Chapter 4
Robust Autopilot Design for Aerodynamically Controlled Reusable Launch Vehicle

4.1 Introduction

The launch vehicle considered in the previous chapter is a symmetric launch vehicle, with no coupling between the pitch, yaw and roll axes. Since the vehicle is flying above the atmospheric phase, there are no aerodynamic effects, and the parameters are slow varying in nature. For this class of launch vehicle, a linear controller with a predesigned gain schedule is sufficient to stabilize and control the vehicle under the entire operating range. In this chapter, the control of more complex launch vehicle configurations is considered. The first configuration is an aerodynamically controlled launch vehicle in the atmospheric flight regime with decoupled pitch/yaw/roll dynamics, and very wide range of parameter perturbations. The second configuration is a typical reentry vehicle configuration which is aerodynamically controlled, has a wide range of parameter perturbations and flight environment variations as well as pitch/yaw/roll coupled dynamics. In order to meet the challenges of such systems, the linear autopilot design adopted in the previous chapter is not sufficient, since a linear design aimed at robustness for a very large range of parameter perturbation leads to an inevitable compromise on the achieved performance.

In the first configuration, the vehicle is highly aerodynamically unstable. The control moment is achieved by the aerodynamic forces produced by deflection of the tips of the fins provided on the launch vehicle, while in the launch vehicle considered in the earlier chapter, the control was provided by deflecting the thrust. Thus unlike the earlier case where the control force is known with reasonable accuracy, in this case, the actual control moment depends on the aerodynamic coefficients and the dynamic pressure, which is a function of the trajectory, in addition to the fin deflection angle. The control effectiveness depends to a large extent on the accuracy of the preflight estimates of the fin aerodynamic parameters. Due to the decoupled nature of the plant, it is still possible to design a classical controller for the nominal plant, however, the high degree of
aerodynamic instability along with high parameter uncertainties severely constrains the achievable stability margins with linear autopilot design. For this configuration, an online gain tuning method is studied in this chapter to improve the robustness of the linear autopilot in the face of high uncertainties while maintaining near nominal performance. In the second configuration, the reentry vehicle is unpowered and aerodynamically controlled through control surfaces. There is strong coupling between the pitch/yaw and roll planes, and the vehicle encounters a wide range of environmental conditions, flying from hypersonic to subsonic Mach numbers. LQG controller design is one of the methods for achieving a decoupled design and has been studied for this vehicle configuration. Since the launch vehicle configuration considered in the present chapter is highly coupled, the standard LQG procedure has been used, with the same weights on the rigid body states as obtained in the TP-LQG procedure developed in the previous chapter. The significant contribution of this chapter is the development of an adaptive gain tuning method to improve the robustness of the linear autopilot for aerodynamically controlled reusable launch vehicle autopilot in the face of high parametric uncertainties.

The chapter is organized as follows:

Section 4.2 describes the adaptive control methodology. The proposed online gain tuning Scheme for robustness improvement of aerodynamically controlled launch vehicle autopilot is presented in Section 4.3. Section 4.4 discusses LQG based autopilot design for aerodynamically controlled descent launch vehicle with high degree of coupling. Section 4.5 summarises the study results.

4.2 Adaptive Control Methodology

In adaptive controllers, the controller parameters are automatically adjusted online\textsuperscript{162,163} as the system changes. The controller therefore acts to maintain the closed loop system response in the presence of variations in the system. The simplest form of adaptive controller is a gain scheduling controller, where the gains are adjusted on the basis of a suitable system parameter, which correlates well with the change in the system dynamics. The gains are often scheduled offline, as a function of time, altitude or dynamic pressure\textsuperscript{167}. Such schemes can only adapt to those variations in the system, which are known a priori and which manifest in a measured system variable. Although gain
scheduling is extremely popular in practice, the disadvantage of gain scheduling is that it is an open-loop adaptation scheme, with no real learning. Hence the robustness of the design is limited. Adaptive gain tuning controllers can cater to much higher levels of uncertainty.

The two approaches for adaptive controller design are the Model Reference Adaptive Control (MRAC)\(^{170,171}\) design methodology and the Self Tuning Controller Approach. In the MRAC approach, a reference model is specified and an adaptation algorithm adjusts the controller gains so that the output of the actual system follows that of the reference model. The self tuning approach was originally proposed by Kalman and clarified by Astrom and Wittenmark\(^{168}\). The controller is called self-tuning since it has the ability to tune its own parameters. Since for the launch vehicle plant, the mapping from the plant parameters to the controller parameters is known with reasonable accuracy, the self tuning controller approach can be adopted. There are effectively two loops in the controller structure: (i) An inner loop consisting of a conventional controller with varying parameters and (ii) an outer loop consisting of an identifier and a design box which adjusts these controller parameters\(^{169}\). The self tuning regulator is very flexible with respect to its choice of controller design methodology and to the choice of identification scheme.

The representative block diagram of a self-tuning controller is given in Fig 4.1.

![Fig. 4.1 Representative Block Diagram of a Self Tuning Controller](image)
4.3 Online Gain Tuning Scheme for Robustness Improvement of Aerodynamically Controlled Launch Vehicle Autopilot

The aerodynamically controlled, unstable launch vehicle configuration has a high aerodynamic disturbance coefficient, due to the large aerodynamic moment arm (the distance between the effective aerodynamic force location and the centre of gravity). The classical digital autopilot (DAP) design carried out based on frozen time slices, for the nominal plant forms the baseline design. In order to achieve reasonable aerodynamic disturbance rejection, it is necessary to have a high bandwidth autopilot design. This results in a compromise between the achievable aeromargin and the gain margin as seen in Fig. 4.2. Hence the system is very sensitive to parameter changes, especially those changes which affect the gain of the system. The major parameter affecting the gain is the variation in the fin tip control effectiveness, the variation in this parameter can be very large due to poor preflight aerodynamic parameter estimates. Hence perturbations in the fin tip control effectiveness have the maximum effect on the system stability. The effect of the other parameter perturbations on the system stability is relatively small. For the lower bound performance, with low fin effectiveness, the low frequency stability margin (aero margin) is lost with the baseline design, and is manifested as low frequency oscillations in the simulations. For the upper bound performance, with high fin effectiveness, the system loses gain margin, which is manifested as high frequency oscillations. The effect of the control effectiveness on the stability is shown in Fig. 4.2. It is seen that with the control effectiveness perturbed by 20% on either side with all other parameters nominal, the system is close to instability. Reduction in the control effectiveness shrinks the entire Nyquist plot and effectively reduces the system gain, affecting the aeromargin, while increase in the control effectiveness expands the entire plot and causes loss of phase and gain margins.

4.3.1 Response with the Baseline Design

The nominal response, obtained through six degrees of freedom trajectory simulation throughout the ascent phase with the baseline linear controller is shown in Fig. 4.3 and Fig. 4.4.
Stability analysis with nominal and perturbed control effectiveness

Fig 4.2 – Effect of Fin Control Effectiveness on Stability

Fig 4.3 Nominal Response with Linear Controller - Body Rates
Six degrees of freedom trajectory simulations are also carried out by varying the fin control effectiveness parameter $C_{Na}$ of all the four fins. The lower bound limiting case beyond which the system is driven to instability is given in Fig 4.5 and Fig 4.6. This corresponds to loss in the low frequency margin i.e. the aeromargin, manifesting as low frequency oscillations of frequency 0.4 Hz with pitch rate as high as 5 deg/s peak to peak at around 40s, which is the region of highest aerodynamic instability. The corresponding upper bound limiting case response is given in Fig 4.7 and Fig 4.8. In this case, the instability manifests as high frequency oscillations of 1.8 Hz, corresponding to loss of gain margin. When control effectiveness is further increased slightly, the instability is clearly manifested (Fig 4.9 and Fig 4.10).

![Fig 4.4 Nominal Response with Linear Controller – Fin Deflection](image)

![Fig 4.5 Response with Lower Bound Control Effectiveness - Comparison of Rates with Nominal](image)
Fig 4.6 Response with lower bound control effectiveness - fin deflection

Fig 4.7 Response with Upper Bound Control Effectiveness - Comparison of Rates with Nominal

Fig 4.8 Response with Upper Bound Control Effectiveness - Expanded Plot

Fig 4.9 Response with Control Effectiveness beyond Upper Bound - Body Rate
Fig 4.10 Response with Control Effectiveness beyond Upper Bound-Fin Deflections

4.3.2 Methodology for Gain Adaptation

The analysis in the previous section shows the sensitivity of the response to variations in the control effectiveness parameter. The effect of control effectiveness is to increase or reduce the effective gain in the system. Once a reliable online estimate of the control effectiveness is obtained, it is possible to tune the gain of the linear controller to compensate for the deviation in control effectiveness from the preflight value, based on which the baseline controller is designed. This makes the autopilot insensitive to perturbations in this parameter, and can improve the robustness of the system significantly. This section describes the proposed adaptive gain tuning method which modifies the linear autopilot gain online, based on the estimate of the parameter uncertainty. The method is demonstrated for the aerodynamically unstable vehicle with control through aerodynamic surfaces.

The success of the scheme depends on the accuracy with which the control effectiveness can be estimated. The parameters of the system vary continuously with respect to time. Using a true estimator can lead to additional delay due to the convergence time of the estimator. Since the system is already having low margins, this can further destabilize the loop. Hence the estimation is carried out using the physical relations between the parameters based on the flight telemetry data and the preflight vehicle data. The actual control moment obtained throughout the flight regime is estimated from the available
telemetry data and the wind measurement. The expected control moment based on the preflight data for which the autopilot design is carried out are calculated. The ratio of the expected and flight control moments is used for computing the adaptive gain.

The estimation of the control effectiveness and the gain tuning is done on a continuous basis. Hence the actual variation over the entire Mach number range is taken into account. The major issue affecting the stability and convergence of this scheme is the speed of adaptation. In this scheme, the control parameters effectively change as a function of the states of the system. The effect of the control parameters on the system matrix coefficients in closed loop is explained below.

Plant equations:
\[
\begin{align*}
\dot{X}_p &= A_p X_p + B_p U_p \\
Y_p &= C_p X_p + D_p U_p
\end{align*}
\]
(4.1)

Controller equations:
\[
\begin{align*}
\dot{X}_c &= A_c X_c + B_c U_c \\
Y_c &= C_c X_c + D_c U_c
\end{align*}
\]
(4.2)

\[
U_c = Y_p \\
U_p = Y_c
\]
(4.3)

(4.4)

From Equations 4.1 to 4.4,
\[
\begin{bmatrix}
\dot{X}_p \\
\dot{X}_c
\end{bmatrix} =
\begin{bmatrix}
A_{eq11} & A_{eq12} \\
A_{eq21} & A_{eq22}
\end{bmatrix}
\begin{bmatrix}
X_p \\
X_c
\end{bmatrix}
\]

\[
Y_p =
\begin{bmatrix}
C_{eq1} & C_{eq2}
\end{bmatrix}
\begin{bmatrix}
X_p \\
X_c
\end{bmatrix}
\]
(4.5)

where,
\[
A_{eq11} = [A_p + B_p D_c C_{eq1}]
\]
\[
A_{eq12} = [B_p C_c + B_p D_c C_{eq2}]
\]
\[
A_{eq21} = [A_c + B_c C_{eq1}]
\]
\[
A_{eq22} = [B_c C_{eq2}]
\]
\[
C_{eq1} = \left[ I - D_p D_c \right]^{-1} C_p \\
C_{eq2} = \left[ I - D_p D_c \right]^{-1} D_p
\]

Thus it is seen that the closed loop system matrices are functions of both plant and controller. If the controller parameters vary as a function of the system states, the linear relationship no longer holds good.

If the adaptation is carried out too fast, the control will change as fast as the states, and the underlying linearity assumption based on which the baseline linear control design has been carried out will be lost, and linear analysis tools cannot be used. Hence the estimation process has to be slow enough to ensure that the dependence of the controller on the plant is very slow compared to the dominant time constants of the system. This consideration decides the averaging window for the estimate, which effectively acts as a low pass filter. Theoretical results for the optimum estimation rate are likely to be conservative; hence the optimum averaging period is arrived at based on simulation studies. The overall schematic is given in Fig. 4.11.

The different steps in the formulation of the adaptive gain control methodology are as follows:

**Step-1: Computation of total pitch moment**

(i) Compute the derivative of the rate measurement ($\dot{\theta}$), obtained from flight to get the angular acceleration ($\ddot{\theta}$)

(ii) Obtain moment of inertia ($I$) variation with respect to time from pre-stored moment of inertia table.

(iii) Compute the total pitch moment as $I \ddot{\theta}$
Step-2: Prediction of disturbance moment from nominal

Total disturbance moment $M_D = \text{aerodynamic disturbance moment } T_{AX} + \text{moment due to cg offset & thrust misalignment } T_{TX}$

Aerodynamic moment $T_{AX} = S.Q. C_{N\alpha} l_\alpha \alpha$.

Surface area $S$ is a constant preflight value.

Obtain dynamic pressure $Q = \frac{1}{2} \rho V^2$ from the flight measured velocity and the altitude, since the atmospheric density $\rho$ is a function of altitude.

Obtain the aerodynamic force coefficient $C_{N\alpha}$ and the aerodynamic moment arm $l_\alpha$ with respect to time from the pre-stored table.

Obtain angle of attack from the measured longitudinal ($V_{long}$) and lateral velocity ($V_{lat}$) components as $\alpha = \tan^{-1}(\frac{V_{lat}}{V_{long}})$.

Obtain the aerodynamic disturbance moment.

Moment due to centre of gravity offset and thrust misalignment $T_{TX}$ is assumed to be zero since for solid motor, these values are very small. Alternatively, they can be estimated from the steady component of the control demand.

Step-3: Estimation of flight control moment

Flight Control moment $M_C = \text{Total pitch moment } - \text{Total disturbance moment } M_D$
Step-4: Computation of the expected control moment for the flight

The control command $\delta_{\text{flight}}$ from flight is available. Assume fin deflection $\delta = \text{control command } \delta_{\text{flight}}$ and compute the expected control moment as follows

$$\text{Expected control moment} = 4 \cos(45) \cdot Q_{\text{nominal}} \cdot S \cdot CN_{\delta_{\text{nominal}}} \cdot L_{\delta_{\text{nominal}}} \cdot I_{\delta_{\text{nominal}}}$$

Step-5: Computation of the factor for tuning the control commands

The ratio of the expected and flight control moments gives the factor by which the control commands have to be increased / decreased. The control gain is modified by this factor.

After extensive studies with different averaging time periods, 5 s averaging was seen to give the best results, since the slow variations in the control effectiveness with Mach number was captured without sacrificing the linear controller design assumptions.

4.3.3 Evaluation of the Scheme with Typical Results

With the adaptive gain tuning scheme, the response with the upper and lower bound of control effectiveness is similar to that with nominal. The range of control effectiveness parameter perturbations was further expanded on either side and the limit values seen. The responses with increased control effectiveness parameter (increased from +73% to +170%) are given in Fig 4.12 and Fig 4.13 and those with reduced control effectiveness (reduced from -40% to -50%) are given in Fig 4.14 and Fig 4.15.

![Fig 4.12 Body Rate with Control Effectiveness Perturbation far beyond the Upper Bound, with Adaptation](image-url)
Fig 4.13 Fin Deflection with Control Effectiveness Perturbation far beyond the Upper Bound, with Adaptation

Fig 4.14 Body Rate with Control Effectiveness Perturbation far Below the Lower Bound, with Adaptation

Fig 4.15 Fin Deflection with Control Effectiveness Perturbation Below Lower Bound, with Adaptation
It is seen that the range of control effectiveness perturbations that can be tolerated by the system has increased tremendously on the upper side, and by additional 10% on the lower side as shown in Fig 4.16.

![Fig. 4.16 Control Coefficient with and without Adaptive Scheme--with Control Effectiveness Perturbation Alone](image)

Since the operating point is brought back to the nominal with gain adaptation, simultaneous perturbations in the aerodynamic disturbance parameters and the control effectiveness are tolerable with this approach, with perturbation levels higher than those tolerable with the linear gain scheduled controller. With worst case parameter perturbations on the aerodynamic disturbance force coefficient and the center of pressure, the tolerable dispersions in the control effectiveness is only 60% on the positive side and 10% on the negative side with the linear controller. This increases to 160% on the positive side and ~30% on the negative side for the scheme with control adaptation, as seen in Fig 4.17.

![Fig. 4.17 Control Coefficient with and without Adaptive Scheme--with Both Control Effectiveness and Aeroparameter Perturbation](image)
Further perturbation studies with different possible combinations of the thrust misalignment, normal force dispersion and pitching moment dispersion were carried out for the upper bound and lower bound values of the fin control effectiveness parameter, over the entire flight regime. Fig. 4.18a shows the pitch (q), yaw (r) and roll (p) rates of a limiting perturbation case which failed with the linear controller. Fig. 4.18b shows the rates for the same perturbation case with the proposed method, which is going through smoothly.

**Fig : 4.18a  Body Rates without Adaptive Gain Tuning**

**Combined Perturbation of Parameters**

**Fig : 4.18b  Body Rates Using Adaptive Gain Tuning**

The method shows promising results, with the allowable range of parameter perturbations increased by a large factor. The main advantage of the scheme is that the operating point is brought back to that of the nominal case, for which the autopilot is originally designed, leading to optimum performance. The proposed adaptive gain tuning scheme is seen to increase the range of allowable parameter perturbation by a large extent.
4.4 Descent Phase Autopilot Design for Aerodynamically Controlled Launch Vehicle

Autopilot design for a typical reentry launch vehicle configuration, which is controlled by aerodynamic surfaces, and which has strong coupling between the pitch, yaw and roll planes is considered in this section. The plant has multiple inputs (elevons and rudders) and outputs (pitch, yaw and roll angles) and is a true MIMO system. Due to the MIMO nature of the plant, the classical controller cannot be used as the baseline controller in this case. The Linear Quadratic Gaussian (LQG) technique, which is suitable for MIMO systems, has been considered for the present design. The total longitudinal and lateral dynamics of the vehicle are considered together, and a controller designed for the coupled launch vehicle. The linearised short period system model is given in Appendix 3. Aerodynamic and control coefficients are given in Appendix 4.

The flight envelope is very wide, with Mach number varying from hypersonic to subsonic, and widely changing aerodynamic characteristics. The system eigen values over the flight regime are tabulated in Table 4.1.

<table>
<thead>
<tr>
<th>Mach 1</th>
<th>Mach 2</th>
<th>Mach 3</th>
<th>Mach 4</th>
<th>Mach 5</th>
</tr>
</thead>
<tbody>
<tr>
<td>-0.3856 + 5.7510i</td>
<td>-0.0602 + 1.8214i</td>
<td>-0.0309 + 0.7650i</td>
<td>-0.0771 + 1.4405i</td>
<td>-0.0357 + 1.1292i</td>
</tr>
<tr>
<td>-0.0505</td>
<td>-0.0262</td>
<td>0.0105</td>
<td>-0.0075</td>
<td>0.0064</td>
</tr>
<tr>
<td>0.0300</td>
<td>0.0152</td>
<td>-0.0032</td>
<td>0.0089</td>
<td>-0.0030</td>
</tr>
<tr>
<td>0.2902 + 1.2363i</td>
<td>1.9616</td>
<td>-1.5090</td>
<td>-2.7226</td>
<td>-2.0516</td>
</tr>
<tr>
<td>0.2902 - 1.2363i</td>
<td>0.0636</td>
<td>1.4854</td>
<td>2.6613</td>
<td>2.0277</td>
</tr>
<tr>
<td>-0.8327</td>
<td>0.0099</td>
<td>0.0298</td>
<td>0.0128</td>
<td></td>
</tr>
<tr>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
</tbody>
</table>

4.4.1 LQG Controller Design

As in the case of the expendable launch vehicle, the weights are tuned based on the transient response results. Weights of 750 on the pitch, yaw and roll attitude angles and 10 each on the other states with unity weight for the control inputs, was seen to give reasonably good transient response. Individual controllers were first designed at each Mach number with these weights. The response was seen to be satisfactory. Even though the model is coupled, only the states corresponding to the longitudinal equation (pitch
attitude angle, body rate, angle of attack and velocity) are significant, when the command is given in pitch (Fig 4.19). Similarly, only those states corresponding to the lateral dynamics (yaw attitude angle, body rate and side slip angle) are significant when the command is given in yaw (Fig 4.20). The design has thus achieved decoupling between the planes.

Fig 4.19  Response to Pitch Command, with Individual Controller Design at Different Mach Nos.

Fig 4.20  Response to Yaw Command, with Individual Controller Design at Different Mach Nos.
In order to reduce the computational burden, the possibility of having a single controller to control the launch vehicle over the entire flight envelope is explored. The open loop response of the system over the entire duration with controller design at Mach 3 is given in Fig. 4.21. It is seen that there is considerable variation in the gain and phase over the flight regime. Hence a single controller is not sufficient to stabilize the system over the entire flight regime. Since the control is through aerodynamic surfaces whose effectiveness changes considerably over the flight duration, it is necessary to compensate for these changes by adjusting the gain of the controller. After experimentation, it was seen that the controller at Mach 3 is the best suited for the baseline design. The gains at other flight regimes are appropriately adjusted, with the following logic: At high Mach numbers, the aerodynamic effectiveness of the yaw control surfaces (rudders) is low and the effective gain in the system is very low. At lower Mach numbers, the vehicle is the denser part of the atmosphere, and the effectiveness of both elevons and rudders is high. However, the aerodynamic disturbance is also very high in this regime. The set of gain factors that stabilize the system are given in Table 4.2

<table>
<thead>
<tr>
<th>Mach No</th>
<th>1</th>
<th>2</th>
<th>3</th>
<th>4</th>
<th>5</th>
</tr>
</thead>
<tbody>
<tr>
<td>Gain factor - elevon</td>
<td>0.8</td>
<td>2</td>
<td>1</td>
<td>1</td>
<td>1.2</td>
</tr>
<tr>
<td>Gain factor - rudder</td>
<td>4</td>
<td>2</td>
<td>1</td>
<td>4</td>
<td>4</td>
</tr>
</tbody>
</table>

With these gain factors applied to the control command to the elevons, reasonably good transient response is obtained with the single controller. The rise time is more at transonic regime (Mach 1) than with the individual design (around 6 s as against 3s). Convergent oscillations are seen in the body rates, which die down within 2 s (Fig 4.22 and Fig 4.23). Similar behaviour is there at other Mach numbers also. The elevon and rudder demands in all cases are quite small (Fig 4.24 - Fig 4.29).
Fig 4.21  Bode Plot of System Response Over the Flight Duration with Controller Designed at Mach 3.

Fig 4.22 Transient Response at Mach 1-Pitch Command
It is seen that gain scheduling with LQG controller is feasible for reusable launch vehicle autopilot design, achieving good decoupling of the pitch, yaw and roll planes. Further, robustness to control effectiveness perturbations, may be achieved as in the ascent phase, by tuning the scheduling gain parameter online.
Pitch transient response at Mach 4, design at Mach 3, with gain tuning

Fig 4.26 Transient Response at Mach 4 - Pitch Command

Yaw transient response at Mach 4, design at Mach 3, with gain tuning

Fig 4.27 Transient Response at Mach 4 - Yaw Command

Transient response in pitch at Mach 5, design at Mach 3, with gain tuning

Fig 4.28 Transient Response at Mach 5 - Pitch Command
4.5 Summary

An online gain tuning method has been studied to improve the robustness of the linear autopilot for aerodynamically controller reusable launch vehicle autopilot in the face of high uncertainties, and demonstrated through long period simulation studies. The method shows promising results, with the allowable range of parameter perturbations increased by a large factor. The main advantage of the scheme is that the operating point is brought back to that of the nominal case, for which the autopilot is originally designed, leading to optimum performance. For the re-entry vehicle configuration, decoupling is seen to be achieved with an LQG controller. It is also seen that gain scheduling can enable a single LQG controller to be used over the entire descent phase regime with gain adaptation.

In such complex systems with wide variation in the system parameters, there is a need to define the worst case set of systems which the autopilot needs to cater to. The evolution of a systematic method for robust stability analysis and design is addressed in the next two chapters.

The significant contribution of this chapter is the development and validation of an adaptive gain tuning scheme which increases the allowable range of parameter perturbations by as large factor, compared to the baseline linear controller.