelled as

$$H_{\beta-sensor}(s) \approx 1 \qquad \text{and} \qquad H_{servo}(s) \approx \frac{1}{0.05s+1}. \tag{2.6}$$

The transfer function between the sideslip angle β and the rudder deflection δ_r usually follows from the airplane model. Now that the model is in place, a nice gain K_β can be chosen for the system. This should then give it the right properties.

There is a small problem with sideslip feedback. It can generate a lateral phugoid mode of vibration. To compensate for this, another feedback loop is often used, where the roll rate is used as feedback for the ailerons. This then reduces the effects of the lateral phugoid motion.