Stellingen behorend bij het proefschrift

Development of a design methodology for handling qualities excellence in fly by wire aircraft
John Gibson, 23 February 1999

1) The serious handling difficulties found in many fly by wire aircraft were caused by artefacts introduced by the control law designers. Excellent handling qualities can be generated easily when their true nature is properly understood. (This thesis)

2) Great efforts are now being made by research organisations to determine criteria by which past high order PIO events could have been predicted accurately. It would be far more useful for this effort to be devoted to improving design methods that absolutely prevent such PIO.

3) The necessary condition for high order PIO to occur is simply stated: the response dynamics permit it. This simple statement is confirmed by its corollary: such PIO is preventable by the provision of easily defined response dynamics that do not permit it. (This thesis)

4) The "First Golden Rule" for elimination of high order PIO in flight control design is that a sufficient measure of unlagged direct signal path must be maintained between the pilot’s inceptor and the control surface actuation. The rare exception is its proof. (This thesis)

5) The "Second Golden Rule" is that the control law structure must be properly defined before the control law design is commenced. For all practical purposes, the fly by wire control law designer will find the pilot models of the 1960s to be sufficient to determine good handling qualities. (This thesis)

6) Optimal control pilot models can be used to explain pilot induced oscillations and other high order handling difficulties but cannot be used to prevent them. (This thesis)

7) Stick force per g has long been assumed to be the primary measure of sensitivity in pitch handling. Its significance appears, however, to be related primarily to traditional structural protection, and is of minor importance for pitch handling. (This thesis)

8) Flight control designers have little inclination to delve into the past, where they would find much of value and importance to the present.

9) "To see what is in front of one’s nose needs a constant struggle." (George Orwell, "In front of your nose", Vol. 4, Collected essays). Examples of this are many of the author’s discoveries about handling qualities, made from material that has been in the public domain for several decades, waiting to be interpreted anew.
10) Occam's Razor, "No more things should be presumed to exist than are absolutely necessary", could be expressed alternatively as "The simplest answer that takes account of all the facts is probably correct". It is an excellent guide in the search for handling qualities design methods.

11) Certain of the many fallacious aerodynamic and flight mechanics theories still widely taught to the general pilot population may lead to misunderstanding and unsafe aircraft control practices. To teach them is pointless and they should have been rooted out half a century ago.

12) The author's first handling qualities theory was hypothesised over 60 years ago, at the age of eight. His later admission that aircraft were not actually steered by articulating the propeller shaft in the desired turn direction is proof that he can occasionally be wrong.

13) To have written a thesis after a lifetime of industrial experience, instead of before, is a great surprise to the author. To have been able to retain his faculties to do this so near to his allotted span of 70 years is reason for gratitude.
1) De ernstige besturingsproblemen van veel “Fly-by-Wire” vliegtuigen werden kunstmatig veroorzaakt door de ontwerpers van het besturingssysteem. Goede besturingseigenschappen kunnen op eenvoudige wijze worden verkregen indien men hun ware aard begrijpt. (Dit proefschrift)

2) Onderzoeksorganisaties doen thans veel inspanningen om criteria te bepalen waarmee gevallen van PIO’s, Pilot Induced Oscillations, uit het verleden nauwkeurig voorspeld hadden kunnen worden. Het zou nuttiger zijn deze inspanningen te richten op het ontwikkelen van criteria die PIO’s in nieuwe ontwerpen met zekerheid uitsluiten.

3) De nodige voorwaarde voor het optreden van ‘High Order PIO’s” kan eenvoudig worden uitgedrukt als: de dynamica van de vliegtuigresponsie op stuursignalen maakt ze mogelijk. Deze bewering wordt bevestigd door het omgekeerde: zulke PIO’s zijn te voorkomen door de vliegtuigresponsie zo te ontwerpen dat PIO’s uitgesloten zijn. (Dit proefschrift)

4) De “eerste gulden regel” voor het elimineren van “High order PIO’s” in het ontwerp van een besturingssysteem is te zorgen voor een voldoende mate van onverdraagde directe signaaloverdracht tussen het besturingsorgaan van de vlieger en de besturingsservo. De zeldzame uitzonderingen bevestigen deze regel. (Dit proefschrift)

5) De “tweede gulden regel” luidt dat de structuur van de regelwet goed moet worden vastgelegd alvorens de regelwet te ontwerpen. Voor het verkrijgen van goede besturingseigenschappen in de praktijk zijn de vliegermodellen van de jaren ‘60 voor de ontwerper van “Fly-by-Wire” besturingssystemen zeer bruikbaar. (Dit proefschrift)

6) De zogeheten “Optimal Control Pilot Models” kunnen gebruikt worden om PIO’s en andere hogere orde besturingsproblemen achteraf te verklaren doch ze zijn onbruikbaar om ze te voorkomen. (Dit proefschrift)

7) Lange tijd veronderstelde men dat de “stuurkracht per g” de belangrijkste maat was voor de gevoeligheid van de langsbesturing van het vliegtuig. De betekenis blijkt vooral gelegen in de conventionele bescherming van de constructie tegen overbelasting, het belang voor de langsbesturing is echter beperkt. (Dit proefschrift)

8) Ontwerpers van vliegtuigbesturingssystemen hebben weinig neiging om in het verleden te graven, hoewel zij daar veel waardevols voor het heden zouden kunnen vinden.

9) “To see what is in front of your nose needs a constant struggle” (George Orwell , “In front of your nose “, Vol. 4, Collected essays). Een goed voorbeeld daarvan is dat veel ontdekkingen over “Handling Qualities” door de auteur werden gedaan aan de hand van materiaal dat al tientallen jaren gepubliceerd en vrij toegankelijk lag te wachten om opnieuw te worden geïnterpreteerd.
10) Het onder de benaming “Occam’s Razor” bekende gezegde, “Men moet niet meer veronderstellingen doen dan absoluut nodig is”, wordt ook wel geformuleerd als: “De eenvoudigste verklaring van alle feiten is waarschijnlijk de juiste”. Dit is een uitnemende leidraad in het onderzoek naar ontwerpmethoden voor ‘Handling Qualities’.

11) Sommige van de vele onjuiste aerodynamische en vliegmechanische theorieën die nog steeds aan vliegers worden onderwezen kunnen leiden tot misvattingen en onveilig stuurgedrag. Het onderwijzen van zulke theorieën is zinloos en zou al een halve eeuw geleden afgeschafft moeten zijn.

12) De eerste theorie van de auteur over besturingseigenschappen werd zestig jaar geleden, op achtjarige leeftijd, geponeerd. Dat hij later moest toegeven dat vliegtuigen in werkelijkheid niet werden gestuurd door de as van de propeller in de gewenste richting van de bocht te draaien, geeft aan dat hij soms ongelijk kan hebben.

13) Het voltooien van een proefschrift na een levenslange carrière in de industrie in plaats van ervoor, kwam als een grote verassing voor de auteur. Dat hij zijn vermogen om dit te doen kon behouden tot zo dicht bij de grens van 70 jaar is reden voor dankbaarheid.
Development of a design methodology for handling qualities excellence in fly by wire aircraft

TR 3298
Development of a methodology for excellence in handling qualities design for fly by wire aircraft

PROEFSCHRIFT

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College of Aeronautics, Cranfield

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SUMMARY

The recent period has been filled with exceptionally interesting developments and advances in aeronautics. The impact of some of these developments on flight control is outlined in this thesis. The results have often revealed a lack of understanding that led to the serious handling difficulties discussed in these pages.

Chapter 2 outlines how an understanding of handling qualities was acquired very slowly indeed, the ability of pilots to cope with an extraordinary range of qualities being a primary reason for this. This ability was also fortunate in that despite the nature of aircraft to be made inherently both controllable and stable without much difficulty, all too often they were either insufficiently stable or were unstable. It took 40 years from the first powered flights to arrive at a comprehensive definition of satisfactory handling qualities. It took another 25 years to expand the definition to the realm of the high performance high altitude jet powered era. In each case the results were available essentially at the end of the major period of development of each class of aircraft, and proved inadequate for the challenges of new technologies.

Chapters 3 to 5 discusses the basic elements of handling qualities. These elements are the dynamics of conventional aircraft, dominated by their aerodynamic and inertial properties, the dynamics of "superaugmented" aircraft, dominated by the flight control system properties, and the dynamics of the human pilot to which the aircraft dynamics should be well matched for satisfactory handling. While the first two element sets can nowadays be computed to an extremely high degree of accuracy, the pilot remains something of an enigma. Decades of research have led to very advanced models of the human pilot functioning that can offer a quite close match to the pilot behaviour in observed events. By contrast, the earliest and very simple pilot models have retained their power to illuminate the response dynamics that satisfy the handling design goals.

Chapter 6 discusses the influence on handling qualities of control system hardware, which has often greatly degraded the handling. Again this was due to misunderstanding and neglect of its importance until the lessons were learned. In aircraft with powered controls, the pilot was now disconnected from the aerodynamic forces on the controls, and artificial feel substitutes had to be supplied. These varied greatly in complexity and effectiveness. The greatly expanded flight envelopes of advanced aircraft often required further hardware complexities in the form of variable or non-linear gearings and inter-axis couplings. In the modern fly by wire era, the pilot-aircraft interface hardware reduces essentially to theceptors and residual artificial feel devices, but handling qualities are as dependent on the qualities of these devices as they ever were.

Chapter 7 presents the core discoveries of the author from which his new methodology for the handling qualities design of control laws was developed. In the circumstances of his industrial employment, it was not possible to conduct experiments except in a very limited way. The material on which his research was based was found in the literature, mainly conference papers, technical journals and research reports in the public domain, dating back to the 1950s. Much of this material, based on parametric formats and criteria, appeared to disagree substantially in the conclusions about response limits for satisfactory handling. From the evidence of poor handling qualities in many early fly by wire aircraft, there was little satisfactory guidance that could be applied to the design of control laws that would ensure acceptable, let alone excellent, handling in such aircraft. The author developed a non-parametric graphical description method that directly illuminated the response qualities desired by pilots. The author found that this method identified rather good agreement in most of the sources about the limits of satisfactory handling.
It could be applied without difficulty to any category of manoeuvre demand control system regardless of its non-classical form.

Chapters 8 and 9 discuss a variety of pilot induced oscillation problems, that is oscillations that are actually sustained by the pilot's efforts to control them. PIO has come to prominence in fly by wire aircraft in the so-called high order PIO. These have caused many catastrophic accidents and serious incidents, mainly in the landing and sometimes in the take off phases of flight. In the aftermath of the author's first encounter with a PIO, he found that there were no criteria or any relevant material in the literature that could have predicted this PIO, but was able to develop both an explanation and a solution. As experimental PIO material became available, it provided further data to confirm the author's initial hypothesis. The author found a universal consistency in the response dynamics associated with high order PIO and in the pilot's reactions to them that enabled him to develop design criteria that have eliminated this particular PIO problem in every subsequent project with which he has been associated.

Chapter 10 describes the application of the author's methodology to the shaping of the time and frequency response dynamics. Since fly by wire systems can be made to provide any desired response dynamics to a reasonably close degree, it is a simple task to design the control laws to the author's criteria, providing good handling from the start rather than after a trial and error process of flight development. The basic tools for this task are a direct feedforward path from inceptor to control actuation to provide instantaneous accelerations and a command path pre-filter as appropriate to tune the response dynamics. The usual aim is to provide an aircraft open loop response that resembles closely, or as closely as is chosen, the idealised $K/s$ integrator-like dynamics with which the pilot can easily interact with low workload. Constraints are placed on the initial time response transients to ensure satisfactory sensitivity.
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**7 The development of new criteria from old data**

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**8 Linear, low and high order pilot induced attitude oscillations**

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Notations and abbreviations

Notations

\( \bar{c}_e \)  mean chord of elevator

\( C_{h\theta} \)  elevator hinge moment coefficient

\( C_L \)  lift coefficient

\( C_{La} \)  lift coefficient due to angle of attack, \( \partial C_L / \partial \alpha \)

\( C_{ma} \)  pitching moment coefficient due to angle of attack, \( \partial C_m / \partial \alpha \)

\( C_{mb} \)  pitching moment coefficient due to elevator angle, \( \partial C_m / \partial \delta_e \)

\( C_{Na} \)  normal force coefficient due to angle of attack, \( \partial C_N / \partial \alpha \)

\( D \)  drag

\( DB \)  attitude dropback

\( dB \)  decibel

\( F_a \)  roll stick force

\( F_C \)  roll demand pre-filter

\( F_e \)  pitch stick force

\( F_1 \)  feedforward pre-filter

\( g \)  normal acceleration due to gravity

\( H_m \)  stick fixed manoeuvre margin

\( H'_m \)  stick free manoeuvre margin

\( H_n \)  stick fixed static margin

\( H'_n \)  stick free static margin

\( H_z \)  frequency in cycles per second

\( K \)  a gain

\( K_c \)  gain of an arbitrary aircraft transfer function \( Y_c \)

\( K_C \)  a command gain

\( K_{DL} \)  direct link roll control gain

\( K_{FF} \)  feedforward gain augmentation

\( K_M \)  roll manoeuvre demand gain

\( K_p \)  gain of the human pilot transfer function \( Y_p \), or roll rate feedback gain

\( K_q \)  gain of an aircraft pitch rate transfer function, or proportional pitch rate gain

\( K_a \)  gain of an aircraft angle of attack transfer function or angle of attack feedback

\( K_\delta \)  stick command scaling gain

\( K_\phi \)  gain of an aircraft roll angle transfer function

\( L \)  lift

\( L_\alpha \)  the lift parameter \( \frac{1}{2} \rho V^2 S \partial C_L / \partial \alpha \)

\( m \)  mass
$n$ normal load factor at centre of gravity
$n/\alpha$ normal load factor per unit angle of attack
$p$ roll rate
$p_{sr}$ steady state roll rate
$q$ pitch rate
$q_c$ commanded pitch rate
$q_e$ closed loop error in commanded pitch rate
$q_m$ peak pitch rate in a time response
$q_{ss}$ steady state pitch rate
$S$ wing area
$S_e$ elevator area
$s$ the Laplace operator
$s_e$ stick deflection
$T_l$ lag time constant in human pilot transfer function
$T_L$ lead time constant in human pilot transfer function
$T_N$ time constant representing pilot neuromuscular lags
$T_r$ roll mode time constant
$T_r$ spiral mode time constant
$T_q$ proportional pitch rate gain in proportional-plus-integral controller
$t_q$ time at peak pitch rate in time response
$T_{b1}$ phugoid numerator time constant in aircraft longitudinal transfer function
$T_{b2}$ short period numerator time constant in aircraft longitudinal transfer function
$t_\gamma$ time delay measure of flight path angle response
$u_{GUST}$ increment in horizontal gust velocity
$V$ true air speed
$V_{md}$ airspeed for minimum total drag
$Y_c$ general aircraft response transfer function
$Y_p$ human pilot transfer function

$\alpha$ angle of attack
$\delta_a$ aileron angle
$\delta_e$ elevator angle
$\delta_{hs}$ pilot roll stick input
$\delta_{es}$ pilot pitch stick input
$\epsilon_q$ closed loop error in commanded pitch rate
$\gamma$ flight path angle
$\dot{\gamma}_{ss}$ steady rate of change of flight path angle

Notations and abbreviations
\( \varphi \)  
roll angle, phase angle of a transfer function

\( \varphi_{os} \)  
overshoot in roll angle after the stick is centred in a roll

\( \{d\varphi / d\omega\}_{\omega(-180)} \)  
rate of change of a transfer function at the frequency for -180° phase angle

\( \theta \)  
pitch angle

\( \theta_c \)  
commanded pitch angle

\( \theta_p \)  
stick pumping amplitude

\( \rho \)  
air density

\( \tau \)  
time delay

\( \omega \)  
frequency

\( \omega_c \)  
frequency of unit gain of pilot-aircraft closed loop frequency response

\( \omega_n \)  
undamped natural frequency

\( \omega_{sp} \)  
undamped natural frequency of short period oscillation

\( \omega_p \)  
undamped natural frequency of phugoid oscillation

\( \omega_d \)  
undamped natural frequency of dutch roll oscillation

\( \omega_\phi \)  
undamped natural frequency of roll transfer function numerator

\( \omega(-180) \)  
frequency of a transfer function for phase angle of -180°

\( 2\omega(-180) \)  
twice the -180° phase angle frequency of a transfer function

\( \xi \)  
aileron control angle

\( \psi \)  
yaw angle

\( \zeta \)  
relative damping ratio

\( \zeta_{sp} \)  
relative damping ratio of short period oscillation

\( \zeta_p \)  
relative damping ratio of phugoid oscillation

\( \zeta_\phi \)  
relative damping ratio of roll transfer function numerator

\( \zeta_d \)  
relative damping ratio of dutch roll oscillation

**Abbreviations**

AC  
Aerodynamic Centre

ACAH  
Attitude Command/Attitude Hold

AGARD  
Advisory Group for Aerospace Research and Development

AIAA  
American Institute of Aeronautics and Astronautics

APC  
Aircraft-Pilot Coupling

BAe  
British Aerospace

CAA  
Civil Aviation Admistration

CAP  
Control Anticipation Parameter

CG  
Centre of Gravity
<table>
<thead>
<tr>
<th>Abbreviation</th>
<th>Full Form</th>
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<tr>
<td>CGI</td>
<td>Computer Generated Image</td>
</tr>
<tr>
<td>CHR</td>
<td>Cooper-Harper Rating</td>
</tr>
<tr>
<td>DOF</td>
<td>Degrees Of Freedom</td>
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<tr>
<td>EAP</td>
<td>Experimental Aircraft Programme</td>
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<tr>
<td>FAA</td>
<td>Federal Aviation Administration</td>
</tr>
<tr>
<td>FBW</td>
<td>Fly By Wire</td>
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<tr>
<td>FCS</td>
<td>Flight Control system</td>
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<tr>
<td>GARTEUR</td>
<td>Group for Aeronautical Research and Technology in Europe</td>
</tr>
<tr>
<td>GCA</td>
<td>Ground Controlled Approach</td>
</tr>
<tr>
<td>LAHOS</td>
<td>Landing Approach High Order Systems</td>
</tr>
<tr>
<td>LOES</td>
<td>Low Order Equivalent System</td>
</tr>
<tr>
<td>MIT</td>
<td>Massachusetts Institute of Technology</td>
</tr>
<tr>
<td>MP</td>
<td>Manoeuvre Point</td>
</tr>
<tr>
<td>mph</td>
<td>miles per hour</td>
</tr>
<tr>
<td>NACA</td>
<td>National Advisory Committee for Aeronautics</td>
</tr>
<tr>
<td>NASA</td>
<td>National Aeronautics and Space Administration</td>
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<tr>
<td>NLR</td>
<td>National Aerospace Laboratory, The Netherlands</td>
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<tr>
<td>NP</td>
<td>Neutral Point</td>
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<tr>
<td>NPL</td>
<td>National Physics Laboratory</td>
</tr>
<tr>
<td>PIO</td>
<td>Pilot Induced Oscillation</td>
</tr>
<tr>
<td>PIOR</td>
<td>Pilot Induced Oscillation Rating</td>
</tr>
<tr>
<td>PR</td>
<td>Pilot Rating</td>
</tr>
<tr>
<td>PRD</td>
<td>Pitch Rate Demand</td>
</tr>
<tr>
<td>QSRA</td>
<td>Quiet STOL Research Aircraft</td>
</tr>
<tr>
<td>RAF</td>
<td>Royal Aircraft Factory</td>
</tr>
<tr>
<td>S/MTD, STOL/MTD</td>
<td>Stol/Maneuver Technology Demonstrator</td>
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<tr>
<td>STOL</td>
<td>Short Take Off and Landing</td>
</tr>
<tr>
<td>TV</td>
<td>Television</td>
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<tr>
<td>USAF</td>
<td>United States Air Force</td>
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<tr>
<td>VAAC</td>
<td>Vectored thrust Aircraft Advanced flight Control</td>
</tr>
<tr>
<td>V/STOL</td>
<td>Vertical and Short Take Off and Landing</td>
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*Notations and abbreviations*
Chapter 1
Introduction

1.1 Background

Handling qualities relate directly to the ease with which a task can be performed. They describe the manner in which the aircraft responds both to the demands of the pilot and to the influence of turbulence, gusts, weapon release, airbrake, flap and undercarriage operation, reheat selection, engine failure, and other disturbances. For most of aviation history, they have depended heavily on the basic aerodynamic stability and control, but in recent times they have become dominated by control augmentation effects in fly-by-wire control systems. Phillips [1989] and Harper/Cooper [1984] give excellent reviews of the whole period.

After the Wright Brothers' first flights in 1903, there was no common ground between pilots and theoreticians in the understanding of aircraft handling qualities. Three decades passed before significant efforts to collect and correlate data on aircraft behaviour began, and to find out what parameters were actually important to pilots.

In the 1940s, quantitative requirements for handling qualities began to be developed into standard sets of criteria. These were refined and expanded in the 1950's and 1960's in a major series of simulator and especially of in-flight experiments with variable stability aircraft. Greater understanding evolved from theoretical studies of the human pilot, and more reliable methods of testing in ground and airborne flight simulators were developed.

Inevitably, most of the research effort was devoted to the needs of military combat aircraft, because of their more demanding and varied tasks. The UK and USA military aircraft specifications have very detailed requirements for handling qualities, though of course the resulting knowledge is directly applicable to all categories since they are related to universal properties of the human pilot. Some of the specifications in current use are becoming lengthy and complex.

The USA and European civil airworthiness authorities each have their own handling qualities requirements to be satisfied for a range of flight tasks [Anon, A, Anon, B]. These have been largely unified and there is little important difference between them. The stringent requirements have an impact on the aerodynamic and engineering design to ensure the highest safety standards. Specific handling requirements tend to be confined to traditional static stability and damping characteristics, and most particularly to the ability to cope with engine failure.

Specification requirements are sometimes the choice of single individuals or small groups, tasked with making sense of often sparse and perhaps conflicting sets of data. One set of parameter limits may be arbitrarily combined with another in untested ways. The result can be an unacceptable or unsatisfactory aircraft satisfying the formal specifications, as illustrated later in this thesis. Flight mechanics can also be involved in ways not quantifiable by single numbers, e.g. pitch and lateral behaviour in the stall region. Where assessment in flight is the only reliable means to determine compliance, the formal method required to demonstrate this does not always represent the way pilots actually form a judgement [Thomas, 1991]. Pilots and control engineers do not always speak the same "language", which can lead to considerable misunderstanding or even mistrust between them.
1.2 New problems

Wilbur Wright [1901] said that the problems of construction and of motive power had been solved to an extent, but that of balance and steering still remained and that other difficulties were of minor importance. This statement is still partly true.

From about the 1970s, the new generations of fly by wire aircraft, in which the handling is essentially dominated by computer control rather than by their aerodynamic properties, exhibited unexpected and often extremely serious handling quality problems. Of these, the emergence of the "high order pilot induced oscillations" (PIO), afflicting the majority of fly by wire aircraft, was the most intractable and led to a number of accidents. There was also a general failure to achieve the high standards of control and handling qualities which had been expected given the power of the new computer control technology. That such problems are not yet universally understood is evident in the recent embarrassingly public accidents to the SAAB Gripen [Knotts et al, 1990, Jensen et al, 1996] and Lockheed YF-22 [Dornheim, 1992] fighter prototypes.

How seriously the overall problem was viewed in the early 1980s can be realised from a widely expressed though informal opinion in that period that digital flight control technology should be abandoned, because of the many failures in its application both to manned aircraft and to missiles. Some strong criticism was made by a senior NASA engineer [Berry 1982], for example:

"Serious problems in flight control emphasise the need for timely and adequate research on flying qualities"

"We have systems capable of providing a wide variety of control responses, but we are not sure what responses or modes are desirable"

- this after 80 years of flight! The early linear quadratic regulator control theories widely used up to that time were savagely criticised by Chandler and Potts [1983] in phrases such as:

"repeatedly found unsuccessful, highly deficient, dangerous assumptions, ignores every real world engineering constraint, opaque and unfathomable, does not address critical issues, provides no insight into the problem, woefully unsuited to quantitative feedback synthesis".

There have always been obstacles to the provision of optimum handling. Thomas [1991] refers to the test pilot's "daily fight" for the best handling qualities. It is not always self-evident to a program manager or cost accountant that these are worth paying for. Penrose [in Whiteman, 1993] quotes the battles lost 50 years ago at the Westland Aircraft company that later cost lives, against a brilliant Chief Designer who knew little about piloting and failed to listen to those who did. Lessons learned about the handling qualities development process in recent years are quoted in Anon [1991], listing many major failures in flight control design that have occurred. It concludes that a successful process should involve "flying qualities engineers, pilots, designers, control engineers, flight test engineers, specialists on aerodynamics, actuation, computer hardware, software, system architecture, avionics, human factors, subsystems, structural dynamics etc....program managers and accountants".

In the growing domination of the scene by computing and software, it became more necessary than ever for handling qualities to be understood in a physically meaningful way. It has almost always been the case that "new" handling problems could have been avoided by a better understanding of both good and bad handling qualities from the past conventional aircraft. This was nearly impossible in the early decades of flight because of the mutual incomprehension between the disciplines of theory and practice.
1.3 Problem statement and motivation of the methods used

Handling qualities are not capable of an exact proof against precise measures, mathematical or otherwise. Hypotheses and theories can only be tested ultimately by their repeatedly successful use in the design of flight control systems. Pilots are not machines, and they tend to display some variability of their own and relative to other pilots' opinions. All that can be said is that there is a high statistical probability that some qualities will be found satisfactory to most pilots. At the other end of the scale, a quality which could lead to loss of control might not reveal itself until after thousands of flights over many years. Both extremes need to be understood.

The author\(^1\) commenced some 20 years ago to develop a fuller understanding of the meaning of handling qualities and a methodology for its application to fly by wire control system design. This was necessitated in the first instance by a Panavia Tornado accident discussed in Gibson [1978], but was also driven by widespread handling qualities problems in the majority of new military aircraft of the period. It is to be noted that such problems have made their appearance in the civil field with the advent of commercial aircraft with fly by wire flight controls.

Handling qualities have traditionally been quantified by mode parameters such as the natural frequency or damping of a short period motion, and so on. This often led to the assumption that handling qualities could be defined simply as a list of numbers or "Greek letters" (the symbols used to identify such parameters), readily obtained by computer batch processing in great quantity, and subjected to a simple pass or fail test. Unfortunately, it is not so simple.

When the pilot moves the stick, something happens. The author demonstrated that the essence of that something can be captured in a non-parametric and graphical "descriptive language" that brings to life the actual physical behaviour of the aircraft. This behaviour can be quantified and measured directly in time and frequency domain calculations. The approach is well justified for practical reasons alone. Moreover it does not interfere with a model-based approach in which all the model parameters are known, since the non-parametric frequency or time domain descriptions can be extracted from that. Examination of experimental results published from the 1950s onwards revealed a rich source of hitherto untapped data. The author was able to identify many aspects of handling qualities in previous research which had not been recognised or understood, and to show that some earlier apparently contradictory results could be reconciled.

Figure 1-1  Basic longitudinal response parameters, showing positive directions of pitch attitude ($\theta$), flight path angle ($\gamma$), angle of attack ($\alpha$), elevator angle ($\delta_e$) and angular pitching velocity about centre of gravity ($q$).

\(^1\) In this thesis, "the author" always refers to the present writer. Other authors are referred to by name.
Figure 1-1 illustrates the basic longitudinal response parameters controlled by a pilot. All pilots use visual attitude references for their primary control cues. Their opinion is sometimes based entirely on the attitude characteristics and in the limit is always influenced by them even where flight path control is the primary task. A similar non-parametric descriptive technique can be applied to the displacement, rate and acceleration components of attitude response.

The following simple examples give a flavour of the method. A pilot controls the angle of attack directly by adjustment of the elevator control, with corresponding changes in the normal acceleration which modifies the flight path. When precise transient control of the flight path is necessary, for example in the landing flare, the rapidity with which a steady rate of change of flight path angle is achieved after the application of a pitch control increment is important to the pilot. The flight path time response is determined by the pitch short period mode, to which limits are set in formal handling specifications in terms of its frequency and damping, though the distance covered in the transition depends on the speed. Such a specification may be inappropriate in modern digital flight control systems with an increased number of mode parameters typically found in their pitch response equations. The conventional parameters of short period frequency and damping can be transformed into the displacement in time between control application and the effective start of the steady flight path angle rotation, Figure 1-2. The resulting response description can be applied to any flight path response regardless of its mode complexity.

The author also found it possible to identify several specific characteristics of the response qualities that are associated with PIO. These are considered separately from the characteristics associated with the basic handling qualities, being in effect parasitic additions to normal behaviour. These might be either linear or non-linear characteristics, but in any case they can be described in terms of their effect on the non-parametric response "shapes" in time and frequency domains. Figure 1-3 sketches two versions of the pitch attitude frequency response for the landing case of the Northrop YF-17 Lightweight Fighter (the forerunner of the McDonnell-Douglas F/A-18). During in-flight simulation [Hall et al, 1975], Version 1 could not be landed because of unstoppable PIO, but the modified Version 2 had excellent handling.

The difference between these versions is not at first sight particularly obvious unless the message delivered by their shapes is understood. Their qualities can be identified by the author's methods with little more than a visual glance at the plots, which contain no parametric information whatever. The methods will be explained in this Thesis.

1.4 Problems addressed and scope of the thesis

The important attitude characteristics had been defined only to a very limited degree in previously available conventional criteria or specifications. The author's methods, coupled with the flexibility offered by digital computing power, opened up wide possibilities to design optimum handling qualities for different tasks. With continuous assessment against available data from other aircraft and experimental research programmes from the USA, the evolving results of this work were applied to the design of several fly by wire aircraft, the Panavia Tornado, British Aerospace Jaguar FBW, EAP, VAAC Harrier, and the four-nation Eurofighter 2000, with outstanding success. Some of the basic design criteria which were developed from this work became widely known as the "Gibson criteria". They appear in the USA military aircraft handling qualities specifications [Anon, 1987] as additional background material for consideration, and essential parts of the handling qualities specifications for the Eurofighter 2000 were based on them. The work resulted in the award to the author of the 1991 Royal Aeronautical Society Bronze Medal for innovations in the handling qualities field.
This Thesis describes the main elements of the author's work. Much of it appears in Gibson [1995c] as student notes for Delft University of Technology. This has been considerably expanded here, with some newer material added from Gibson [1995a, 1995b] and elsewhere. The discussion centres almost entirely on the pitch axis, because this has caused most problems. Less commonly, essentially identical problems have arisen in the roll axis, A few examples from the roll axis are included for a more complete picture.

Chapter 2 provides an overview of the early years of aviation and the slow development of any form of specified handling qualities. This was due both to the inability to connect the extensive theoretical knowledge of the time with the needs and perceptions of pilots, and to the human ability to control aircraft of quite remarkably bad handling qualities. Much experimentation in the 1950s onwards provided an extensive database of understanding and resulted in the comprehensive series of U.S. military handling specifications.

Chapter 3 presents the basic linear equations of motion and their simplifications used in handling qualities analysis.

Chapter 4 discusses the general behaviour of aircraft with several forms of manoeuvre demand flight control system. In important respects this is different from conventional behaviour and it is essential for the controls designer to appreciate the significance of this.

Chapter 5 discusses the human pilot as a crucial element in the overall control system. Some of the basic assumptions made about pilot response dynamics and their interaction with aircraft dynamics and control tasks are outlined.

Chapter 6 discusses how aircraft design parameters and flight control system hardware have affected handling qualities in the past and the different effects to be found in fly by wire aircraft.

Chapter 7 discusses how the author's methodology for the description and analysis of handling qualities was derived from studies of past data. The resulting enhancement in the description and understanding of overall handling qualities is demonstrated. Consideration is given to a number of other recent criteria and their contribution. The characteristic features of conventional handling that lead to good pilot opinion and task performance are discussed.

Chapter 8 discusses the nature of essentially linear high order pitch attitude PIO phenomena introduced by improper control law design, and the response characteristics that can lead to PIO. Response limits are presented that can eliminate PIO by design.

Chapter 9 discusses the nature of non-linear PIO phenomena introduced by control system saturation effects, motion-induced biomechanical PIO, and non-attitude related PIO. Appropriate models to describe the actions of the pilot are discussed.

Chapter 10 describes a process for optimising handling qualities by identifying optimum shapes and limits of response characteristics. The absolutely crucial importance of investigating the design for PIO-prone features and eliminating them is emphasised. Methods of achieving such aims are discussed and several criteria to this end are presented.

Chapter 11 contains concluding remarks and recommendations.
1.5 Summary

The serious handling problems discovered in most of the new generations of fly by wire combat aircraft in the 1970's and later revealed a very incomplete understanding of handling qualities in the aviation industry. A new methodology has been developed by the author that builds on past knowledge to extract a wider understanding of the qualities that enable a pilot to perform tasks easily and efficiently. It defines these by a non-parametric graphical representation that can be applied to the design of complex control law structures to ensure satisfactory handling.

The highly subjective nature of pilot opinion and handling qualities prevents the analytical or mathematical proof of the methodology. Its proof lies in its successful application to a series of fly by wire combat aircraft designs with which the author has been directly associated and in the adoption of some elements of it in the designs of others.
Figure 1.2 Illumination of response by non-parametric description

Notes:
1. Descriptive language: Time displacement \( f_y \) is universally applicable to any order of response.
2. Conventional parameters ("Greek letters"): Flight path response 
   \( \approx f_n(\omega, \zeta) \) (the short period frequency and damping)
3. The physical nature of the response is not visually apparent from the symbols (applicable only to low order response).

Figure 1.3 Frequency response qualities illustrated by non-parametric shape
Sketch derived from [Hall et al., 1975]
Chapter 2
Early handling qualities theory and practice. The beginnings of methodical research

2.1 Introduction

"Progress, far from consisting in change, depends on retentiveness ... Those who cannot remember the past are condemned to fulfil it" [Santayana, 1905]. The quotation is particularly apt in the study of aircraft handling qualities. With the centenary of the Wrights' powered flight and the bicentenary of Cayley's ground-breaking perception of the fundamental aeroplane configuration [Gibbs-Smith, 1962] less than a decade away, it is a sobering thought that the subject was so little understood that Berry [1982] could draw the conclusions noted in Chapter 1.

These conclusions were prompted by the handling qualities problems exhibited by a number of fly-by-wire aircraft, discussed in Klyde et al [1995] and Anon [1997] with many later examples. These aircraft were made possible only by the application of advanced technology in many fields that would have been inconceivable to the pioneers. The one unchanging factor was the human pilot, a being evolved over hundreds of thousands of years, whose needs formed a continuous thread throughout aviation history that was all too often misunderstood or unrecognised.

This chapter outlines the confused early concepts of control and handling qualities, the absence of any significant guidance to desirable qualities until the end of the fourth decade of flight, and the gathering pace of research from the 1950's onwards.

2.2 Early handling qualities theory and practice

2.2.1 Stability and control problems

Some basic theory of aircraft stability appeared very early. Cayley in 1804 [Gibbs-Smith, 1962], Pénaud [1872] and Zahm [1893] had all proposed the tailplane as a basic element of pitch balance, stability and control, though these ideas could not be completed until the aerodynamic centre was theoretically confirmed in the 1920's. The concept of lateral stability through the effects of wing dihedral was known to them. Bryan and Williams [1904] derived the longitudinal dynamical equations of gliding flight. Bryan [1911] extended this to the six degrees of freedom simultaneous differential equations of motion in a form recognisable today, identifying the long and short period oscillations in the pitch axis and the roll, spiral and Dutch roll modes in the lateral-directional axes. Lanchester [1908] derived the long period pitch motion from physical considerations and named it the phugoid. None of this theory was of much assistance to early designers, because the static element was not understood and the dynamic element was very difficult to apply. For example, Bryan assumed the presence of positive static stability as a prerequisite for dynamic stability, but could not define its physical source because the aerodynamic centre was unknown.

The Wrights are justifiably credited with developing the concept of good control [Walsh, 1976]. It could also be said that they were compelled to do so because the 1903 "Flyer" was almost unflyable [Jex et al, 1985]. Severe handling difficulties resulting from large pitch instability prevented their designs from achieving wide practical use. When they supplied an observation aircraft to the US Army Signal Corps in 1908, it was required to be capable of being steered easily in all directions without difficulty under perfect control and equilibrium. The Army test pilot reported it to be "about as stable as a bucking bronco" [Foulois, 1980]. The Wrights had not de-
liberately made their design unstable though they certainly recognised the instability and eventually its cause, but they could do little about it until they removed the foreplanes and installed a conventional tailplane at the rear in 1911. The design was completely out of date by then.

However, the Wrights succeeded brilliantly in lateral-directional control. They recognised from the very beginning that turning flight had to be performed by banking the aircraft. Discovering on the 1901 glider that roll control generated adverse yaw, they added a rear fin to the radically new 1902 glider to reduce sideslip when control was applied. Then they made the fin moveable and connected it to the roll control to apply further automatic resistance to adverse yaw. Finally they added a third pilot's control to move the rudder independently, always with the sole purpose of preventing sideslip and most certainly not to steer with. By 1905 they could turn in extremely small circles, possibly only a few tens of metres in diameter [Walsh, 1976].

The principle that they evolved of turning aircraft by banking with roll control (wing warping in their case) and co-ordinating with the rudder to prevent sideslip is exactly the method used to this day. The rapid evolution of their ideas and capabilities in lateral stability and control, based on a total flight experience measured in very few hours but thousands of flights, is a remarkable example of systematic problem-solving in a discipline that was entirely new and lacking in practical or theoretical assistance from any other source. In contrast, the Europeans had deliberately sought passive inherent stability, using the rudder to steer their aircraft in flat turns which could not always be contained within airfield boundaries. They were astounded by the demonstrations of the Wrights' aircraft in 1908 at Le Mans. The crowds were terrified at its steeply banked turns and thought it about to crash [Walsh, 1976].

It seems that the lesson about banking was only partly learned, probably because of the Wrights' obsessive secrecy about their methods. For many years afterwards the standard official method taught for turning, even in the USA, was by the primary use of rudder. It was taught that an inherently stable aircraft would take up by itself the correct bank angle to provide the necessary centripetal turning force [e.g. Loening, 1916], and that further adjustment need be applied by the ailerons only for balance to minimise slip or skid. This same fallacy was expounded to Gilbert [1970] in respect of an original Fokker D-VII operated in the USA in the 1960s.

A collection of early articles in Oppel [1986] provides a fascinating insight into the fallacies about flight mechanics and control technique of the early period up to about 1912. Much of this is summarised in Gibson [1996]. Post [1911] comments on the need to pitch down when turning a low powered aircraft due to the loss in speed because of the drag caused by the machine slewing sideways. Post comments that, whereas French types were "steered by the feet and balanced by the hands", American machines differed radically in that usually they were "steered by hand and balanced by shoulder or body movements". Post categorizes the latter as typified by the Curtiss, "copied more than any other by other builders", and considered it "perhaps the most natural to operate". It had a wheel, pushed fore and aft for pitch control, and turned to operate the rudder for steering. The ailerons (individual "little wings" mounted half way between the upper and lower wings) were operated by rocking the pivoted seat-back from side to side. The pilot's left foot operated the throttle and the right the engine cut-off.

Eccentric though the Curtiss arrangement might seem now, it was at least logically connected to the (erroneous) control concepts of the time. With the same concepts, European aircraft generally had pilot's inceptors (stick, wheel, pedals etc.) recognisable today. The Wrights, despite being the only pioneers to get the control concept correct, had the worst inceptors of all, especially in their side-by-side two-seat designs from 1908. These had three fore-and-aft sticks, the two
outer connected together to control the foreplane. The centre one controlled the wing warping, forward for right bank and aft for left bank, and the top part was articulated laterally for rudder control. This confused even the Wrights, leading to a serious test flight crash and causing Wilbur Wright to make many mistakes at Le Mans, one of which led to another accident. Instructors in Wright aircraft were usually able to fly only in their normal right-hand seat, since in the left hand seat the mental workload in using their untrained right hand for roll control was too great [Coombs, 1990].

By about 1914, however, the standard configuration of stick (or wheel) and rudder bar had become essentially universal, even if the use to which they were put was often partially based on ignorance of the optimum control techniques. The turning fallacies were apparently not entirely eliminated from many pilots’ consciousness even in the 1930s. The excellent explanation of the art of flying by Langewiesche [1944, re-published 1972] was prompted by his discovery early in his flying career in the 1930s that pilots’ words about flying and what they actually did were not in agreement. Despite such words of wisdom, however, the power of aviation folklore to survive demolition by the growth of knowledge is extraordinary. In 1992 the author was told by a veteran instructor to steer a Cessna 152 down the final approach with rudder alone.

2.2.2 The absence of task-related handling qualities concepts

The relationship between task performance and handling qualities is well known today, but in the early days of aviation it was a remarkable feat to sustain flight at all, while stability and control were little understood. For example, although long flights were possible in the Wright aircraft, this was insufficient because most pilots found it hard to master, and it was already archaic before its handling qualities were developed enough to be useful. Aircraft performance developed rapidly during the First World War (1914-1918), but it was often a matter of luck if the designs had good handling, encompassing the whole range from good to bad. As discovered in later years, task performance was not always well correlated with handling qualities. At least theceptors (the control sticks or wheels and pedals) had evolved to satisfactory forms, however.

The handling peculiarities of many First World War aircraft would astound modern-day pilots. Some are discussed briefly in Gibson [1995c], and further in for example O’Gorman [1919], Gilbert [1970], Bowyer [1978], Harper and Cooper [1984], Loftin [1985], Reynolds [1993], Stinton [1996], Abzug and Larrabee [1997]. Two pilots in that war, Lewis [1936] and Yeates [1934], describe vividly what it was actually like.

Piloting techniques were naturally influenced by what would be seen today as serious aero-
dynamic and control deficiencies. For example, ailerons were inefficient, produced adverse yaw, and were hard to deflect at combat speeds due to lack of balancing. Directional stability was often low and sometimes zero. Consequently ailerons produced low roll rates in combat even when the rudder was used to prevent sideslip. High roll rate was achieved by stalled flick rolls, initiated by applying full up elevator and full rudder and centralising them when the desired manoeuvre was completed. Longitudinal instability was commonplace, and the gyroscopic effects of rotary engines could dominate the response at low airspeeds, with fatal consequences for many pilots of some types. Poorly flown loops could leave an aircraft trapped upside down with insufficient aft pitch control left to initiate a pitch-down to recover, or alternatively it might flick out into a spin.

There was no way to measure aircraft dynamic motions or to relate them to pilot needs. At a time when it was considered useless to open fire in a general melee at ranges of more than a few metres, aircraft performance and pilot tactics might well have seemed the only important mat-
ters. The "Fokker scourge" stories which dominated British newspaper reporting of the air war in 1915 arose from mounting Allied losses to the synchronised gun firing through the propeller disc in the Fokker monoplane fighter, rather than from its aerodynamic or handling qualities which were extremely primitive. Pilots who survived the expected life span in combat of only three weeks could become sufficiently expert to accept the deficiencies of their aircraft. The concept of handling qualities as a work load parameter to help or hinder the pilot in task performance was impossible in the circumstances.

The stability fallacy
Theoreticians did not understand flying, pursued inherent stability, and thought of handling qualities in terms of the long period phugoid and spiral modes which were typically poorly damped or unstable. Pilots, scarcely affected by these, fought in combat through the short period pitch, roll and yaw modes and had no knowledge of theory. There was no common ground or language between the groups.

The explanation for the widespread use of reduced directional stability was a theoretical concern about spiral instability, a tendency for the bank angle to diverge gently from level flight that could be minimised in this way. Although pilots thought that the consequent ready ability to skid their aircraft to point its guns at an enemy a few metres away was a good thing, it was the cause of great numbers of accidents at low speeds where the combination of inadvertent or deliberate sideslip with the typically abrupt stall led to spins. It was to no avail in any case since spiral instability is not a significant problem for pilots.

The principal fallacy that has survived to this day is that longitudinal stability is the enemy of manoeuvrability, which may be the result of confusion over the role of stability in handling qualities. The most frequently cited "proof" was the example of the RAF BE.2C reconnaissance aircraft [Loftin, 1985], which was deliberately designed for stability and could be flown hands off for long periods. It was shot down in large numbers, which was attributed to lack of manoeuvrability due to its stability. Its handling was actually entirely conventional and straightforward. However, its armament was ineffective and its speed and rate of climb were a small fraction of a typical fighter, a lack of operational performance that was fatal.

The success of the Wrights in demonstrating the power of control over an unstable airframe contrasting with the European attempts at "inherent stability" with the consequent poor control qualities, the combat success of some famous but unstable fighters in the hands of pilots who survived long enough to master them, and the disastrous experiences with the stable BE.2C may have obscured the real lesson. The combination of high performance, good short period stability and powerful control was not only feasible but resulted in extremely effective combat aircraft in which desperately inexperienced pilots could concentrate all their attention on the enemy. Some First World War examples are the Sopwith Pup, the Sopwith Triplane and the Fokker D.VII.

2.3 The beginnings of methodical research

2.3.1 Stability margins
The measurement of aircraft qualities began slowly. The UK National Physical Laboratory was testing wind tunnel models of whole aircraft by 1912. Figure 2-1 shows the conventional lift, drag and pitching moment of a Bleriot model. The effects of sideslip on all of the six axes were also measured in these tests. The US MIT built a tunnel in 1915 to the NPL design. Figure 2-2 shows early test results of the lift and pitching moment of a Clark aircraft. These results contain a trap into which the unwary continue to fall up to the present day. They appear to show that
static stability as measured by the slope of the pitching moment curve against angle of attack depends on the tail setting relative to the wing. For these tests the centre of gravity (CG) was altered so that the trim angle of attack was maintained at about 2 degrees for each tail setting. Of course, the change in the CG position determined the stability while the overall balance determined the tail angle required for trim [Perkins, 1970].

The earliest measurements of stability and control appear to have been made at NACA Langley in 1919. Figure 2-3 shows control position and stick force plotted against airspeed for the Curtiss Jenny trainer and the De Havilland DH.4 light bomber. The Jenny was slightly unstable except at the stall, while the DH.4 was clearly rather stable at all speeds. These differences were recognised but their significance seemed to be uncertain [Perkins, 1970].

Elementary stability theory development
The pace of aircraft development slowed almost to a stop at the end of the First World War. A major advance in the field of theoretical aerodynamics took place shortly after, however, which led to the first development of static stability theory. Measurements of aerofoil characteristics in wind tunnels had long been known to show that the lift force appeared to act at the fixed quarter chord point when the angle of attack was altered, because the pitching moment at this point was independent of the lift. In the 1920's, theory showed that all lifting aerofoils have the property that the incremental lift due to angle of attack acts at 25% of the chord, e.g. Glaeurt [1926]. Lift due to section camber and proportional to its depth was found to act somewhat approximately at 50% of the chord and to be independent of the angle of attack. It creates the pure couple at zero total lift known as the zero lift pitching moment2.

Shortly after Glaeurt's work on this, work at RAE Farnborough began on the development of static stability theory, publishing the expression equating the slope of the wing pitching moment versus lift coefficient to its stability margin [Gates, 1927]. This is the non-dimensional distance between the CG and the "aerodynamic centre" (AC), the name he chose for the 25% chord point. The theory was developed in the next few years to include the effects of the tailplane and wing downwash field to determine the overall AC of the whole aircraft. This is known as the neutral point (NP), the CG position at which the pitch control angle necessary to hold different speeds in non-maneuvering flight is constant. Work on the "diving pull-out" defined the manoeuvre point (MP), the CG position at which the pitch control angle to hold different normal accelerations at fixed airspeed is constant. The MP is aft of the NP by a distance proportional to the damping of pitch rotation due to the tail.

From this evolved the basic relationship of control displacement with speed change, related to the CG margin or the distance between the CG and NP, and with manoeuvre at constant speed, related to the manoeuvre margin or the distance between the CG and MP. Another early reference to the subject of elevator trim curves is van der Maas [1929], with the concept of "control stable" and "control unstable" aircraft defined by the direction of the stick deflection in speed changes and presenting the influence on stability of the CG position, propeller, etc. Stability is further differentiated by stick free characteristics associated with stick forces. The stick free NP and MP, which determine the amount of stick force necessary to hold a change of speed or of normal acceleration respectively, are located according to the pitch control hinge moments in response to control deflection and angle of attack changes.

2 These two lift components can be combined into a single lift force acting at the centre of pressure or CP, which moves chordwise as the lift varies in proportion to the zero lift moment. The CP concept, known well over a century ago, is responsible for a remarkable amount of confusion about pitch stability, for despite appearances its movements have no effect on the NP whatever.
These initial elementary results remain sufficient to perform all the pitch stability and control calculations needed for simple low speed aircraft. By determining the stick fixed NP and locating the CG at a minimum distance ahead of it, typically 5% to 10% of the Mean Aerodynamic Chord (MAC) for example, both speed and manoeuvre stability would usually be sufficient to enable adequate handling to be obtained. However, development to enhance stick free stability became necessary, leading to better design methods to obtain the desired control hinge moment response to angle of attack and control deflection. This was driven by the needs of many World War 2 aircraft types, often lacking in static stability due to inadequate design or to rearward CG shifts with the inevitable increase in weapon loads. Improvement of their stick fixed stability would have required a new and larger tail, costly in time. Substantial improvements to the "hands off" stick free stability could be obtained by relatively small changes to the elevator balance, and this in any case came to be recognised as more important.

One major stimulus for this work was the need to solve the problems of the Supermarine Spitfire, which suffered from a serious lack of longitudinal stability throughout World War 2. In its original form it had a CG range of 2.7% chord, adequate then for its limited fuel and weapon load stored on the CG. Eventually its weight and power doubled, its CG range increasing to 11%. It was often operated in an unstable condition that sometimes led to fatalities. The Mk.21 was finally judged to be quite unacceptable, and for the Mk.22 the tail areas were increased by 27% (yaw stability had also been less than desirable). To some surprise, apparently conditioned by the old fallacy about the incompatibility of stability and manoeuvrability, the handling was found to be superb [Quill, 1983].

The beneficial effect of good short period stability on combat effectiveness was demonstrated by aircraft such as the Hawker Hurricane, Typhoon and Tempest, all considered to be much better gun platforms than the Spitfire (whose nose would wander as a result of engine and propeller gyroscopic torques). Opinions on this continued to vary, but it is noteworthy that two leading test pilots with more experience than most in both World War 2 combat and test flying (Yeager in the USA and Beamont in the UK) were firmly in favour of very positive stability. It was a further 20 years before this view was given formal support in the minimum short period frequency requirements in the military handling qualities specifications [Anon, 1969].

Advanced stability theory
As aircraft speeds increased greatly, it became necessary to account for two effects of this on static stability. Aeroelastic distortion due to air loads introduced moment changes influencing the trim characteristics as a function of speed. As the wing AC moved rearwards due to the effects of compressibility, the MP would migrate aft causing an increase in manoeuvre margin. At the same time the NP would migrate forwards causing a reduction or even instability of the static margin, related to the "speed stability" through angle of attack control expressed as control angle and force versus speed. An aircraft could have negative static stability and positive manoeuvre stability simultaneously.

In piston engine fighters, large compressibility effects were unlikely except in full power vertical dives from high altitude. Recovery from these at first seemed impossible and the explanation was believed to be control reversal. Some rather heroic flight tests proved this was not so [Abzug and Larrabee, 1987]. The nose down trim shift, increase in angle of attack stability, and reduction in elevator control power which appeared to lock the aircraft into the dive would diminish as lower altitudes and decreasing Mach numbers were reached, after which a normal recovery was possible. On the Lockheed P-38 Lightning, forward under-wing flaps were added to provide a recovery moment.

Early handling qualities theory and practice 13
Jet powered aircraft could cruise at conditions where negative static margins were likely. It was predicted that the four-jet Handley Page Victor high altitude bomber aircraft would have a manoeuvre margin of $0.2\ell$ and a static margin of $-0.17\ell$ [Anon, 1951]. It was calculated that the time to double amplitude would be about 15 seconds. Flight tests found no difficulty in flying a de Havilland Vampire jet fighter with a manoeuvre margin of $0.05$ to $0.1\ell$ and a static margin of $-0.18$ to $-0.8\ell$. More concentration was needed by the pilot but the slow airspeed divergence was not difficult to suppress.

The autopilots of the day could also maintain stability to relieve the pilot for long cruising periods. The design of supersonic fighters usually postpones significant static instability to the transonic region where cruise is not performed. It can be sufficiently mild in effect that pilots do not notice it during a transition to higher Mach numbers, as manoeuvring control remains normal.

### 2.3.2 Dynamic stability

During much of the period up to the 1930's, guidance to good handling would be mainly qualitative, with simple rules such as limits on the steady values of stick force per unit change of speed. Adjustments would be made if possible during test flying until the pilots were satisfied, but this was largely a matter of individual opinion. In the 1930's it began to be evident at some research establishments that the dynamic behaviour of aircraft had a rather different influence on pilot opinion than had been supposed from the theory. As noted earlier, this disparity had always existed, but was not noticed at a time when the disciplines did not understand each other and flight records of aircraft motions were not possible.

While longitudinal dynamic stability was obviously desirable, this was generally assumed to refer only to the phugoid which was usually poorly damped and sometimes slowly divergent. Soulé [1937] measured the dynamic and static stability of eight aircraft for comparison with pilot opinion. These measurements were no more than observations of whether the phugoid was stable or unstable, and of how static "stiffness" was reflected in elevator force and movement, any tendency to pitch in rough air, and so on. It was found that pilot opinion had no correlation with the phugoid motion. The short period dynamic motion, which would naturally have been heavily damped, does not seem to have been observed directly other than through the turbulence response. It was thought of only as the "rapid incidence adjustment" of little other significance to the pilot.

In the lateral-directional axes, it was assumed that spiral stability meant a considerable self-leveling ability. As noted by Koppen [1940], an early USA CAA bulletin gave a formal test for lateral stability as follows:- "With the rudder held in neutral, the airplane is rolled to a fairly pronounced bank angle and the control stick immediately released. If the airplane recovers to level flight, it is laterally stable." Koppen states that few aircraft certified airworthy by the CAA would be laterally stable by this definition. Thomas [1991] says that the later FAA method of applying crossed rudder and aileron to maintain a steady heading, although accepted as a means for demonstrating compliance with the FAR, is not how pilots assess lateral stability in practice either, and that they know if an aircraft has sufficient stability by flying normally in turbulence.

Detailed measurements of handling qualities started in the 1940s with new photographic recording instrumentation. This was still in use in the 1950s, aided by additional analogue pen recordings of some parameters. Both methods required considerable manual effort to transcribe and plot the data. By obtaining flight data from special manoeuvres and comparing these with pilot opinions, it became possible to formulate design requirements. Tests on many types led to the first military specifications for flying qualities based on the proposals of Gilruth [1943].

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list of the parameters considered is shown in Table 2-2. These would be completely sufficient for application to any modern design of low speed aircraft. While the static characteristics were included as before, manoeuvring characteristics were introduced, associated with the short period response modes, dynamic behaviour in stalls etc.

By 1948 experience of how handling qualities could influence task performance had been greatly enhanced by Second World War combat in the USA, the UK and Germany. The earlier Air Force and Navy specifications in the USA were replaced by a unified version [Anon, 1954]. It was based mostly on the data extracted from the subsonic piston engine era. Consequently, 50 years after the first manned flights, a comprehensive set of criteria and specifications was at last available, only to be overtaken by the start of a rapid expansion in performance and aerodynamics that introduced a whole new set of handling problems. These resulted from increased wing sweep, wing loading, altitudes, speeds and Mach number.

2.4 The experimental era

The 1950s and 1960s saw an intensive effort to study handling qualities experimentally, rather than by studies of existing data. The use of ground based flight simulators became routine. These have been developed continuously to the present day, and play a major part in flight control development and clearance for flight of most new designs. Motion was introduced at an early stage to reduce the deficiencies. Much development was necessary to prevent degraded results, discussed variously in Anon [1981, 1983, 1985a, 1985b]. To a lesser extent, but with an impact far beyond its quantity, in-flight simulation in variable stability aircraft at last enabled research to be conducted in a pilot’s true environment. In the air, all the cues through which the handling is ultimately evaluated are available. The need to extrapolate back to the real world is a deficiency of ground based devices that requires constant vigilance and understanding to avoid misinterpretation. Most of the wide range of variable stability aircraft used up to the present time are listed in Table 2-3.

Cowley and Skan [1930] had noted the difficulty of analysing piloted control mathematically, due to the ability of pilots to adapt to extremely wide variations in aircraft dynamics. This adaptability had been amply demonstrated in the First World War, and was a serious impediment to understanding the importance of handling qualities. A profoundly influential development in the field of theoretical pilot modelling was begun by Tustin [1947], and was developed by McRuer [1960] and many others. This enabled systems analysis methods to be applied to the understanding of the closed loop behaviour of the pilot-aircraft combination. It became possible to relate the dynamic qualities of an aircraft response to the performance of tasks by the pilot.

The further measure of handling qualities was made possible through the use of pilot opinion ratings. These were developed by Cooper and Harper [1969] into the rating scale in Table 2-1, universally used for this purpose ever since. Application of the Cooper/Harper scale is a process that must be meticulously applied if results are not to be misleading, however. In the author’s experience, pilot comment data has been even more important because they identify precisely the response characteristics that shape the pilot’s opinion. Boothe et al [1974] discuss the importance of pilot comment, which revealed that pilots primary concern in tracking manoeuvres was precision control of pitch attitude, while control of normal acceleration was of secondary importance. This could not have been found from numerical opinion ratings alone.

With the large quantity of new research data available, a new version of the US handling qualities specification was published [Anon, 1969]. This incorporated pilot-in-the-loop research re-
sults, and introduced the concept of handling quality "levels" that are related to opinion ratings, aircraft states (including system failures), the kind of task being undertaken, aircraft performance envelopes, and flight control augmentation effects. A major enhancement was its companion user guide document [Chalk et al, 1969], which provided the basic data and evidence upon which each requirement was based. Its several hundred pages remain a reference work of the utmost value.

A further update, [Anon, 1980], included criteria accounting for the "high order effects" which had appeared in many fly-by-wire control systems. It did not perform this intention very successfully, as handling problems continued to occur. Finally, after more research to analyse the problems and the publication of many new criteria, completely new design standards [Anon, 1987] were published. These requirements were based primarily on the earlier ones, but many alternative criteria were given to offer a wider insight, and these could be used also if the manufacturer and customer jointly agreed.

Unfortunately, the formal assumptions about the underlying response characteristics remain largely of the classical form. No significant approval is offered for new "short period" dynamics which are made possible by computer controlled systems and which can improve task performance. The way in which these can also lead to serious control problems is inadequately explored, and pilot induced oscillations continue to present unpleasant surprises. Apart from limits on allowable flight path instability due to operation on the back side of the drag curve, a classical performance problem which has little to do with flight control design, there is no guidance on problems associated with speed dependent or long period modes which depart from the classical phugoid characteristics. Such changes can be beneficial or harmful.

2.5 Summary

This chapter reviews the past from the beginning of aviation, with the aim of explaining the lack of understanding of good handling qualities so widely exposed in many fly by wire projects of the past two decades, and shows that this lack had always existed.

In the beginning, handling could be quantified by little more than qualitative comments. It was a sufficiently hard goal merely to obtain satisfactory flight performance. Despite the early solution of the equations of motion, these were extremely difficult to use and there were few practical means by which good handling could be designed a priori. While aircraft dynamics remained of classical or traditional form, determined almost solely by the natural laws of aerodynamic and inertial forces, it was eventually possible (after some four decades) to establish a number of elementary markers of desired handling.

Although experimental and theoretical studies from the 1950s onwards greatly enhanced the understanding of handling qualities, this was still directed largely to the identification of satisfactory parametric qualities of classical aircraft dynamics. These could not be readily applied to the design of the more recent fly by wire aircraft whose dynamics are dominated by the flight control system. This Thesis describes the author's approach to a better understanding of the reason for good handling by "remembering the past", to find a non-parametric design methodology that can be applied to the modern flight control system.
### Table 2-1: The Cooper-Harper Pilot Opinion Rating Scale

<table>
<thead>
<tr>
<th>Aircraft Characteristics</th>
<th>Demands on the Pilot in Selected Task or Required Operation</th>
<th>Pilot Rating</th>
</tr>
</thead>
<tbody>
<tr>
<td>Excellent</td>
<td>Pilot compensation not a factor for desired performance</td>
<td>1</td>
</tr>
<tr>
<td>Highly desirable</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Good</td>
<td>Pilot compensation not a factor for desired performance</td>
<td>2</td>
</tr>
<tr>
<td>Negligible deficiencies</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Fair - some mildly unpleasant deficiencies</td>
<td>Minimal pilot compensation required for desired performance</td>
<td>3</td>
</tr>
<tr>
<td>Minor but annoying deficiencies</td>
<td>Desired performance requires moderate pilot compensation</td>
<td>4</td>
</tr>
<tr>
<td>Moderately objectionable deficiencies</td>
<td>Adequate performance requires considerable pilot compensation</td>
<td>5</td>
</tr>
<tr>
<td>Very objectionable but tolerable deficiencies</td>
<td>Adequate performance requires extensive pilot compensation</td>
<td>6</td>
</tr>
<tr>
<td>Major deficiencies</td>
<td>Adequate performance not attainable with maximum tolerable pilot compensation; Controlability not in question</td>
<td>7</td>
</tr>
<tr>
<td>Major deficiencies</td>
<td>Considerable pilot compensation is required for control</td>
<td>8</td>
</tr>
<tr>
<td>Major deficiencies</td>
<td>Intense pilot compensation is required to retain control</td>
<td>9</td>
</tr>
<tr>
<td>Improvement mandatory</td>
<td>Control will be lost during some portion of required operation</td>
<td>10</td>
</tr>
</tbody>
</table>

The flowchart illustrates the decision-making process for determining the adequacy of a selected task or required operation based on aircraft characteristics and pilot opinion ratings. The process involves multiple decision points, including whether the task is satisfactory without improvement, whether the performance is adequate with tolerable workload, and whether the performance is controllable.
REQUIREMENTS FOR SATISFACTORY FLYING QUALITIES OF AIRCRAFT
(from Gilruth, 1943)

A. Requirements for longitudinal stability and control:
   (1) Elevator control in take-off
   (2) Elevator control in steady flight
   (3) Longitudinal trimming device
   (4) Elevator control in accelerated flight
   (5) Uncontrolled longitudinal motion
   (6) Limits of trim due to power and flaps
   (7) Elevator control in landing

B. Requirements for lateral stability and control:
   (1) Aileron control characteristics
   (2) Yaw due to ailerons
   (3) Rudder and aileron trim devices
   (4) Limits of rolling moment due to sideslip
   (5) Rudder control characteristics
   (6) Yawing moment due to sideslip
   (7) Crosswind force characteristics
   (8) Pitching moment due to sideslip
   (9) Uncontrolled lateral and directional motion.

C. Stalling characteristics:
   (1) Pitching-moment characteristics
   (2) Rolling- and yawing-moment characteristics
   (3) Control forces
   (4) Recovery

Table 2-2 The first comprehensive handling qualities requirements

<table>
<thead>
<tr>
<th>Aircraft</th>
<th>Handling Qualities</th>
</tr>
</thead>
<tbody>
<tr>
<td>F6F</td>
<td>MIRAGE 3</td>
</tr>
<tr>
<td>B26</td>
<td>HFB 320</td>
</tr>
<tr>
<td>YF-86D</td>
<td>P2V-7</td>
</tr>
<tr>
<td>TWIN-BEECH</td>
<td>VAAC</td>
</tr>
<tr>
<td>F100</td>
<td>F4U</td>
</tr>
<tr>
<td>NAVION (2)</td>
<td>F-94A</td>
</tr>
<tr>
<td>LEARJET (2)</td>
<td>F-86E (2)</td>
</tr>
<tr>
<td>707-80</td>
<td>NT-33</td>
</tr>
<tr>
<td>X-22A</td>
<td>NF-8D</td>
</tr>
<tr>
<td>JET RANGER</td>
<td>F-16 VISTA</td>
</tr>
<tr>
<td>C-131H TIFS</td>
<td>JETSTAR</td>
</tr>
<tr>
<td></td>
<td>X-14</td>
</tr>
<tr>
<td></td>
<td>BELL 47G</td>
</tr>
<tr>
<td></td>
<td>CH-47</td>
</tr>
<tr>
<td></td>
<td>BEAGLE 206</td>
</tr>
<tr>
<td></td>
<td>ATTAS</td>
</tr>
<tr>
<td></td>
<td>QUEEN AIR</td>
</tr>
<tr>
<td></td>
<td>ASTRA HAWK</td>
</tr>
</tbody>
</table>

Table 2-3 Variable stability research aircraft

18  Chapter 2
Figure 2-1 Lift $L$, drag $D$ and pitching moment $M$ of a Bleriot model

Figure 2-2 Lift and pitching moment (at varying CG positions) of a Clark model

Figure 2-3 Control position and stick force of Jenny and DH-4

Material extracted from [Perkins, 1970]
Chapter 3
Basic elements of handling qualities 1: The conventional aircraft dynamics

3.1 Introduction

Before consideration of the dynamical descriptions of aircraft response by means of which their handling qualities can be analysed, it is interesting to examine quotations from important early writings on the subject, one by a practical author and one by a theoretical specialist.

Diehl [1936] noted the advances in the subject after Bryan's remarkable work [1911] at a time when there were few pilots and even fewer successful aeroplane designs. Bairstow et al [1913] derived the approximate stability biquadratic solution and applied the method to the design of the BE.2C in 1913 (see §2.2.1 and Loftin, 1985). Glauert [1927] simplified the equations by introducing non-dimensional derivatives, and Gates [e.g. 1927] began to develop stability theory further. Despite that, Diehl noted the theoretical difficulties of calculating dynamic stability remained - "an academic exercise involving a tremendous amount of drudgery ... It is not surprising, therefore, to find that practically all aeronautical engineers have ignored dynamic stability as a design factor in the past." His book devotes 65 pages to static stability and control power. Despite recommending the study of dynamic stability for future more advanced designs, he gives it only a brief treatment.

Gates' 1927 paper explains how the unequal distribution of the total damping results in very low damping of the speed oscillation, or phugoid, but extremely high damping of the angle of attack oscillation, the "quick oscillation". The latter "is easily treated mathematically but is practically of small importance because of its rapid rate of decay, but the phugoid motion is of great importance because it is essentially of poor damping and is for that reason rather intractable to mathematical treatment". In fact, "Longitudinal theory in its classical form is so encumbered with a multiplicity of variables and of complicated algebraic expressions that aeroplane designers have been deterred from making it part of routine design. It is customary to provide some small degree of weathervane stability, and this simple policy has in general been justified by the results. But it is by no means clear that there may not be some conditions of flight in which dangerous instabilities may develop even though the slope of the static pitching moment curve is thoroughly stable". Gates provided simplified charts for frequency and damping, but only for the phugoid.

The last quotation from Gates appears to echo Bryan's belief that aircraft might go out of control in a climb because of phugoid instability when a critical pitch angle was exceeded. These authors all believed that the important elements of dynamic stability resided only in the phugoid oscillation. This academic misunderstanding of what constituted good handling from the pilots' point of view was universal.

The discovery in practical flight tests by Soulé [1937] (see also §2.3.2) that the phugoid dynamics are usually of singularly small interest or importance to pilots did not appear to be noticed. However, it has long been recognised in the trivial or non-existent requirements placed on the phugoid mode in handling qualities specifications. In strong contrast, the dynamics of the short period modes are of major importance. It is their deficiencies, or those of their later equivalent in fly by wire aircraft, that have caused many difficulties and even accidents. The physical basis of the short period response of conventional aircraft has to be understood, and then quantified in ways that can be applied to the potentially very different fly by wire response modes.
3.2 Factors influencing pitch handling qualities analyses

To describe the motions of an aircraft completely requires the solution of six simultaneous nonlinear force and moment equations of such complexity that it is feasible only by computer. The reduction of this problem to manageable proportions for the study of generic handling qualities, divorced from direct representation of a particular aircraft and capable of simple physical interpretation, requires very considerable simplification. Bryan showed that the six linear equations were separable into two independent groups of three symmetric or longitudinal equations and three asymmetric or lateral equations. They revealed the five modes of aircraft motion, the phugoid and short period longitudinal oscillations, and the roll, spiral and dutch roll lateral modes.

Development of new analytical tools for the design of feedback amplifiers in the 1940s led to their adoption for aircraft flight control system design. Bollay [1951] summarised the importance of the Laplace transformation, frequency response techniques, the Nyquist stability criterion, root locus methods and the Nichols chart. The aircraft modes of motion were expressed analytically by transfer functions relating some output variable, such as pitch rate, to an input variable, such as elevator deflection. Their poles and zeros can be given as either approximate symbolic expressions in terms of the stability derivatives, or more simply in numerical expressions such as the mode frequency and damping.

Many references exist on this subject, notably by Ashkenas et al. [1958, 1960, 1962], McRuer et al [1960, 1973], Mooij [1984], Gerlach [1983], and textbooks e.g. by Perkins and Hage [1949], Etkin [1972], Roskam [1995], Hancock [1995], and Cook [1997]. (Mooij noted that the approximation process for solution of the equations of motion was initiated by Jones [1935].) The symbolic derivative forms are complex, and as will be seen in later chapters have no relevance to the thrust of this thesis and are not given here. A full treatment can be found in Cook [1997].

3.2.1 The classic small perturbation pitch transfer function

Transfer functions in the Laplace form are used throughout this Thesis. Frequency response functions derive from them by replacing the complex variable s by jω. The transfer functions in this Chapter are introduced only to illustrate generic response "shapes" such as were noted in Chapter 1. They play no other part in the author's design methodology.

Of all the transfer functions possible, the most significant is that linking the pitch attitude response to the elevator control inputs. This is because pilot control of attitude is the core inner loop through which the majority of flight tasks is performed. The classical attitude function is:

$$\frac{\theta(s)}{\eta(s)} = \frac{K(1 + T_{\theta 1}s)(1 + T_{\theta 2}s)}{(1 + \frac{2\xi_p}{\omega_p}s + \frac{s^2}{\omega_p^2})(1 + \frac{2\xi_{sp}}{\omega_{sp}}s + \frac{s^2}{\omega_{sp}^2})}$$  \hspace{1cm} (3.1)

(Phugoid oscillation)  \hspace{1cm} (Short period oscillation)

where: $\theta$ is the pitch angle,
$\eta$ is the elevator control deflection,
$K$ is the attitude gain per unit elevator,
$T_{\theta 1}, T_{\theta 2}$ are the phugoid and short period numerator time constants relating the attitude response to elevator input,
$\xi_p, \xi_{sp}$ are the damping ratios of the phugoid and short period oscillations,
$\omega_p, \omega_{sp}$ are the natural frequencies of the phugoid and short period oscillations,
s is the Laplace operator.

The small perturbation response is completely defined by this expression. A characteristic of this transfer function is that the steady state of the step response \((t \to \infty)\) is a constant pitch angle, with zero pitch rate, which follows from letting \(s \to 0\) in Eq. (3.1). This expression is completely invalid for general response calculations, of course. For example, if high enough engine power is applied to maintain constant airspeed, the pitch rate will continue indefinitely and a loop maneouvre will be performed.

**Practical difficulties**

Calculated time histories of the short period and phugoid pitch response of two aircraft types, taken from Baarspu [1990] and Valasek [1996], are given in Figure 3-1. The classical simple explanation of the phugoid is that it is just a periodic exchange of kinetic and potential energy visible as oscillations in speed and altitude, with constant angle of attack. On this assumption the damping is zero and the period is given by the expression found in most textbooks, \(\sqrt{2\pi V/g}\), which depends only on aircraft true speed \(V\) and the acceleration due to gravity \(g\). The examples (using an estimated Beaver speed) follow this variation correctly but are 25% longer. A simple expression for relative damping, \(1/\sqrt{2(LID)}\), cannot be checked here but the better damping of the higher drag A4D is in qualitative agreement.³

Other factors also affect the phugoid dynamics. The pitch attitude oscillation, although slow, influences the angle of attack through the pitch rate damping to an extent dependent upon the centre of gravity position and static margin. Both the period and damping are altered. Perkins and Hage note that the actual period is typically 25% longer, in quite good agreement with the examples. Some propeller driven aircraft exhibit high cruise phugoid damping, due to thrust variation during the speed oscillations. Yet others may have slightly unstable phugoid damping at high power low speed conditions such as the climb out, for example where the constant speed propeller control system is saturated [Cook, 1997]. Because of the small forces and moments involved, an accurate prediction of phugoid characteristics is unlikely.

It is obviously difficult to examine the short period response in a time history that includes the phugoid as well. In the Beaver example, the response is initiated by an elevator input applied at time zero and removed after two seconds. The short period response barely lasts longer than that, and is compressed into a few percent of the record. The A4D short period response lasts somewhat longer than the control input because of its low damping, but it still comprises a very small part of the record. This remains a problem even if the record is terminated after two phugoid oscillations, the minimum in which its damping can be observed. If it were to be terminated after only a few seconds, the short period response is still deformed by a phugoid-related attitude change that grossly exceeds the angle of attack disturbance.

A visual impression is given of overwhelming dominance of the phugoid over the short period. Nevertheless it is common experience that the great majority of pilots never experience a phugoid oscillation, since its suppression by means of the normal tight attitude control exercised by pilots is completely instinctive. The input in the two examples represents test conditions, not pilot control in typical maneouvring situations.

³ Kamesh and Pradeep [1998] show that the expressions in standard textbooks for the short period \(\omega_p\) and \(\zeta_p\) are quite accurate, but those for the phugoid \(\omega_p\) and \(\zeta_p\) can be very inaccurate. They develop new highly accurate expressions for the latter, though that for phugoid damping is extremely large.
The same dominant impression of the phugoid is given in the frequency domain, for example in the attitude Bode plot derived from Weingarten [1981], Figure 3-2. The phugoid resonance appears visually to be the most important element, while the gentle bend at the short period frequency does not send an obvious message. McRuer et al [1960] show that pilot closed loop control of attitude is scarcely affected by the phugoid, even when it is poorly damped, because the task requires only occasional monitoring. A phugoid with slightly negative damping is also acceptable, though it may become unacceptable if it is too divergent.

The Beaver pitch dynamics in Figure 3-1 were calculated using the full non-linear and cross-coupling effects described in Baarspul [1990]. These cause roll and yaw motions from an elevator input, and pitch and flight path motions from aileron and rudder inputs. The flight testing of many aircraft to obtain flight matched aerodynamic data requires manoeuvres to be performed at constant Mach number, sometimes requiring a steep descent to maintain constant speed, in which no phugoid motion can exist. In other manoeuvres the non-linear inertial cross coupling terms can cause departure from controlled flight at worst or unsatisfactory handling at least, and so must be accounted for in pre-flight clearance calculations.

While design and flight clearance using the full equation sets, in the realms of flight mechanics and simulation, is an essential part of the handling qualities design of all aircraft, it has proved to be important for fly by wire control design to be able to subdivide the handling of conventional aircraft into basic linear elements, to understand their importance, and to quantify them in different forms. For analysis of the majority of handling quality characteristics, it is in practice essential to be able to assume a fixed speed condition.

3.2.2 The fixed speed assumption

Studies of the short period handling qualities have for decades routinely used the approximation of fixed speed, to avoid the unnecessary interference of the phugoid dynamics. In flight mechanics and simulation work, this is achieved by specifying fixed speed in the force and moment equations. Extraction of the fixed speed subset of the linear small perturbation equations was set out in Ashkenas [1958, 1962 etc], noting in the latter that this was first derived by Bryan [1911]. It is discussed in Moolij [1984], and recently again in Cook [1997].

The resulting simplified pitch transfer functions are:

Pitch attitude:
\[
\frac{\theta(s)}{\eta(s)} = \frac{K_q (1 + T_{02}s)}{s(1 + \frac{2\xi_p s}{\omega_p} + \frac{s^2}{\omega_p^2})}
\]  

(3.2)

Pitch rate:
\[
\frac{q(s)}{\eta(s)} = \frac{K_q (1 + T_{02}s)}{(1 + \frac{2\xi_p s}{\omega_p} + \frac{s^2}{\omega_p^2})}
\]  

(3.3)

Pitch acceleration:
\[
\frac{\dot{q}(s)}{\eta(s)} = \frac{K_q (1 + T_{02}s)s}{(1 + \frac{2\xi_p s}{\omega_p} + \frac{s^2}{\omega_p^2})}
\]  

(3.4)

where: \(q\) is the pitch rate,
\(K_q\) is the pitch rate gain per unit elevator.
Note that unlike Eq. (3.1), in Eq. (3.3) the steady state of the step response \((t \to \infty)\) is a constant pitch rate. In Eq. (3.2) the corresponding pitch attitude, or the integral of pitch rate, is infinite. In practice, the pitch rate achieves a steady value immediately after the short period oscillation transient has died out, and the attitude never reaches infinity for obvious reasons.

Also (simplifying by neglecting the effects of lift due to elevator deflection):

Normal acceleration:

\[
\frac{n(s)}{q(s)} = \frac{V}{g(1 + T_{\theta2}s)}
\]

\[
\frac{n(s)}{\eta(s)} = \frac{K_q V}{g \left(1 + \frac{2T_{\theta2}s}{\omega_p} + \frac{s^2}{\omega_p^2}\right)}
\]

Flight path angle:

\[
\frac{\gamma(s)}{\theta(s)} = \frac{1}{(1 + T_{\theta2}s)}
\]

\[
\frac{\gamma(s)}{\eta(s)} = \frac{K_q}{s \left(1 + \frac{2T_{\theta2}s}{\omega_p} + \frac{s^2}{\omega_p^2}\right)}
\]

Angle of attack:

\[
\frac{\alpha(s)}{\eta(s)} = \frac{K_a}{(1 + \frac{2T_{\theta2}s}{\omega_p} + \frac{s^2}{\omega_p^2})}
\]

where: \(n\) is the incremental normal acceleration or load factor,
\(g\) is the acceleration due to gravity,
\(V\) is the true air speed,
\(\gamma\) is the flight path angle,
\(K_a\) is the angle of attack gain per unit elevator, \(= K_q V / (n/\alpha)\),
\(T_{\theta2} = V / g (n/\alpha)\) secs.

\(n/\alpha\) is the normal acceleration per unit angle of attack,

**Response shapes**

The corresponding pitch attitude Bode plot for two degrees of freedom \((\alpha, q)\) is compared in Figure 3-2 with the complete plot for three degrees of freedom \((\alpha, q, V)\). They are virtually identical above frequencies about mid-way between the phugoid and short period modes. The characteristics of the latter are accurately delineated, while the usually irrelevant phugoid mode no longer confuses the picture. It is the shape of the plot that is most significant to the understanding of the handling qualities, rather than the aerodynamic and inertial derivatives that create the response mode parameters. This enables the transference of knowledge about the handling of conventional aircraft directly to that of fly by wire manoeuvre demand systems, to be discussed in more detail in Chapter 4. In these the qualities are determined almost completely by the flight control system with little influence from the aerodynamics.

Although piloted closed loop control of aircraft makes their frequency response characteristics of exceptional importance, most of the time in flight is spent controlling in a more relaxed open loop manner. For this it is the transient response behaviour that is important, as the pilot acquires a “library” of learned pre-programmed inputs to perform a variety of routine tasks. Most
of these involve control of flight path, but pitch attitude is always a vital primary cue. Of course, if normal acceleration due to elevator incremental lift is neglected, the effects of an elevator input are first a moment causing a pitching acceleration, then the pitch angle increases causing a change in angle of attack, causing an upward increment in normal acceleration that in turn starts to alter the flight path.

Thus the normal acceleration performs a vital function, but it is not the pilot's primary concern unless it reaches uncomfortable or dangerous limits too easily. Its simplified transfer function in Eq. (3.6) is only a general expression of the response, however, and only holds for the centre of gravity. Its detailed transient behaviour depends on where it is being considered in the airframe, as it contains an element of the pitching acceleration proportional to the distance of the relevant point from the instantaneous centre of rotation. This is the point about which the initial transient due to elevator deflection contains rotation but no normal acceleration response, which is forward of the centre of gravity if the elevator lift force is significant. Myers [1987] analyses the Space Shuttle landing manoeuvre where this effect initially caused additional sink at the cockpit when the pilot commanded a reduction in descent rate, since the cockpit was located some 3 metres aft of the centre of rotation.

It is the short term relationships between attitude and flight path (Figure 1-1) that are of most significance. Figure 3-3 illustrates the time response dynamics graphically. The individual elements of the transfer function response shapes defined by Eqs. (3.4) to (3.9) are visible in different parts of the transient response. The attitude is initially altered at a rate equal to the rapid angle of attack increase, identified in the numerator first order element. The flight path begins to be accelerated upwards as the normal acceleration builds with the angle of attack, both defined by the dynamics of the second order denominator, the short period oscillation mode. As these reach their steady value after the mode oscillation ceases, the response settles to a steady rate of change defined by the pitch rate gain. Although the flight path cannot easily be discerned precisely the pilot, it differs from attitude only by the angle of attack. Since the attitude is visible directly to the pilot, it can be used as a flight path surrogate.

Once again it is the response shape that most clearly illuminates the handling qualities and can quantify the handling qualities of fly by wire aircraft with differently generated dynamics. Although the short period transfer function is strictly derived from small perturbation assumptions, the fact that speed variation has been eliminated permits its use in the consideration of large perturbation response studies, provided only that the aerodynamics remain essentially linear and that no actuation non-linearities such as position, rate or acceleration limits are exceeded. The matter is discussed further in subsequent chapters.

3.2.3 Significant phugoid regimes
The phugoid is essentially a nuisance mode of oscillation, and in conventional aircraft there is almost nothing that can be done by the aircraft designer to influence it to any extent. While it is the short term response behaviour that is most important, the absence of a phugoid in many implementations of fly by wire control can have implications that need to be accounted for, some good and some not. The significance of the mode in conventional aircraft should be understood.

The phugoid motion arises from the tendency of a statically stable aircraft to return to a trimmed angle of attack after disturbance from it. To the aerodynamicist, the stability is expressed as a stable slope of \( C_{ma} \), the pitching moment coefficient due to angle of attack, about a trimmed angle of attack for which there is only one airspeed at which the lift equals the weight. Steady level flight can only take place at this speed. To a pilot, this is seen as "speed stability" but it is in-

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dicated in practice by how much the stick must be moved or pushed forwards to maintain an increase in airspeed, or rearwards for a decrease, by altering the trimmed angle of attack. For this reason it may also be called stick position stability or stick force stability. If the speed is disturbed, with no change in the angle of attack, lift no longer equals weight and the aircraft either begins to accelerate in a descent or decelerate in an ascent back towards the trimmed speed. Because the damping of this motion is very low, a prolonged slow oscillation ensues in which height and speed are exchanged, and this is the phugoid. The direct effect of increasing or decreasing power causes, not a change in airspeed, but a relative ascent or descent at a nominally constant airspeed.

In the approach to a landing, many aircraft depend on a fairly prolonged period in which extremely steady conditions of speed and flight path must be established. Most aircraft approach in a high drag flap configuration that improves the phugoid damping. It is also customary to accompany changes of pitch attitude to alter the flight path with an appropriate power adjustment, so that the speed is not disturbed and the phugoid is not excited. However, before the advent of automatic or piloted landing systems coupled to guidance beams, and the normal use of auto-throttles which make the speed control task a trivial one from the pilot's point of view, blind approaches had to be conducted by ground control approach (GCA) procedures, or even by carefully timed descent rates from range markers on a beacon course. The presence of an unsatisfactory phugoid can make the task much more difficult because of the variations in the flight path and speed. Newell [1957] reports on these problems, including a fatal airliner crash event in 1954 that is illustrated in Klyde et al [1995].

The phugoid zero frequency static attitude gain also contributes to good control of the landing flare manoeuvre. Figure 3-4, derived from Weingarten [1981], sketches a generic low speed pitch response to elevator inputs. The angle of attack response closely follows the control input, the pitch rate decays gradually from the initial value as the speed decays due to the change in flight path, and the attitude shows a trend to a fixed value at a new steady state flight path angle. After stick release, the aircraft pitches down, eventually returning to the original attitude and flight path. As noted above, the pilot uses attitude as a surrogate for flight path. These effects, although initiated here in level flight, can be assumed as increments to the typical 3 degree approach glide path. An actual landing control input is likely to be more gradually applied, resulting in less of a phugoid than in a gradual transition, usually also with reduced power, to a new steady speed at higher angle of attack and increased attitude. The touchdown is performed as the attitude rate and rate of descent reduce towards zero.

The customary close attention paid to controlling pitch attitude is sufficient in most cases to prevent the phugoid oscillation. There is however one potential form of phugoid excitation which has an impact on flight safety, the "micro-burst" [White, 1990 and Fujita, 1986]. This is a mass of rapidly descending cold air falling from a thunderstorm system, which spreads out horizontally in all directions near the ground. An aircraft penetrating this during the approach to land will experience a headwind-tailwind sequence that can be of a very large magnitude. Because of typical micro-burst dimensions and the normal approach speeds of transport aircraft, the airspeed oscillation frequency could well be close to that of the phugoid. If unchecked, the resulting attitude excursion may be greatly increased by the poorly damped phugoid resonance, adding to the serious difficulties in control of altitude and airspeed often experienced in this situation. This is more of a problem for civil aircraft with low surplus power margins than for high powered military combat types, but it is never insignificant.

Although the phugoid motion seldom causes problems at high speeds for conventional aircraft,
the pilots’ difficulties in the crash landing of a Douglas DC-10 in 1989 [Faith, 1996] were compounded by a large phugoid motion reported to reach peak vertical velocities of 800 metres per minute. This was after a complete loss of hydraulic power to the controls due to an engine turbine disc disintegration, leaving engine power adjustment as the only means of control. At hypersonic speeds the very large kinetic energy could cause phugoid oscillations of immense magnitude, easily a kilometre vertically from peak to peak, for small speed changes. In these conditions the altitude and Mach variations have a strong influence on the motion, for which the conventional dynamic assumptions are inadequate [Sachs, 1990]. A fly by wire control system could eliminate the phugoid and this problem with it, though according to Sachs it is necessary to employ a more complex feedback system than in the normal flight regimes.

The effects of the phugoid are particularly significant for flight at low airspeeds, where the oscillation is also likely to be most pronounced, with a relatively high frequency and low damping. Obviously these effects cannot be examined in the fixed speed approximations, but in flight simulations where such flight is qualitatively examined, full equations are invariably used.

3.2.4 Flight above and below the minimum drag speed

The relatively small but sometimes important part of the flight envelope below the minimum drag speed is the source of much confusion about apparent speed instability. It was known as early as 1910 [Hancock, 1995] that maximum power in level flight is required at both minimum and maximum speeds, with a condition of minimum drag lying between them. It was then believed that as the drag increased below this minimum drag condition, the aircraft would slow to the stall with no pilot intervention, that is the speed would be unstable. Although this was disproved on theoretical grounds a decade later, there being no unstable roots in the airframe dynamic equations, the fallacy persists among many pilots to this day. The phenomenon had already been explained correctly in practical terms by Loening [1916], who called it “the region of reversed flight”.

When at speeds above minimum drag, on the “front side” of the drag curve, Figure 3-5, the path responds as expected if power or thrust is not adjusted. Pulling the stick back slows the aircraft and reduces the thrust required, and so the aircraft climbs. If this is done below minimum drag, on the “back side” of the drag curve, the path response reverses. Pulling the stick back slows the aircraft just as before, but as the thrust required increases the aircraft descends. Piston engines operate approximately as constant power devices. The "region of reversed control" is below the minimum power speed, which is about 13% less than the minimum drag speed, and is usually too close to the stall to be significant for normal approach speeds.

The variations in speed with changes in angle of attack, which is controlled directly by the elevator, are entirely normal. The appearance of instability arises only from ignoring the effect on speed of diverting the pitch control to an attempt to maintain flight path. The theory of “stability under constraint” was given by Neumark [1957]. In military handling qualities specifications e.g. [Anon 1969] it is referred to as flight path stability under pitch control at constant power. Typically it will be required that this must not be negative, or at most it may be only slightly so, which can determine the minimum approach speed.4

A distinction must be made between the aircraft front or back side drag characteristic and the pilot techniques referred to as front side or back side. When well on the back side of the drag curve, the back side technique of power for flight path and stick for speed is the only feasible

---

control. For most aircraft, extension of high drag flaps reduces the minimum drag speed to below, or close to, normal approach speeds. On the front side of the drag curve, the front side technique of controlling the flight path with the stick and the speed with the throttle is feasible. With a slow thrust response, especially for higher approach speeds, this is the most desirable approach technique. Of course, the steady condition is always trimmed by the stick for speed and by the power for flight path.

However, climb or descent in the front side regime is usually effected through power alterations. After an initial perturbation of the speed, which can be prevented by a transient pitch attitude adjustment, the aircraft will settle at the new flight path and attitude more or less at the same speed as before. By contrast, to change the speed in level flight the power setting is typically changed immediately to that required for the final speed, and level flight is maintained by continual attitude (actually angle of attack) adjustments with the stick until the final speed is reached. Hence either technique can be used interchangeably depending on the precise nature and size of the change demanded and the engine response.

Naval aircraft represent an extreme example by approaching at very low airs speeds, as little as 1.08 times the stall speed. At such backside speeds, satisfactory path control is only possible if the engine thrust response is sufficiently rapid at approach power settings. In land based aircraft the demands for fuel economy often require high bypass ratio engines with slow throttle response at low power, usually necessitating the front side technique. When both the flight path and engine response are inadequate, as in early General Dynamics F-16 versions [Milam, 1982], it is difficult to achieve satisfactory approach control.

Given appropriate inceptors and characteristics, pilots adapt well either to a front side (path by stick, speed by power) or to a back side (path by power, speed by stick) technique. This is supported by results from the flight tests of the NASA Augmentor Wing Jet STOL Research Aircraft [Hindson et al, 1978]. Landings were made in three modes, basic (natural aerodynamics), front side with autothrottle, and backside. There was no significant difference in pilot opinion, workload or performance, though the tests were conducted only in calm weather conditions. Milam [1982] discusses how one or the other, or both, techniques are taught according to aircraft characteristics. Both techniques are taught to elementary pupils at the Royal Air Force Central Flying School [Bruce, 1996].

3.3 Factors influencing lateral-directional handling qualities analyses

Motions in the roll and yaw axes are coupled in a way that makes it difficult to consider them separately with any exactness. As shown by Baarspul [1990] and also by Irvoas [1982], it can be possible for them to interact with the symmetric motions as well. The classic assumption of separation from the pitch motions is nevertheless satisfactory for general handling studies. Although the lateral-directional dynamics are very different to the pitch dynamics in nature, the same general comments can be made about them.

Bryan [1911] and other early theoreticians made the same mistake about the importance of the long period lateral spiral motion as about the phugoid. Despite the extremely weak forces available to provide lateral static stability, efforts were made to devise aircraft configurations with high inherent stability in the belief that this was important to pilots. The result, as noted in Chapter 2, was often poor and sometimes dangerous control qualities that caused many accidents. It is usually difficult to obtain positive spiral stability at high lift coefficients, but the resulting divergence is very slow and will often not be noticed while the pilot controls the aircraft.
Positive spiral stability is desirable in cruising flight for unattended operation, but this is readily achieved at the lower lift coefficients in the cruise.

3.3.1 The classic small perturbation lateral-directional transfer function

The transfer function of roll angle to aileron control is:

\[
\frac{\varphi(s)}{\xi(s)} = \frac{K_p \left(1 + \frac{2\zeta_{\varphi} s}{\omega_p} + \frac{s^2}{\omega_p^2}\right)}{(1 + T_s s)(1 + T_s s)(1 + \frac{2\zeta_d s}{\omega_d} + \frac{s^2}{\omega_d^2})}
\]  

(3.10)

where: \( \varphi \) is the roll angle,
\( \xi \) is the aileron control angle,
\( K_p \) is the roll angle gain,
\( \zeta_{\varphi} \) is the roll numerator damping ratio,
\( \omega_p \) is the roll numerator natural frequency,
\( T_s \) is the spiral mode time constant, secs.,
\( T_s \) is the roll damping mode time constant, secs.,
\( \zeta_d \) is the Dutch roll mode damping ratio,
\( \omega_d \) is the Dutch roll mode natural frequency.

The steady state of the step response (\( t \to \infty \)) is a constant roll angle. The roll mode represents the response of direct concern to the pilot, the control of roll rate and roll angle. The spiral and Dutch roll modes are nuisance effects that may exert undesirable influence on the response, typically by making the roll response "untidy". The numerator represents also the disturbing cross-axis effects of the aileron, particularly significant in the excitation of the Dutch roll mode.

Practical difficulties

Figure 3-6 illustrates a calculated roll response of a Beaver aircraft [Baarspul, 1990]. The transient response associated with the two-second application and removal of aileron control is completed shortly after. The response takes some 50 seconds to return to a steady state, the roll angle gradually drifting back to wings level while the heading changes slowly. The roll response is heavily contaminated by the large sideslip, setting off a well damped Dutch roll.

Normally, however, a pilot would transform the response by simply applying sufficient rudder with the aileron to prevent the initial sideslip and Dutch roll disturbance. In a fully augmented control system this would be performed automatically by crossfeed signals between the roll and yaw axes. This has the effect of making the numerator dynamics effectively the same as the Dutch roll dynamics, counteracting the latter. Unlike the phugoid, the spiral convergence does not produce a large effect, and can reasonably be ignored in studies of transient roll responses.

Just as for the pitch axis, a full set of non-linear equations of motion is always used for full design, simulation and clearance purposes. In fly by wire aircraft, the treatment of the Dutch roll mode is generally identical to that familiar in conventional aircraft. For the general study of basic roll control parameters and problems, a simplified linear subset of the transfer functions provides most of the dynamics needed on which to base an understanding of conventional behaviour and to quantify them in forms suitable for use with fly by wire systems.

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3.3.2 The simplified roll control subset
These are as follows:

Roll angle:

\[
\frac{\psi(s)}{\xi(s)} = \frac{K_p}{s(1 + T_r s)}
\]  

(3.11)

Roll rate:

\[
\frac{p(s)}{\xi(s)} = \frac{K_p}{(1 + T_r s)}
\]  

(3.12)

Roll acceleration:

\[
\frac{\dot{p}(s)}{\xi(s)} = \frac{K_p s}{(1 + T_r s)}
\]  

(3.13)

where \(p\) is the roll rate

where \(K_p\) is the roll rate gain,

where \(T_r\) is the roll mode time constant equal to \(p_{st}/\dot{p}_{max}\), secs.

where \(p_{st}\) is the steady roll rate after a step aileron input,

where \(\dot{p}_{max}\) is the maximum initial roll acceleration after a step aileron input.

The simplified expressions assume only the existence of the roll damping mode. The spiral mode is assumed to be neutrally stable with an infinitely large time constant, becoming in effect the integrator function \(1/s\). The disturbing effect of the Dutch roll mode is eliminated by the assumption that the augmentation design will prevent its initiation in any roll response. The roll angle in the step response now has an infinite steady state gain, Eq. (3.11), and a constant roll rate, Eq. (3.12), which is consistent with normal roll control behaviour in practice.

Response shapes
The control of roll angle tends much more to a general pursuit tracking activity than to tight compensatory tracking, discussed in Chapter 5. Typically the pilot applies the aileron, observes the resulting roll rate and removes the control at a pre-judged point so that the roll ceases at the desired roll angle a small time later. Roll characteristics are quantified most often for conventional aircraft by a simple statement of the roll mode time constant, the lag between the control action and the roll response.

The roll mode time constant is by the usual definition the time to reach 63% of the steady rolling velocity following a step aileron input. It is unlikely that this measure is of any significance to a pilot. Figure 3-7 shows the idealised roll angle, rate and acceleration transient responses with the time expressed in \(T_r\) units. This shows that there are more natural definitions of response characteristics that can be used. In particular, the pilot observes the roll damping by the ratio of initial acceleration to the steady roll rate, and especially by the overshoot in roll angle or the time taken to arrest the roll rate after the control is removed. The time constant derived from the latter may differ from the onset value because any residual sideslip at the end of the manoeuvre will modify the recovery to zero roll rate, so it must be derived from use of the full equations.

The extrapolation of conventional behaviour to the analysis of fly by wire systems is readily achieved by application of such alternative shape measures. This is discussed in Chapter 7. Oth-
er factors that are absent conventionally, but that can lead to pilot induced oscillations in fly by wire aircraft, are discussed in Chapters 8 and 9.

3.3.3 Significant cross-coupling effects

Unlike the pitch axis where large transient manoeuvres can often be illuminated usefully by the linear small perturbation fixed speed approximation, considerable non-linear kinematic and inertial cross-coupling exist between all three axes in all but gentle rolling manoeuvres. This necessitates the use of the full non-linear equations for investigation of roll handling limits. In the early jet era, the linear equations commonly used up to that time were abandoned following a number of aircraft losses due to inertially coupled departures in rapid roll manoeuvres at low or negative angles of attack. Abzug and Larrabee [1997] and Day [1997] give a detailed summary of the events of that period with many references. Phillips [1948], Pinsky [1958], Cronvich et al [1961] typify analyses of the problem. The difficulty of calculating the motions was eased by the development of analogue computing. With today's digital computing and comprehensive wind tunnel testing, it is possible to model quite accurately a motion as complex and non-linear as departure into a full spin.

Such motions are generally quite violent and large in scale, but another coupled motion earned the title of "the graveyard spiral" despite involving small roll rates and normal angles of attack. The typical scenario is an inexperienced pilot who enters instrument conditions, fails to notice a slow spiral divergence, then observes only a loss in height, and pulls back on the stick. The roll angle, turn rate, descent rate, airspeed and normal acceleration increase until the aircraft is in a steep high g spiral dive. The mechanism of this is seldom discussed, but was investigated by Velger et al [1983]. Three almost distinct phases were found: an immediate and normal nose up pitch response; a divergent lateral motion with increased roll rate, sideslip and yaw rate; and a kinematic response in which the pitch attitude turns downwards, dominated by the increasing roll angle and yaw rate. The recovery action is trivial, requiring only that the wings be levelled before pulling up.

3.4 Summary

The complex mathematics required for an accurate description of aircraft motion dynamics presented considerable difficulty in computation until the 1950s, when analogue computers became widely used. With only pilot descriptions of the handling available to them for many years, aircraft designers eventually had access to some rather basic handling qualities guidelines and criteria, essentially confined to small perturbation characteristics. Dating from the 1940s, with a major update in 1969 (Chapter 2), the criteria remain useful and valid to this day for conventional aircraft. The primary reference for this material is Chalk et al [1969], with comprehensive discussions of the criteria and lists of all the source material.

These criteria are primarily dependent on parameters of the linear small perturbation equations of motion, typically the mode natural frequencies and damping ratios. Their numerical values were linked to ranges of satisfactory or unsatisfactory handling qualities, an approach that failed with the advent of new response mode characteristics in fly by wire aircraft. This chapter introduces the concept of author's new approach by the use of the simplified forms of the dynamic equations to illustrate graphically a much wider range of physical response characteristics identified in the generic response shapes. How these are related directly to a pilot's opinion, and can directly link the handling of fly by wire aircraft to conventional aircraft by an essentially graphical and non-mathematical design process, is described later in Chapters 7 and 10.

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(a) Light wing loading moderate drag aircraft, a De Havilland DHC-2 Beaver from [Baarspul, 1990]

(b) High wing loading high drag aircraft, a Douglas A4D Skyhawk at $M = 0.2$, sea level [Valasek, 1996]

Figure 3-1 Examples of computed short period and phugoid oscillations
Figure 3-2 Generic pitch attitude frequency response plots, in three and two degrees of freedom
\[ \gamma = \int n g / V \, dt \]

\[ \dot{\gamma}_{ss} \equiv q_{ss} \]

\[ q_{max} \]

\[ \alpha_{ss} \]

\[ T_{\theta_2} \]

\[ \theta = \text{pitch attitude} \quad q_{\text{max}} = \text{maximum pitch rate} \quad q_{ss} = \text{steady state pitch rate} \]

\[ \alpha = \text{angle of attack} \quad \alpha_{ss} = \text{steady state angle of attack} \]

\[ \gamma = \text{flight path angle} \quad \dot{\gamma}_{ss} = \text{steady state flight path angle rate} \]

\[ n = \text{incremental normal acceleration} \]

\[ V = \text{true air speed} \quad T_{\theta_2} = V / g (n / \alpha) \, \text{secs.} \quad n / \alpha = n \text{ per unit angle of attack} \]

Figure 3-3  Generic shape relationships of angle of attack, attitude, flight path and normal acceleration

Figure 3-4  Sketches of generic responses at landing speed of conventional aircraft, adapted from [Weingarten, 1981]
(a) Basic formulation of aircraft drag curve

(b) Effect on flight path of pitch control of airspeed at constant thrust setting

Figure 3-5 Front and back side flight regions and flight path instability

Conventional aircraft dynamics
Figure 3-6 Example of lateral-directional response to aileron control of a DHC-2 Beaver [Baarspul, 1990]

Figure 3-7 Generic roll response shapes related to the roll mode time constant
Chapter 4
Basic elements of handling qualities 2: Dynamics of augmented and superaugmented aircraft

4.1 Introduction

This chapter discusses in general terms the typical handling characteristics of the main classes of manoeuvre demand flight control systems. A simple example is shown in Figure 4-1, which combines the functions known as "stability and command augmentation". Here the pilot's controller signals a demand for a specified pitch rate manoeuvre response, rather than the deflection of a control surface as in conventional aircraft. The aircraft response differs in kind and degree from the conventional, and cannot be described by the normal pitch modes. These systems are often described as "high order". They have been extensively reviewed elsewhere, e.g. Myers et al [1984], McRuer et al [1982, 1983, 1986].

The handling qualities problems that have been encountered tend to be related specifically to three areas. These are:

- long term low frequency effects (the “phugoid area”),
- immediate short term effects (the “short period area”),
- pilot induced oscillations (PIO) unique to high order systems.

The short period and phugoid motions are symmetric oscillation modes. PIOs have occurred mostly in the symmetric short period mode, but have also occurred in the asymmetric modes.

Long term and short term effects in superaugmented aircraft only superficially resemble the conventional responses discussed in Chapter 3. They are discussed briefly in this chapter, and more extensively in Chapter 7 where it is shown how the response characteristics can be correlated. High order PIO, and other types such as pitch bobble, the pitch tracking "PIO syndrome" and flight path PIO which have always existed in more classical designs (usually with less serious effects), are discussed separately in Chapters 8 and 9.

The principal driver for the development of fly by wire technology has been the improvement of combat aircraft performance by reduced or negative angle of attack stability, i.e. with negative static and manoeuvre margins. The most extreme examples of unstable designs would have a divergence in pitch attitude with a time to double amplitude of as little as 0-2 seconds or even less in the absence of stability augmentation. As an aircraft with far less instability than this cannot be controlled by a human pilot, it is necessary to employ continuously operative stability and command augmentation using multiple signal paths for reliability and safety.

The term "superaugmentation", a useful name for many modern forms of fly by wire control systems, appears to have been coined by McRuer [1986] to denote an important class with specific characteristics: the aircraft are statically unstable without augmentation; they have a degree of pitch attitude stability with respect to inertial space provided by the flight control system (as opposed to the conventional weathercock stability with respect to angle of attack and sideslip); and the commanded response behaviour is largely independent of the aerodynamic stability derivatives. The author prefers to use the term for any flight control system that provides behaviour almost independent of the aerodynamic qualities through the use of integral-plus-proportional action on the demand error, e.g. $q_0$ in Figure 4-1, but extended to include other demanded responses such as normal acceleration, angle of attack and flight path angle.
To obtain sufficient integrity and reliability in the flight control system of an unstable aircraft, multiple sensors are needed at least for the core or heart of the system. This is difficult to achieve with angle of attack sensors or measuring devices, which are both vulnerable to damage and unlikely to provide identical signals from necessarily different locations on the fuselage. Such difficulties do not apply to inertial sensors such as rate gyro's or accelerometers, though the latter especially need very careful location or compensation to avoid unstable coupling with aircraft structural modes. Pitch rate demand systems, therefore, have become widely (though not universally) used as the system core, especially in combat aircraft. In Anon [1959], it was concluded that fighter or interceptor aircraft should employ pitch rate and attitude based systems. Research was conducted at the Netherlands NLR that reached the same conclusion for the low speed control of civil aircraft with fly by wire systems [Mooij et al., 1979, 1984].

Transport aircraft policy has been more conservative in retaining enough stability to ensure safe control of the unaugmented aircraft. The first fly by wire transport was of course the supersonic British Aerospace/Aérospatiale Concord, but the technology was primarily used to overcome difficulties in design of the mechanical control system in an extreme environment. Enhancement of operational efficiency and safety with fly by wire control was led by the Airbus A.320, A.330 and A.340 series and has been followed by the Boeing 777. The idea of further improving efficiency by relaxing the static stability requirements is now becoming accepted. It is believed that the USA's proposed supersonic High Speed Transport will feature substantial static instability, presumably in the subsonic region so that the stability in the supersonic cruise can be optimised. This was addressed in Concord by pumping fuel aft for the cruise and forward for the subsonic slow-down and landing.

Sufficient advantage may accrue in the future to justify fly by wire technology even where the aerodynamic and environmental requirements are modest and do not in themselves create a need for it. The cost of ownership of transport aircraft includes considerable maintenance and replacement costs for mechanical control systems. Electrically signalled spoiler control has been used widely for this reason from the Boeing 757 and Airbus A-310 onwards. Integration into the increasingly complex future air traffic control may require more automated flight control and management than at present. Such considerations could make the economics of fly by wire more generally attractive even in much smaller aircraft than the types mentioned above.

4.2 Long term effects in manoeuvre demand types

Insufficient pitch stability or instability, with too small or negative static or manoeuvre margins, can be corrected or enhanced by any of angle of attack, normal acceleration or pitch attitude feedbacks, the latter being implemented in practice by integral pitch rate. It is usual, as is assumed here, to use the proportional plus integral structure of Figure 4-1 in order to achieve a uniform and zero error steady state response to the pilot's control inputs. This enables the ready implementation of functions such as manoeuvre limiting to protect the airframe from over-stressing or from the stall, regardless of weight or centre of gravity variations. In combat aircraft, such facilities allow the pilot to concentrate on tactics and weapon operation without distraction by concern for the safety of the aircraft itself. In airliners, they provide the means to maximise the aircraft lift performance with the obvious benefits of a positive protection from the stall.

Such manoeuvre demand systems can have unexpected and undesirable effects, due to failure of the designer to assess the characteristics in full. One potential problem lies in the likely absence of the classical phugoid oscillation. Although the phugoid itself is an undesirable nuisance
mode, it exists because of the angle of attack stability to which is related the fundamental properties of speed stability in conventional aircraft, discussed in Chapter 3. Its absence may therefore be a mixed blessing, since this generally indicates zero speed stability. This is not always undesirable, but it must be taken into account and rectified when necessary. Provision of speed stability, by a system referenced to speed rather than the conventional angle of attack, will usually restore a long period phugoid-like type of oscillation as well.

4.2.1 Pitch rate demand

The typical basis of a proportional plus integral Pitch Rate Demand or PRD system is shown in Figure 4-1. This may sometimes be referred to as a pitch rate demand/attitude hold system, but the attitude element is only implicit, relying on the ability of the integrator to maintain zero pitch rate in the absence of a pilot demand. Hence there is a considerable resistance to pitch attitude disturbance, though it is not absolute because there is no restoration to a defined attitude datum. A true attitude hold system uses attitude feedback to perform this function. In that case the attitude is maintained or restored after a gust induced disturbance, for example.

Steady flight with zero pitch rate is commanded with the stick central, and no stick trimming is needed as speed changes. This creates zero static or speed stability, equivalent in piloting terms to zero stick trim stability. The elevator itself of course continues to adopt the usual trim position variations in a stable or unstable sense according to the actual airframe stability characteristics. The static stability is also zero in the sense that $C_{ma}$ is effectively made zero by the control system, which suppresses the aerodynamic pitching effects of angle of attack variations. The stick commands a pitch attitude response resembling the two degree of freedom approximation discussed in Chapter 3, Figure 3-2. The small-perturbation attitude no longer has a steady state value, which results in conventional aircraft from adoption of a new steady speed at an increased angle of attack. The phugoid mode is suppressed.

In a conventional, statically stable aircraft, reducing or increasing the power to descend or climb achieves this end without the need for a pitch control input and with little change to the steady speed, though it is normal to adjust the attitude simultaneously simply to prevent a transient speed disturbance. With a PRD it is necessary rather than optional to adjust the attitude by pitch control when a climb or descent is required at constant speed. If a level flight acceleration is commenced at low speed with a large AoA, it is first necessary to push the nose down to maintain level flight. At high speeds the flight path changes are small and require only slight temporary attitude adjustments to maintain level flight. No trimming is necessary.

There is a significant consequence of a PRD system to flight safety when below the minimum drag speed. This is related to the conventional flight path instability discussed in Chapter 3, that is the unstable flight path reversal in response to pitch control inputs. In a PRD system the elevator is used to maintain constant attitude without any pilot control input, and consequently the angle of attack can diverge substantially if the pilot fails to notice. A stall is possible in an essentially level attitude. The pitch attitude, closely tied to the flight path through angle of attack in conventional aircraft, no longer serves as a surrogate cue to the flight path because there is no inherent control over angle of attack.

A pitch rate develops in all aircraft when the lift vector is directed by a bank angle to one side of the vertical. If the lift is not increased to maintain level flight, a turning descent will commence. A PRD will reduce the angle of attack to prevent the undemanded pitch rate, so that the flight path angle accelerates downwards more rapidly due to the greater vertical lift deficiency. As pitch rate per $g$ is slightly greater in a steady turn than in level flight, a PRD will tend to pro-
duce a slightly greater pitch up when the wings are levelled than with conventional control if the pitch input is not carefully co-ordinated. These effects are noticeable at first in simulation, but they appear to be easily adapted to and are scarcely noticed in flight.

4.2.2 Normal acceleration demand

The use of an accelerometer-based demand system introduces some further problems, noted in McRuer et al [1986]. These include instability below the minimum drag speed (with a PRD the attitude stabilisation is not affected even though the angle of attack diverges), and a bias exists if the accelerometers are not oriented along the level flight axis. The sensors measure all rigid and flexible body acceleration at their location, including that due to pitch acceleration, elevator lift force, airframe vibration and even roll rate. While some of these could in principle be compensated, the effects can make the choice of their mounting location quite difficult.

In a system referenced to uncompensated acceleration fixed to the airframe (as the sensors are), the sensed specific force decreases with increasing pitch angle, and therefore also with angle of attack in level flight. This produces an apparently negative static stability as a forward stick command is required with decreasing speed. This is equivalent to negative elevator trim stability, but again the actual elevator adopts the angles determined by the underlying airframe stability. It is possible to obtain steady flight for a zero command by a gravity compensation term to demand zero inertial acceleration relative to the earth. No trimming is then required for any steady condition, whether in level, climbing or diving flight and the static stability appears neutral. Such systems suppress the phugoid.

A stall will be induced if the non-linear lift region is entered, increases in the AoA being unable to satisfy the demand. Unlike a PRD system which retains attitude control, a g demand system will tend to pitch up rapidly, since increasing the AoA reduces and usually reverses the lift slope, further increasing the lift deficiency. Both of these system types require the pilot to monitor the angle of attack closely if they are manoeuvred below the minimum drag speed and especially near to the stall.

4.2.3 Angle of attack demand

In common with the classical response type, aircraft in which the angle of attack is explicitly stabilised and controlled can hold airspeed and flight path with considerable precision. The usual phugoid is present, together with static stability and the resulting trimming required with variations of speed. With the speed set by the stick/pitch control setting and the flight path by the power setting, the aircraft returns without pilot intervention to these values after being disturbed from them.

With integral control, however, typically the static stability can appear excessive. It is difficult to replicate the normal classical characteristics to which pilots are accustomed. The distinction between static and manoeuvre stability is removed, so that instead of the stick force for a given change angle of attack in level flight being less than in a constant speed manoeuvre, it remains the same. In conventionally stable combat aircraft, a reduction in static stability is common with increasing Mach number, so that less forward stick pressure is necessary in diving attacks with less risk of an excessive normal acceleration response when the pilot relaxes the stick pressure. In the author's experience this demand mode has been greatly disliked by pilots in simulation assessments of potential control designs. Mooij et al [1979, 1984] also found a dislike of angle of attack demand systems in their investigations of the landing approach in fly by wire civil aircraft, and McRuer et al [1986] compare these systems unfavourably with pitch rate types.
4.2.4 Flight path angle rate demand

Flight path angle rate demand systems are as yet largely untried in flight, but are becoming feasible as an alternative. In principle they go directly to the heart of flight path control by using it as the controlled parameter instead of the surrogate parameters above. Several standard autopilot modes already perform such a function, of course, and it is sometimes possible to control through them by "little stick" or control stick steering. These are confined to essentially wings-level or moderately banked flight, however.

In gross manoeuvres, the problem arises as to the definition of the required flight path angle, e.g. it remains zero in a highly banked level turn and thus provides no response stimulus. In order to provide the required pitch axis commands, pseudo flight path angle rate is possible by using the modified pitch rate command \((q - \dot{\alpha})\), equivalent to \(\dot{\gamma}\) in level flight. This arises from the relationship between flight path, pitch angle and angle of attack, shown in Figure 3-3:

\[
\gamma = (\theta - \alpha)
\]

\[
\text{hence } \dot{\gamma} = (\dot{\theta} - \dot{\alpha}) = (q - \dot{\alpha}).
\]

Such a mode appears to have the same limitations as PRD or g demand systems in that there is no constraint on the demanded AoA, with the potential for inadvertent stalling. There is similarly no natural static stability. No definitive consensus appears to have emerged about such a mode at the time of writing. However, successful studies of flight path control laws are reported by van der Geest et al [1992] (at Fokker) and Verspay et al [1996] (at NLR). Joint studies by GARTEUR\(^5\) are discussed by Nicholas et al [1991].

4.3 Short period effects

These embrace the essentially constant speed transient manoeuvres that immediately follow any control input, as opposed to any quasi-steady manoeuvre that may ensue at fixed or varying speed. The characteristics are conventionally defined by the short period pitching oscillation. It is in this region of the response characteristics that most unexpected handling qualities deficiencies have arisen, through insufficient understanding of conventional characteristics.

4.3.1 Pitch rate demand

A PRD system will usually confer excellent attitude disturbance rejection qualities with good behaviour in turbulence, though this is at the expense of increased heave motions because the conventional aircraft gust alleviation response of pitching into gusts is prevented. This can be an acceptable compromise because pilots feel more in control when the attitude is not being randomly disturbed by external influences, particularly in flight near the ground. Excessive heave can be moderated if necessary by the use of direct lift gust alleviation without losing the attitude steadiness. The controlled response, on the other hand, is not explicitly shaped in terms of the attitude parameters shown to be so important to pilots (see Chapter 7).

The explicit control of pitch rate implemented in a PRD system as in Figure 4-1 can lead to sluggishness in the flight path response. This is indicated by the generic sketches of Bode amplitude asymptotes\(^6\) in Figure 4-2 from Myers et al [1987], examining the poor landing control qualities of the Shuttle Orbiter. The conventional attitude numerator break point at the \(1/T_{\theta 2}\)

\(^5\) Group for Aeronautical Research and Technology in Europe

\(^6\) The use of Bode asymptotes in this manner, rather than exact response plots, is commonplace in handling qualities literature. It clearly reveals the skeleton structure of the elements of the response,
frequency is replaced by a break at the much higher $1/T_q$ frequency, the ratio of the integral to proportional gain. This produces excellent control of pitch rate, but suppresses most of the attitude lead relative to the flight path generated by the angle of attack response in a conventional aircraft. Hoh et al [1996] note the high $1/T_q$ in the F-16 to be a main factor in its imprecise pitch control in the landing.

Expressed in another way, the initial pitch rate overshoot in a conventional transient response performs the task of rapidly increasing the angle of attack and with it the normal acceleration (see Chapter 7). In a PRD this overshoot may be eliminated or very greatly reduced. The angle of attack can then initially increase only at a rate equal to the final steady flight path angle rate since (initial $\dot{\alpha}$) = $\dot{\alpha}_{ss} = \dot{\gamma}_{ss}$ (see Figure 3-3). The angle of attack rate decreases further as $\dot{\gamma}$ reaches its final value since $\dot{\alpha} = (q - \dot{\gamma})$. If the required AoA is large relative to the steady flight path angle rate, the initiation of the flight path changes is very sluggish. The effect can be inferred in Figure 4-2 from the lag dynamics added to the flight path Bode plot at $1/T_{\alpha 2}$, typically at about 0.5 to 0.7 radians per second for most aircraft in the landing condition. Although the gain asymptote only departs from the normal at this point, there is already 45 degrees extra phase lag which increases rapidly beyond this. The effects lie in the middle of the range of frequencies which dominate the piloted closed loop control of flight path.

However, this problem had already been observed and disposed of in the late 1970s in initial development of the PRD control laws of the BAe FBW Jaguar [Daley, 1984]. Although the pitch response was fast and well damped, with an apparently excellent "short period frequency" mode, preliminary simulation assessment found that the flight path response was sluggish and unacceptable. The aircraft had a high wing loading and typically low wing lift slope, so that a relatively large angle of attack increment per g was required, with a result resembling the superraugmented transient response in Figure 4-2. The solution was both obvious and simple, in the form of a feed forward signal from the stick to the pitch actuator command that by-passed the integral controller to provide the missing pitch acceleration. This technique is discussed further in Chapter 10.

4.3.2 Normal acceleration demand

Normal acceleration demand can be expected to show quite conventional control response characteristics, since it commands the parameter from which flight path follows directly. In turbulence, the flight path is less disturbed than with a PRD because the AoA is continuously altered to maintain a near constant g. However, this causes greatly increased attitude disturbances which can be mentally uncomfortable for a pilot in low level flight. If the pilot responds by suppressing the attitude excursions, the flight path performance will be no better than that of a PRD but the pilot will experience an increased workload.

While this command mode is often thought to be the optimum for precision flight path control, practice has shown this is not necessarily true. Milam et al [1982] discuss the change from normal acceleration to pitch rate control for the F-16 to improve its unsatisfactory handling in the approach mode (see also §4.5.1). In fact, PRD is used for landing mode in the majority of recent military FBW aircraft, and it was the recommended control mode in the NLR research work on fly by wire transport aircraft [Mooij, de Boer and van Gool, 1979].

4.3.3 Angle of attack demand

AoA demand also provides conventional short period responses, including a significant attitude response to turbulence. This can be quite objectionable to pilots, who are more comfortable with reasonably steady attitude behaviour especially when flying at high speeds and low altitude. The
excessive angle of attack and sideslip stability provided by the initial control laws of the SAAB Gripen were contributory factors to the crash of the first prototype [Knotts et al, 1990].

Excessive and potentially destructive control commands may be generated by passage of an angle of attack sensor through the core of another aircraft's tip vortex. An instantaneous change in indicated vertical velocity of more than 30 meters per second, as measured by the AoA sensor output, was recorded during such an encounter in combat test flying of the British Aerospace EAP aircraft (the Experimental Aircraft Programme, Hartley [1988], McCuish et al, [1996]).

Such problems result from the use of conventional local flow direction detectors such as vane sensors. These reflect both the inertial AoA variations and the local airflow fluctuations due to turbulence. The solution is to use complementary filters to provide high pass short term inertial and low pass steady state aerodynamic measurements of angle of attack. However, the difficulty of providing a sufficiently redundant source of angle of attack to satisfy reliability requirements makes it unlikely as a primary control mode.

4.3.4 Flight path angle rate demand
A flight path rate system might be expected to have the sharpest flight path response. The initial values of these parameters immediately following an input are almost identical, and there is almost no initial feedback signal to reduce the acceleration. Accordingly the response to turbulence could be even sharper and less acceptable than a $g$ or AoA system. In the author's experience of one simulation comparison, its path holding ability was less good than a simple PRD. If struck by a down gust immediately after its nose down reaction to an up gust, the aircraft with a $(q - \dot{z})$ system was flung downwards further as a result of the reduced AoA which could not be instantly increased again. The flight path variation was very substantially worse than in a competing PRD system. The latter, by maintaining constant attitude in the approximately zero mean vertical velocity of random turbulence, maintained a reasonably constant mean flight path.

In the S/MTD, or STOL/Manoeuvre Technology Demonstrator, a McDonnell Douglas F-15E modified with added foreplanes and vectored thrust for a USAF study of possible improvements in take-off and landing performance and enhanced manoeuvrability [Moorhouse et al, 1994], it was found that an approximate flight path rate demand for STOL approach, using a $(q - \dot{z})$ system, was not successful. A PRD with autothrottle gave superior qualities (§4.5.5).

4.4 Landing and take off

4.4.1 Pitch rate demand
In Chapter 3 and Figure 3-4, the general properties of landing flare behaviour in conventional aircraft were illustrated, with an angle of attack response closely following the control input. The characteristics of the basic PRD system requires the stick to be operated in more of a pulsed manner than the conventional continuous technique. Pilots can adapt quickly to the pulsed control required in the approach to adjust the attitude, especially with a small stick controller suited to this [Myers et al, 1987]. As illustrated in Figure 4-3 the pitch rate in the landing flare closely follows the stick input which has to be removed to prevent the attitude from rising too far. As the speed decreases and the aircraft sinks, the uncontrolled angle of attack will increase, and it is no longer easy to judge the flight path as it diverges from the attitude. After touchdown the nose may have to be pushed down with a positive forward stick input. This is unnatural for the pilot and can make a precise landing very difficult to achieve [ibid].

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However, the PRD systems in the BAe EAP and the Eurofighter 2000 provide extremely precise approach path and touchdown point control with a natural technique. This is due to shaped feedback and feedforward filters to optimise the transient angle of attack and flight path responses, as discussed in Chapter 10, and to speed scheduled pseudo-static stability as described in §4.5.2. The Grumman X-29 with its original PRD system had a touchdown point scatter of several hundred metres, until it was similarly modified by NASA resulting in greatly improved precision.

At touchdown the natural tendency to hold a constant attitude is favourable if a bounce occurs. With a pseudo-static stability function, the stick must be held increasingly aft to maintain flight path with reducing speed in the flare, so that the nose begins to drop after touchdown if the stick is simply returned to neutral. In the take off rotation, the desired attitude is easily controlled. The stick is immediately in trim when released at the desired attitude. This is highly favourable in the event of an engine failure at lift-off, because the aircraft maintains a safe flight path while the pilot's attention is diverted.

4.4.2 Pitch attitude demand
The dynamics of attitude demand systems have been shown to be very suitable for the landing flare manoeuvre [Anon, 1991, Weingarten et al 1986, Berthe et al 1984, Mitchell et al 1990a]. It provides the conventional characteristics of direct control over the attitude and the normal nose drop on stick release after touchdown. A number of potential difficulties are noted in Hoh [1996], and it is entirely unsuitable for any task requiring more than minimum manoeuvres due to excessive trimming requirements and/or heavy stick forces.

The system is switched in automatically at a height of 15 metres on the Airbus A-320. It is not otherwise generally in use for fixed wing aircraft. Many examples have shown that excellent landing qualities are provided by other demand system types that require no last minute mode change. The attitude demand system is quite widely used in recent helicopters, however.

4.4.3 Normal acceleration demand
With normal acceleration demand, the stick input must be removed in the landing flare to avoid pitching up beyond the desired attitude. At the touchdown a bounce will cause the flight control system to pitch the nose down in response to the sensed upward acceleration, which will be considerably larger than the rather small load factor variations in flight just before touchdown (§4.5). In the take off rotation, there is no natural function of normal acceleration representing a desired lift-off attitude while the main wheels are still on the ground. There is in fact little to recommend such a system for the take off and it is extremely unlikely to be considered.

4.4.4 Angle of attack demand
With angle of attack demand, touchdown at constant attitude and rate of descent causes the AoA to be instantaneously reduced by an amount equal to the path angle, possibly causing the flight control system to pitch the nose up after a hard landing. In the take-off rotation, the attitude is equivalent to the desired angle of attack, for which there is a unique stick position. The trim setting choice is simpler (with a proportional plus integral system) than in conventional aircraft because it is independent of the configuration and centre of gravity. While this might well be set for the take-off speed, a much higher trim speed would probably be chosen if the forward acceleration is very high as in some combat aircraft (20 knots/second or more). In the case of engine failure at lift-off the aircraft might then be badly out of trim.
While the phugoid characteristics can be modified by speed and speed rate feedbacks, the effects may not be desirable. This was studied for conventional aircraft in the Calspan NT-33 [Chalk 1966]. Augmentation to a higher phugoid frequency by increasing the pitching moment due to airspeed caused excessive attitude response and stick force cues due to forward gusts or wind shear. The effects extended well towards the short period frequency. MCRuer [1986] notes that speed rate feedback to improve the damping may have equally severe effects. Rate damping can be effective but this could alter the nature of the response towards the attitude types.

Improving the phugoid relative damping beyond about 0.15 had little influence on handling. The biggest improvement in approach handling came from increasing the short period mode natural frequency from 1.5 to 2.5 radians per second, quickening the flight path angle changes. Even the former value is nearly twice the minimum of 0.85 radians per second permitted for satisfactory handling in the military specifications, e.g. Anon [1969]. As discussed elsewhere in these pages, satisfaction of specification requirements is not a guarantee of good handling. The issue of the flight path response is discussed in Chapter 7.

An in-flight simulation study of large aircraft handling in the approach and flare [Weingarten and Chalk, 1981] found that the touchdown control with an angle of attack demand system was inferior in the presence of turbulence to a PRD system, due to the inherent gust disturbance rejection and attitude stability of the latter type that provided precise and easy control. A conclusion of Gautrey [1998] is that the major improvement in poor aircraft handling qualities comes from increasing the pitch damping and stiffness, both hallmarks of the PRD system.

### 4.4.5 On-ground behaviour

Since there can be no aircraft response to the pilot's input until the rotation speed is reached, any integrators must be inhibited while the aircraft is on the ground and be activated only when the weight is off the nosewheel or the mainwheels. Similarly, proportional command paths must not be allowed to saturate excessively before response feedbacks can be generated. Generally it will be necessary to devise special "on-ground" control laws and to ensure a smooth transition between them and the in-flight laws by careful choice of fading and blending functions. Examples of some potential difficulties and solutions in the F-16 fighter are given in Bakker [1982], such as over-rotation, tail strikes, and abrupt pitch up on mainwheel switch operation in nose-high slowdowns.

During the rotation phases, a new instability mode is sometimes encountered as the motion takes place around the wheel contact with the ground and not the usual centre of gravity. This was seen in an early Shuttle Orbiter landing in the form of a pilot induced oscillation after touchdown [Gilbert 1983]. Since such a mode would be encountered in high speed taxi trials, it is important to have searched for it in simulation. This can only be successful if the undercarriage and ground effects are accurately modelled, of course.

### 4.5 Combined demand systems

None of the basic manoeuvre demand systems on its own has completely favourable qualities. These can be unconventional in both favourable and unfavourable ways. Most FBW aircraft employ some combination of demand parameters, attempting to combine the elements which are optimum for specific flight regimes and requirements. This needs a clear appreciation of the factors which lead to good handling, discussed in Chapter 7.
An early stability augmentation system on a number of aircraft used a feedback of normal acceleration combined with pitch rate. With retention of the conventional direct control of angle of attack this modified the short period oscillation but did not constitute an \((n + k\alpha)\) demand system. Tobie et al [1966] devised the \(C^*\) handling criterion based on the additive shape of the pitch rate and normal acceleration time responses. The criterion adds together the pitch rate in radian units and the normal acceleration in \(n\) units scaled to equal magnitudes at 400 miles per hour true speed. Their total therefore contains more of pitch rate at low speeds, where pilots are concerned with attitude control, and more of normal acceleration at high speeds where pilots are concerned with structural limits. The criterion was not generally successful, but its name lives on in the \((n + k\alpha)\) demand systems of the Airbus A-320 and the Boeing B-777.

In the linear lift slope regime, the short period dynamic response of the different systems to the pilot’s control input can be made essentially identical to each other. The basic dynamic angle of attack deficiency of an unstable aircraft can be restored by any of the three equivalent feedbacks shown in Figure 4-4, each of which may use the same feedback gain \(K_a\). It is possible to utilise any or all of them together. The principle can be extended to incorporate pitch rate, normal acceleration and angle of attack demand modes in proportions appropriate to the task, while maintaining effectively constant short period dynamics.

Switching of demand modes for different flight phases can be desirable, as can modifying their dynamics. In the flight path demand research of Nicholas et al [1991], the primary modes were of the flight path angle rate type, but these were switched to pure pitch rate modes at a wheel height of 100 feet for the final approach and landing. This permitted the pilots to make landings “which were good for a simulator” (see Chapter 5). Pilot ratings for the PRD back-up systems were Level 1 for low turbulence and Level 2 for high turbulence, probably due to the lack of precise path holding capability as noted earlier. In the flight path demand system of Verspay et al [1996], optimum attitude control was achieved by a stick feed-forward to the elevator as defined in Gibson [1995c] and Chapter 10.

4.5.1 Static stability
A PRD system has many excellent qualities, requiring no trimming with speed variation in most of the flight envelope or in climbing, diving and inverted flight, greatly reducing the pilot’s general workload in both combat and transport aircraft. Its basic deficiencies have also been noted. Its lack of exact flight path holding ability is not especially intrusive in combat aircraft and merely requires a little more attention from the pilot, but for transport aircraft some improvement may be necessary.

Particularly below the minimum drag speed, there is no stick force cue to alert the pilot to an inadvertent drift down to low speeds and high angle of attack. Neutral stability at low speeds may be made more acceptable by an over-riding stall protection scheme which prevents the maximum safe AoA from being exceeded. This was the solution adopted for the Airbus A-320, the \(C^*\) law having no inherent static stability.

A combined \((q + k\alpha)\) system, that is a pitch rate plus a proportion of AoA demand, provides static stability and resistance to inadvertent stall. Normal acceleration demand systems may be enhanced by similar means. However, the original \((n + k\alpha)\) landing control mode of the General Dynamics F-16 acted effectively only as an AoA demand due to the small load factor variations [Milam et al 1982, Bakker 1982]. It was generally agreed that the approach handling was unsatisfactory and precise attitude control was not possible. These problems became worse after some years of operation when the enlarged tail was adopted.
A flight research investigation led to the choice of a PRD law, giving precise attitude control. Because some static stability was necessary at very low speed, this was changed to a \((q + ka)\) system, but this degraded the precision of the PRD. Alleviation of this required many test flights and modifications to the AoA feedback gain and to the AoA value above which it became active. The \((q + ka)\) system worked well on the BAE FBW Jaguar, however. This had an exceptionally crisp pitch response in the landing condition.

4.5.2 Pseudo static stability

In the development of the PRD control laws for the unstable BAE EAP in the early 1980s, it was necessary to devise a means to provide an equivalent of low speed static stability without using angle of attack, unavailable for use until later versions of the FCS after the sensors had been calibrated in flight. The author noted that the major practical effect of conventional static stability lies in the “stable” movement of the stick to maintain changes in speed, and that this is easily generated by an increasingly nose down pitch rate command as speed decreases, in parallel with the pilot’s input path. This proved to be extremely successful [McCuish and Caldwell, 1996]. When the use of angle of attack became possible, and was incorporated in the wheels-up laws to provide low speed static stability in slow-downs to the stall, the pilots’ preference for the speed-related pseudo-static form in the landing approach was so marked that it was retained for wheels down conditions.

Angle of attack is not explicitly controlled, but the nose will drop automatically in a slow-down, giving the conventional speed loss cue to the pilot as well as providing some protection against a stall. The landing flare behaviour becomes sufficiently like the conventional to enable precise touchdown control to be exercised. Interestingly, this system generates a phugoid-like mode, resulting not from the normal aerodynamic forces but from the speed dependent variations in pitching demand (see “Wind shear effects” below and Figure 4-5). If there is a rapid speed loss from a steep climb, a pseudo-static system will cause a substantial pitch down but there will be no explicit maintenance of angle of attack. The system will also cause recovery from the ensuing dive to begin but the aircraft will not recover to level flight without a pilot input once the speed has increased past the upper limit of the system.

4.5.3 Manoeuvre limiting

Where precise limiting of normal acceleration or angle of attack is desired for the provision of so-called “carefree handling” in a combat aircraft without restraint on the flight attitudes, clearly the best systems would be normal acceleration and angle of attack demand, respectively. They are easily combined with a basic PRD system by blending functions which transfer control from one to the other as the stick input and manoeuvre amplitude increase, as was the case in the British Aerospace EAP [McCuish and Caldwell, 1996]. Transition between the regimes was completely unnoticed by the pilots. The carefree aspect was achieved by the scaling of full stick inputs to demand the limiting manoeuvres.

Manoeuvre limiting for a transport aircraft does not need such an explicit control of normal acceleration in turns, as limiting can be performed implicitly by automatic turn rate with bank angle limits, for example. However, pull-up protection may be considered necessary.

4.5.4 Trim enhancement

Most fly by wire aircraft are equipped with a reduced size stick and simple feel system such as a mechanical spring and damper. If a stick trimming system is required, it may be only needed for the limited speed range of the landing approach. Typically, to avoid undue mechanical complication, the trim switch will signal the manoeuvre demand system directly instead of switching
on an electro-mechanical actuator to alter the feel spring datum (Chapter 6). In the pseudo-static stability system discussed above, the trim input counters the speed-dependent nose down command so that the pilot can relax the stick force to zero at the desired speed. The correct trim can be preselected on a simple indicator since there is a unique relation between trim speed and the trimmer output.

If the trimmer is left active in other flight regimes, however, pilot instincts to remain constantly in trim will lead to attempts to use it inappropriately. This could happen as a result of transient disturbances due to transonic pitching moment changes or release of a weapon store, for example. Although the pitch integrator would eventually restore a steady trim state, it may act too slowly to satisfy the pilot. Use of the trimmer to assist then leaves the aircraft out of trim shortly after. This urge to trim can be accommodated by modifying the trimmer function to assist the integrator by sending a signal directly to move the tailplane more quickly to the correct position. There is no effect on the steady trim state since this is ultimately controlled by the integrator.

4.5.5 Autothrottles

Autothrottle control of engines has long been an essential part of autopilot systems for airliners, maintaining constant speed in the cruise and offering precise speed control with low workload in the more demanding approach and landing. Combat aircraft generally have had simpler autopilot functions, often without an autothrottle. Research by Mooij et al [1976, 1978] has suggested that neutral speed stability as expressed by zero stick force to hold speed changes, found in PRD control systems for example, may not have significant consequences for landing approaches without an autothrottle when operating at or above the minimum drag speed. However, the flight path instability that occurs below the minimum drag speed (see Chapter 3) can be stabilised by an autothrottle, which maintains a constant speed while the path is changed by elevator operation. It is necessary to be able to over-ride the autothrottle easily for the landing.

The McDonnell Douglas S/MTD F-15 demonstrator [Moorhouse et al, 1991] had pitch rate, angle of attack and normal acceleration pitch command parameters and a direct lift function available. The mode eventually selected for landing on a runway 450 metres long by 15 metres wide, after touch down in a box 18 metres long by 6 metres wide, was PRD with a speed command autothrottle. The latter was extremely precise, being effected by thrust deflection vanes with none of the usual engine thrust lag. The flight path could be easily adjusted with pitch inputs while the speed remained constant, a combination which was found to function very effectively as a flight path angle rate command system although no explicit path feedbacks were used. The final test of this aircraft was a successful night landing on the defined runway space, with no ground lighting or external flight path cues and using only on-board guidance.

4.5.6 Integrated flight and power control systems

There is an increasing interest in the further integration of flight and power control systems as the demands for enhanced flight envelope usage grow in the short take off and landing regimes. They will become vital for a future generation of combat aircraft where the engine power and thrust vector configurations cannot be controlled adequately by the pilot for these tasks, necessitating the use of fly by wire techniques that also enable complex integration.

In the McDonnell Douglas C-17 military transport [Kendall, 1996], the approach configuration flaps are externally blown by engine exhaust and produce large changes in lift with thrust. Initially the throttles caused unacceptable changes in airspeed, but a solution was developed with optimised flap settings and extended spoilers, the latter modulated by thrust demands. A crisp flight path response to throttle commands is achieved with little airspeed variation at a constant
attitude, while the airspeed is effectively controlled by the stick pitch attitude commands with little flight path variation. Although operation is just below the minimum drag speed, very precise slow speed approaches are achieved with low workload.

More complex integration in the NASA QSRA (Quiet Short take off and landing Research Aircraft) is described by Franklin et al [1986, 1996]. Upper surface blown flaps and ailerons, spoilers, elevator and thrust are controlled through a system of three-variable look-up tables of the aerodynamic force and thrust coefficients. These provide independent control of flight path by the stick and airspeed by a speed demand controller, generating the forces required to satisfy the commanded values of the axial and normal velocity vector components. The methods have been applied in design studies for STOVL (Short Take Off and Vertical Landing) fighters.

In the VAAC (Vectored thrust Aircraft Advanced flight Control), a modified Harrier research aircraft [Shanks et al, 1994], the aerodynamic, thrust and nozzle angle controls are closely integrated with a fly by wire system. The right hand centre stick controls pitch rate, flight path angle and vertical velocity as the airspeed progressively reduces from normal flight through transition to the vertical landing regime, the modes blending seamlessly with no pilot selection. A single left hand inceptor controls forward speed as a thrust lever for normal flight and as a speed demand controller at transition and landing speeds. All the necessary conversion of forces and moments to satisfy the velocity vector demands through nozzle and thrust variations is performed automatically. In a further simplification, the speed control function was transferred to a switch on the stick, allowing complete control with only one hand.

It was found that pilots with no Harrier experience, and even a non-pilot, could immediately perform precise decelerations and vertical landings in the VAAC, a task requiring considerable training and practice in normal Harriers even for skilled pilots. Despite their initial scepticism, Harrier pilots also found both the two and one inceptor front side technique very easy and rated the system as significantly superior to the standard controls.

4.5.7 Wind shear effects
Chapter 3 discussed the potential for coupling with the downburst phenomenon in thunderstorm conditions, in which the headwind-tailwind cycle of the downburst can excite a large phugoid response. The existence of this coupling in fly by wire aircraft depends on the form of manoeuvre demand system adopted.

Figure 4-5 shows a range of pitch frequency response amplitudes to forcing by horizontal wind shear or turbulence for the FBW Jaguar research aircraft, which was flown with several versions of a pitch rate demand system. The conventional Jaguar response and that of the proportional-plus-integral \((q + ka)\) demand system are very similar, with the usual large phugoid amplitude ratio. With a pure PRD system there is no phugoid and the general low frequency amplitude gain is very considerably reduced. The addition to the latter of a pseudo-static stability mode, of the speed to pitch rate demand type (§4.5.2), introduces a phugoid-like response but with a peak amplitude ratio that was only about 10% of the traditional version. Such a response would greatly reduce the influence of a down burst penetration on the aircraft attitude.

4.6 Lateral-directional handling

The advent of fly by wire technology has affected lateral and directional axes handling qualities as profoundly as the pitch axis. The capability can provide consistent and precise roll behaviour
from small to gross manoeuvres, with no limitations on allowable control inputs even at low to negative normal accelerations. Sideslip can be contained in any rolling manoeuvre to small values with no pilot inputs to the rudder, or to within structurally safe limits at all speeds when full pedal inputs were applied.

These results are obtained with control law structures that are in the main basically similar to conventional systems. Instead of introducing new and unfamiliar response and control modes, the technology simply extended the existing augmentation methods to the ultimate limit of full authority. The differences lie in the use of gain variations and shaping in the forward command paths, with multiple cross-axis decoupling paths or interconnects to optimise roll coordination and to counteract the inertial cross-coupling factors. These have transformed the controllability and safety of combat aircraft in the violent manoeuvres often required of such types. The need for the difficult to observe limitations such as "half stick rolls only at 0-5g or lower" which were typical of older generations of aircraft has been eliminated.

Very high damping of the roll mode is possible without the loss in roll performance that would result only from aerodynamic damping, by balancing the feedback by increased command gain. This is equivalent to providing aileron travels of several times the physically possible limit. Because larger than normal aileron angles are not required to achieve the desired roll rates, the actual angles applied are only increased from normal transiently if at all. Turbulence-induced disturbances are heavily attenuated. However, the roll acceleration following large stick inputs can be excessive for the pilot. This is readily alleviated by command path filtering that effectively provides dual roll mode time constants - a small regulated one for high disturbance rejection and a larger controlled one for smooth roll handling. This technique is elaborated in Chapter 10.

The directional stability can be heavily augmented with high levels of damping by yaw rate feedback and suppression of sideslip disturbances by sideslip or lateral acceleration feedback. The latter can actually be counterproductive with the effect of increasing heading oscillations in turbulence, a very irritating feature for a pilot. However, several modern aircraft have relatively low or even negative directional stability at flight envelope extremes such as high Mach number or angle of attack. Augmentation is then necessary, but only to a sufficient level. Suppression of sideslip in commanded manoeuvres has been best achieved by attention to the coordination cross-coupling paths, given a basically satisfactory level of natural or augmented stability.

While the rather accurate control achieved by a proportional plus integral system is often necessary in the pitch axis, this is seldom required in the roll axis. Completely adequate control of roll rates can be achieved by simple forward path proportional gain scheduling. Integrator control is seldom used, only three examples being known to the author. The Grumman X-29 used it at low angles of attack only, and the Rockwell-DASA X-31A used it at high angles of attack only. The reasons are cited as high angle of attack problems, though different ones [Clarke et al 1996, Beh et al 1996]. The McDonnell Douglas F-18 used it in the early control law versions, but it was abandoned because of serious controllability problems [Walker et al 1982]. While this was associated with many failures in design process and could probably have been overcome given a better understanding of handling qualities, no need for integral roll control has been found in a number of other successful and highly agile aircraft.

4.7 Summary

This chapter has discussed the variety of ways in which aircraft with manoeuvre demand flight control systems may differ in their response from conventional types. These differences have
led to unexpectedly poor handling in some aircraft. They are significant in both the long term and short term response regimes equivalent to the conventional phugoid and short period response. An appreciation of the differences is essential when attempting to design good handling qualities in fly by wire systems, so that those features of conventional handling that are desired by pilots may be retained, emulated or enhanced. Identification of these features, their measurement and application to the design process are discussed in Chapters 7 and 10.
$\delta_{e_t} =$ pilot's stick input $\quad K_\delta =$ stick scaling gain $\quad q_c =$ demanded pitch rate
$\varepsilon_q =$ pitch rate error $\quad K_q, T_q =$ proportional gains $\quad 1/s =$ integrator
$\delta_e =$ elevator angle $\quad q(s)/\delta_e(s) =$ pitch rate transfer function $\quad q =$ pitch rate

Figure 4-1 Generic Pitch Rate Demand system (PRD)

Figure 4-2 Generic effects of PRD system on flight path responses, based on figures from [Myers et al, 1987]
Figure 4-3 Sketches of typical responses at landing speed of a PRD system, adapted from [Weingarten, 1981]
(a) Angle of attack feedback
\[ \delta_e = \text{pitch control} \quad \alpha = \text{angle of attack} \quad K_a = \text{angle of attack feedback gain} \]
\[ \alpha(s)/\delta_e(s) = \text{angle of attack dynamics} \]

(b) Normal acceleration feedback
\[ n = \text{normal acceleration} \quad m = \text{aircraft mass} \quad g = \text{acceleration due to gravity} \]
\[ \rho = \text{air density} \quad V = \text{true speed} \quad S = \text{wing area} \]
\[ C_{Na} = \text{normal force coefficient} \quad n(s)/\delta_e(s) = \text{normal acceleration dynamics} \]

(c) Pitch attitude feedback
\[ \theta = \text{pitch angle} \quad \theta(s)/\delta_e(s) = \text{pitch angle dynamics} \]
\[ T_{\theta_2} = \frac{m}{\sqrt{\rho V^2 SC_{Na}}} \]

Figure 4-4 Equivalent methods for augmenting or approximating angle of attack stability in short period dynamics
Figure 4-5 Low speed pitch responses to horizontal gust excitation of the conventional Sepecat/BAe Jaguar and the BAe FBW Jaguar

Artificial phugoid with pseudo-static stability by $\Delta q$ demand with speed
(Chapter 4, para. 4.5.2)

FBW Jaguar, unstable configuration

$\frac{\theta}{\mu_{GUST}}$ dB
[deg/m/sec]

0 10 20

0 0.05 0.1 0.5
Gust frequency [Hz]

Conventional Jaguar

$\Delta q$ demand

$\Delta q$ demand

$\Delta (q + kq)$ demand

Phugoid

Short period
Chapter 5
Basic elements of handling qualities 3: The human pilot

5.1 Introduction

With the power to shape the airframe control response and disturbance rejection by modern FCS methods to suit the needs of the pilot, today’s designer ought to have an easier task than in the past. For example, it is possible to optimise the two functions separately, which could not easily be done when aerodynamic and inertial qualities were dominant. Knowledge of the effect on pilot workload and task performance of various features of aircraft behaviour allows the control laws to be selected to emphasise the most favourable. The conventional modes of control of pitch, roll and yaw can be improved to an unprecedented degree by FBW technology through the so-called superaugmentation, manoeuvre limiting and spin prevention methods.

Such techniques can offer high precision handling, stabilisation of configurations with extreme aerodynamic instability, and exploitation of the full aerodynamic performance and manoeuvre capability with little risk of loss of control. They bring an unexpected result: the less that pilots need to intervene to maintain the flight course by continual corrections, normal in a conventional aircraft, the more critical they become of the control response characteristics. This is probably due to the fact that greater attention can be given to those qualities which are optimum for specific tasks rather than to the general task of maintaining control over the aircraft. However, this has been a continuing process since the advent of the first elementary stability augmentation systems. The better the handling became, the more pilots objected to otherwise minor irritations such as trimming imperfections or continual atmospheric disturbances, as well as to more fundamental imperfections in the pilot-aircraft coupling interactions.

For such reasons the rather wide ranges of handling qualities permitted by traditional requirements, partly a consequence of the limited ability to modify the behaviour by earlier control technologies as well as of pilot adaptability, must be supplemented with new criteria. There is still only a small data base of widely available handling information from modern FBW aircraft or their representation by in-flight simulations. Nevertheless, these sources have enabled development of new methods of describing handling qualities which permit direct comparison with older low order characteristics. The fresh light cast on the latter as a result of these studies has illuminated several qualities which were previously ignored or only accounted for implicitly, but which are important to the pilot as discussed in Chapter 7.

Although FBW technology has had a major impact through the Concord, the later Airbus series, the Boeing 777, and large transport aircraft such as the McDonnell Douglas C-17, the Antonov An-124 and An-225, it has as yet had relatively little influence on civil aircraft in general. It can be expected that the trend to FBW will spread as cost, operational and traffic control benefits appear. Military specifications contain sections dealing with large aircraft typical of airliners, but the data base for establishing new criteria suitable for highly augmented control systems is rather sparse. The outstanding example of a large non-military aircraft with an advanced FBW control system is the Space Shuttle, which has demonstrated excellent handling for the unusual parts of its flight tasks, but which also requires exceptionally high standards of pilot skill and training to land safely because of unsuitable handling for this universal task. It can be taken that the general principles developed for military aircraft tasks also apply to civil aircraft. The pilots are the same and have the same inherent behaviour and sensing abilities.
5.2 Pilot models

Traditional handling qualities specifications attempt to define the desired response characteristics by placing limits on values of mode parameters such as short period frequency and damping, static, manoeuvre and speed stability effects, stick forces, roll rates, and so on, covering every aspect of piloted control of an aircraft as in Figure 2-4. These all specify the aircraft behaviour according to past experience, custom and practice, and the results of many experiments in flight and in simulators. Such specifications cannot by themselves define the form of the control laws. Because the nominally satisfactory range of each mode parameter is usually derived from studies of variations it itself alone, a combination of parameters at their limits is likely to produce poor handling. To design satisfactory control laws requires some understanding of the interactions between the pilot’s mental and physiological processes and the physical aspects of the aircraft’s responses to control demands.

Descriptive models of the pilot can take many forms. One may be a verbal description of the actions undertaken in order to perform a given task, or a mathematical model of the pilot as an element in a feedback loop. The real pilot is used in simulations in which the aircraft is modelled by a computer and the pilot’s view of the world outside is modelled by a visual display system. The FCS designer must check a new design against all these before the aircraft flies.

It is difficult to measure cognition. Human adaptability enables pilots to learn to do what they have to with as little conscious thought as possible. This state may be reached quite quickly or it may take a considerable amount of practice. For a given performance the subjective workload as reported by pilots may be low or high, and their opinions of the handling will range from good to bad accordingly. The essence of good handling is that the response is predictable, the goal of handling qualities design criteria. This is encapsulated by a phrase sometimes used by pilots to describe desirable handling: “The nose follows the stick”.

5.2.1 The pilot as a sensor

“Flying by the seat of the pants” is a well-worn phrase but it is almost meaningless. If this were the only source of information available to the pilot, flight would be impossible. The dominant sensors possessed by the human are the visual, tactile and vestibular systems, by which motions and attitudes relative to the environment can be detected. Although falling outside the scope of this thesis, the subject ought to be of interest to FCS control law designers. An understanding of the sensing abilities of the pilot helps to focus attention on the particular features of aircraft response characteristics which are of primary importance to him.

The characteristics of these sensors are the natural result of evolution to operate in an essentially 1g environment for the purposes of keeping the human balanced upright, the visual environment stable and keeping precise track of targets. The human can physically generate only quite low linear accelerations and velocities, but high angular accelerations and visual target rates are possible through rapid head and eye rotation. In the unnatural flight environment the capabilities of these sensing systems result in an acute consciousness of attitude displacements, rates and angular accelerations, but only a rather generalised and sometimes misleading sense of linear velocity and acceleration.

The eyes are by far the most important sensors the pilot has. It would be possible to maintain control with these alone and a view of the outside world, as can be demonstrated in any fixed-base flight simulator. The fundamental method of flying any aircraft is to adjust its attitude against a visible background, which would be the earth’s horizon if it can be seen, the ground
beneath in poor visibility from which an experienced pilot can acquire sufficient cues for normal flight with limited manoeuvring, or an attitude reference instrument. Without attitude cues, control can last only for a very brief period. The only additional item of information usually necessary for routine flight is airspeed, although even this can be dispensed with in some simple low performance aircraft. The skilled pilot can infer a level attitude from vertical velocity and turn rate information alone, though only within limited departures from straight and level flight.

The vestibular system contains the semi-circular canals which detect angular accelerations, and the otoliths which detect specific force\(^7\). Located in the inner ear, their signals are processed by the central nervous system, which interprets them as angular and linear velocities over a substantial range of frequencies, and also the otoliths provide orientation to the gravitational vertical. Neither sensor can maintain a perception of steady velocity for more than a few seconds, and when the velocity ceases it is then perceived as motion in the opposite direction. Both the canals and otoliths are subject to thresholds in detection of accelerations and a latency time before the sensation becomes apparent, e.g. Meiry [1966], Hosman and Van der Vaart [1980].

Direct eye tracking provides the basic orientation reference. The semi-circular canals and the neck proprioceptors, the sense organs which measure the head rotation on the trunk, provide rate feedback to the eye control system, producing compensatory eye movements in the presence of head rotation to maintain accurate tracking. Despite the power of this compensation, it is not a substitute for an external visual reference, without which the eye can not be held for long in a fixed direction.

Manoeuvring flight requires an increase in normal acceleration which can be sensed by the otolith system and by the tactile sensors which measure the pressure on the parts of the anatomy supporting the body. Combat pilots learn to estimate the level of normal acceleration quite closely, though sometimes at the cost of cues such as pain in the lower arms at 9 g. Despite this, most people will scarcely notice the g at all in airliners turning typically at 30 or 40 degrees of bank with 15 to 30% increase in g. Even after their weightless orbits, a Shuttle orbiter crew did not notice the variations in g in manoeuvre tests after re-entry involving tests between 0-4 and 1-8 g [Mattingly et al., 1982]. Control of a co-ordinated level turn is achieved not by direct application of the required g but by whether the aircraft tends to climb or descend after the desired bank angle is achieved, requiring respectively a decrease or increase in applied g.

In striking contrast, the performance of the semi-circular canals is an order better. Many passengers will be familiar with the sensation of rolling and turning induced by even the modest manoeuvres of an airliner, followed by an apparent ability to "see" the actual motion visually even when nothing is visible outside and the g sensation is negligible. Known as "the oculogyral illusion", this is the result of the angular acceleration and rate sensing compensation to the visual system, although it is impossible to determine the actual bank angle.

5.2.2 Pilot perceptions
Although the flight path is altered by normal acceleration, this is produced by angle of attack changes not directly perceptible to the pilot. Aspects of the normal acceleration transient response have to be respected to achieve satisfactory short term control of flight path changes, but they give little clue to longer term changes. From the pilot's point of view, these are generally effected by attitude rates and displacements. Most handling qualities are to a great extent perceived through the qualities of the attitude response of an aircraft. These are clearly very appro-

\(^7\) Specific force is defined as the force per unit of mass, and as a consequence can be regarded as the vectorial sum of linear accelerations and forces due to gravity.
praited for the tasks which require precision pointing of the airframe. It is equally true for tasks with other controlled variables such as flight path, represented by displacements of a head-up display symbol, or HUD.

In flight, an abnormal condition for the human, the senses reinforce each other so long as well co-ordinated visual flight is possible. Regardless of the attitude, if the normal acceleration remains vertical with respect to the airframe then all feels well. Lateral specific force gives a feeling of falling over sideways, extremely uncomfortable to the human. When the outside world or its representation is not visible, the short-term nature of the vestibular sensors ensures that the pilot will lose control after a very brief period. Flight in instrument conditions with only limited visual cues is made more difficult if the handling is unpredictable and badly co-ordinated, causing potential conflict of the senses. A sudden loss of visual cues during the course of manoeuvres can lead to complete disorientation and loss of control.

Earlier studies up to the 1970s had provided a good knowledge of the visual and vestibular system dynamic characteristics, but there was little understanding of how the vestibular system contributed to the pilot’s perception of motion. This became of great importance to the growing dependence of the airline industry on ground-based simulation for flight crew training and even if possible for zero flight time type-conversion. Hosman [1996] and van der Vaart [1992] have described the work conducted over the past two decades at the Delft University of Technology to establish the independent and combined contributions of the central and peripheral visual system and the vestibular motion sensing.

Handling qualities are linked with the visual system in another way. In a complex modern aircraft there is a mass of information about the on-board systems, navigation and flight conditions, and so on, which are presented to the pilot by an array of instruments and displays. In a military combat aircraft there is even more information to be absorbed from weapon and attack displays. Under high workload conditions the ability to absorb this information becomes less and less. This is evidenced by a tendency to fixate on the most important instrument and for the scan pattern and rate to become disordered, e.g. Senders [1983].

Experienced pilots are able to maintain a more normal scan, an ability acquired from hundreds of hours of training and which distinguishes them from the inexperienced. A workload that is too high will lead to a collapse in performance at any skill level, and indeed attempts have been made to measure workload from observation of pilots’ visual scanning patterns. Suchomel [1996] gives an overview of a current programme effort in the USA to develop automatic rating techniques to measure flying qualities directly from physiological measurements in flight.

If the FCS provides excellent handling qualities and protection from over stressing or loss of control, the pilot is relieved from the task of maintaining continual observation of the appropriate instruments and limits, can therefore absorb more effectively the details of the task displays, and can thus devote the maximum attention to the task in hand for greater efficiency. This is the absolute justification for optimisation of handling qualities.

5.2.3 The pilot as a servo element
Feedback control of closed loop systems, in which an output variable is controlled by comparison of the error between its current value and the input demand, is analysed by the methods of general servo theory. In the 1950’s the possibility of analysing the handling qualities of the pilot-airframe system by these methods was studied widely, and was well established by the 1960s. The hypothesis was that, if human behaviour could be mathematically described in the

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same way as a closed loop system element, it would allow handling qualities and task performance to be analysed to give a better understanding of pilot opinion and performance.

The method used to obtain the mathematical models of pilot behaviour was to record his response to the task of tracking a single axis target in pitch or roll in a flight simulator, which was often just a chair with a simple stick in front of a cathode ray tube. Pilot describing functions were derived that most closely matched the recorded response. Many excellent descriptions of this work are available, e.g. McRuer et al [1957, 1960, 1965, 1988]. Studies continued further, and have been extended by the application of modern optimal control model theory, e.g. Baron and Kleinman [1969]. The great complexity of the resulting models have generally excluded their use in the design of flight control systems, and they seem to have been employed mainly in after-the-event analysis of pilot behaviour. For all practical purposes, the FBW control law designer will find the models of the 1960s sufficient.

A typical tracking loop closed by the pilot is shown in Figure 5-1(a). The input is the attitude required to minimise the target error. The error signal to which the pilot responds is the difference between the forcing function of the target motion \( \theta_c \) and the controlled aircraft motion \( \theta \), presented to the pilot as \( \theta_e \) in the tracking display. In the original experiments, only the error itself was displayed to the pilot, often as a spot on a cathode ray tube driven by random or random appearing inputs to prevent prediction of its motion, which the pilot attempted to keep centred in the display. This is the essential feature of "compensatory" tracking. There is also always to be found a portion of the pilots activity, usually small, that is not linearly correlated with the forcing function. Known as the remnant, it can be neglected for the purpose of linear modelling and studies of generic handling qualities.

In the majority of real flight situations, additional information is available to the pilot, such as the actual position and motions of the target. Knowledge of the output allows the pilot to improve the tracking performance by compensating for the aircraft dynamics by the feed forward path, the so-called precognitive open loop input. Relatively little study has been made of this, as the additional path dynamics are difficult to measure experimentally. The compensatory closed loop remains active as a vernier adjustment or to improve the closed loop stability of the system [McRuer 1966]. Only the compensatory dynamics are considered here.

5.2.4 The general pilot describing function
The nominal "aircraft dynamics" in the experiments were very simple, \( K, K/\dot{s} \) or \( K/s^2 \) being typical forms, i.e. constant gain, rate or acceleration, sometimes with an added single first order time constant. The describing function \( Y_p \) of the "standard pilot model" derived from these experiments for compensatory control tasks is given by McRuer et al [1960]:

\[
Y_p = \frac{K_p e^{-\tau_s}}{(T_1 s + 1)(T_2 s + 1)}
\]

(5.1)

- The pilot gain \( K_p \) is expressed with the same unit dimensions as the aircraft frequency response, for example as pounds of control stick force applied per degree attitude error and degrees of aircraft attitude response per pound of control force, respectively. Any other units may be used such as stick displacement, from a full unit stick input down to a millimetre. So long as the same units are used the total gain around the loop is non-dimensional, as is required for the analysis.
- The reaction time delay \( r \) represents the finite time which elapses between the human's decision to act and the physical act itself. It is always present when tracking a random-seeming target, typical values being around 0.2 to 0.3 seconds. Its effect is to add phase lag to the pilot's output proportional to the frequency but it does not alter the amplitude. There is also a small time constant \( T_n \) representing neuromuscular lags, often given in a second order form. It is significant for activity in the 2 to 3 Hz region, and can be neglected for the purpose of most handling qualities design.

- The lead time constant \( T_L \) represents the pilot's ability to predict and advance a control action, required when the aircraft response is slow or unresponsive so that it has to be speeded up by anticipatory inputs. It can also be utilised to reduce the pilot's inherent time delay or neuromuscular lag although it is then not available for its usual purpose. The amount of lead which can be generated easily is quite small, about a one second time constant being the largest without serious degradation of pilot opinion.

- The lag time constant \( T_L \) represents the pilot's ability to apply smooth inputs to overcome an oscillatory aircraft response. The latter may be the result either of low damping or of an excessively "jumpy" response which makes it difficult to achieve a steady closed loop tracking solution, both requiring some attenuation of the activity. Often the lag is associated with a smaller lead needed to compensate for the extra phase lag at higher frequencies caused by smoothing. Quite large values of lag time constant can be employed, up to several seconds, but the aircraft response demanding this is would have poor handling which is easily avoided in a FBW system.

The gain, lag and lead comprise the pilot equalisation or compensation efforts. These were not assumed \textit{a priori}, but were found to be the consequence of pilots' efforts to minimise the rms tracking error, and subject to adjustment rules to obtain the following briefly stated properties:

- The lead and lag are adjusted to achieve a -20 dB/decade slope of the combined \( Y_pY_c \) response over a range of frequencies near to the crossover frequency.
- The gain is adjusted to achieve a desired crossover frequency (that is where the gain of \( Y_pY_c \) is unity).
- The resulting open loop response is known as the "crossover model":

\[
Y_pY_c = \frac{\omega_c e^{-rt}}{s} \tag{5.2}
\]

where \( \omega_c \) is the crossover frequency.

\[5.2.5\] The "crossover" model and the K/s goal

Figure 5-1(b) shows the nominal gain and phase angles of the crossover model. Ashkenas and McRuer [1962] found, based on data from Hall [1958], that pilot opinion is highest (or the workload is lowest) when no equalisation is required and the pilot can operate as a gain only, though the time delay remains. They thought that "perhaps this will ultimately become the basis for a standard "good" configuration". R. H. Smith \textit{et al} [1979] gave a "No tracking hypothesis", that optimum handling qualities requires minimum pilot closed loop control. When pilot compensation is not a factor or minimal compensation is required for the desired task performance, this leads to the desired Cooper-Harper rating of "satisfactory".

From this it follows that for \( Y_p = K_p e^{-rt} \), the optimum aircraft dynamics are

\[
Y_c = \frac{K_c}{s} \tag{5.3}
\]
and from Eq. 5.2

\[ \omega_c = K_p K_c \]  

(5.4)

Hence the open loop \( Y_p Y_c \) dynamics resemble a simple integrator with -20 dB decade gain slope and -90 degrees phase angle, but with the additional phase lag proportional to frequency due to the pilot's time delay. The aircraft dynamics most desired by the pilot are the rate-like "\( K/s \)".

Figure 5-2 outlines the idealised \( K/s \) response time history to step inputs, with a proportional rate response and an integrating attitude output. As a result the aircraft nose can be said to "follow the stick", a pilot comment of approval. In the practical case, the short period dynamics intrude at the start and end of the outline, but within reasonable limits the response can still be classified as "\( K/s \) - like". It is usually sufficient for the system response gain slope to be close to -20 dB/decade for a moderate range of frequencies around the crossover, at which the system phase lag will in general be close to 90 degrees. The point is discussed in Chapter 7, where it is also applied to flight path response. The introduction of the necessary equalisation directly into the control law structure to provide the desired \( K/S \)-like dynamics is discussed in Chapter 10.

5.2.6 Limitations to pilot equalisation

True aircraft dynamics are more complex than the simple models of these experiments. Hall [1958, 1963] tested a wide range of short period frequency and damping at a constant wing loading, lift slope and flight condition in a fixed base simulator. These results were analysed in McRuer [1960]. Hall found two well-delineated regions in the short period frequency and damping plane of the pitch attitude dynamics, Figure 5-3. In one region, almost all of the pilot input was linearly correlated with the task and could be matched by the standard pilot describing function model, Eq. (5.1). In the second region, a pilot equalisation model was required that was more complex than the standard describing function, with an extremely poor opinion rating.

In the upper linear region, the pilot equalisation changes from a lag/lead through none to a lead/lag as the short period oscillation changes from high frequency/low damping to low frequency/high damping. Typically the remnant (pilot activity not linearly correlated with the forcing function) comprised only some 5% of the total. In the lower non-linear region the pilot describing function had to be changed to a double lead/double lag at low frequency/low damping, and the uncorrelated remnant activity comprised some 50% of the total or even much more.

Ashkenas and McRuer [1962] noted that the remnant, while an experimental and analytical artefact, was a very useful device for the study of non-linear systems, but confidence in it required the linear describing function component to account for most of the total output. Their experimental results showed that this was so except for the most arduous control tasks. Nevertheless the pilot is usually "quasi-linear", on a smoothed on-average basis. Clearly this was not so in the lower region of Figure 5-3. This subject will be returned to in Chapters 8 and 9, with a more rational explanation of the observed pilot behaviour.

---

8 The expression "\( K/s \)" is universally known and understood in the handling qualities community as a description of a particular quasi-integrator handling qualities characteristic, and not as a mathematical abstraction. Hence "\( K/s \)" is used throughout this thesis even where "\( K/j\omega \)" would be strictly correct, as on Nichols and Bode frequency response plots for example.

Similarly, the use of \( T, \tau \) and \( \tau \) in this thesis reflects long established "custom and practice" in the field of handling qualities rather than strict mathematical consistency. Since to conform here to the latter would be inconsistent with the author's own published work and confusing to other practitioners, the author asks for the reader's indulgence in this matter.
5.2.7 Use of the Nichols plot

At this point it is necessary to introduce the Nichols plot, Figure 5-4, a format widely used in the following pages. It is not always well known to control engineers, but it has major advantages for assessing the handling qualities in analyses of pilot-aircraft closed loop situations. The open loop characteristics of the aircraft ($Y_c$) and pilot ($Y_p$) are plotted on the rectangular grid, combining the gain and phase in a single curve along which the frequency varies. The total open loop ($Y_pY_c$) is found by adding their gain and phase at each frequency point. Here no pilot equalisation is assumed, so that ($Y_p$) = $K_p$ and only the pilot gain is added. The stability margins and crossover frequency are obtained as usual. Closed loop gain and phase characteristics are read off directly from the curved grid.

The thick 0 dB closed loop line signifies a division into droop and resonance behaviour. The $Y_pY_c$ response sketched here has a closed loop gain $\theta/\theta_c$ of less than one at low frequencies, known as "droop". As the frequency increases the closed loop gain becomes greater than one and reaches a peak resonance amplitude before decreasing rapidly at higher frequencies. These dynamics are sketched in Bode plot form in Figure 5-5. Good tracking performance is characterised by a straight, closed loop unity gain (or the 0 dB curve in the Nichols plot) without droop or resonance, in which the output follows the input exactly for all input frequencies. It is clearly impossible to reduce the droop merely by raising the pilot gain because even a small increase would result in unacceptably large resonance for little improvement in droop. This can be seen from Figure 5-4 by moving the $Y_pY_c$ plot upwards. Improved tracking performance could be achieved only by including pilot phase equalisation in the form of lead and/or lag.

The desirability of the nominal $K/s$ aircraft dynamics can be seen in Figure 5-6, which sketches the basic crossover model of Figure 5-1(b) in Nichols plot form. Good closed loop tracking performance is achieved with a range of pilot gain, the crossover frequency being limited mainly by the pilot's time delay. This can be minimised by use of the otherwise unused spare lead capacity. As noted by McRuer [e.g. 1996], the tolerance to pilot compensation variations is of central and overriding importance as the governing principle in favourable pilot-vehicle dynamics.

The closed loop grid is frequently omitted as only the open loop response need be plotted for handling qualities purposes. The closed loop response can then be inferred, leading to simple criteria based on the "shape" of the open loop response alone. This is the basis of the author's methodology presented in Chapters 7 and 10.

5.2.8 Pilot gain sensitivity

It has long been recognised that the pilot gain required to perform precision tracking tasks is a dominant influence on the pilot's opinion of the handling qualities. Figure 5-7 summarises a result from McRuer et al. [1960], from the experiments by Hall [1958]. Typically there is an optimum aircraft gain at which the pilot can readily achieve a desired crossover frequency. The pilot can accommodate moderate variations in aircraft gain from this level by personal gain adjustments to maintain a constant crossover frequency with an acceptable penalty in workload. Larger gain variations exact increasing workload and worsening pilot opinion, leading ultimately to PIO problems for aircraft gains that are either too large or too small. However, the perception of sensitivity can also be influenced as much by the inceptor characteristics as by the aircraft dynamics, as discussed in Chapter 6.

An excellent example from recent development of the McDonnell Douglas C-17 military transport aircraft is given in Ilouputaife [1996]. The flight refuelling pitch handling qualities were
very unsatisfactory, and attempts to improve them by modifying the flight control laws were unsuccessful. Transformation from unsatisfactory to very satisfactory handling was finally achieved by an adjustment only of the pitch stick gain and hence the response sensitivity. An unexpected effect was that without any change to the throttle control, longitudinal positioning became much easier simply as a result of the improved pitch control. The reference notes earlier similar concerns with sensitivity in the Northrop fighter prototypes, the F-20 [Illoputaife, 1987] and the YF-17 [Gallagher, 1977], indicating a universality of pilot needs from small fighters to very large transport aircraft.

McRuer [1996] notes: "The determination of the optimum controlled gain is clearly a matter of supreme importance to assure a favourable pilot-aircraft interface, effective pilot-vehicle interactions, and an absence of PIO tendencies. ... the proper setting of controlled element gain has become a non-trivial development aspect on every new aircraft that introduces a new inceptor. In the absence of an extensive background of data for these there is no basis other than experiment to determine the optimum gains."

However, the data that have been published, coupled with a growing body of the author's inhouse experience, has provided satisfactory pre-flight selection of control gains for the series of fly-by-wire aircraft control laws on which this thesis material is based. The problem is related mostly to attitude rate control. The author has found the traditional stick force per g requirements to be of minor significance (Chapter 7).

5.3 The pilot in the simulator

Only a limited mention is made here of simulation technology. A recent and extremely comprehensive review of the subject is given by Baarspur [1990].

5.3.1 Fixed base simulators

The majority of simulators within the aircraft manufacturing companies and research establishments are fixed base, especially so for combat aircraft design. The pilot therefore has to assess the handling of an aircraft in very artificial circumstances. Although the pilot's visual sensing system is sufficient on its own to maintain control of a simulated aircraft, this task performance and opinion rating will be degraded without the information supplied in flight by the vestibular sensing systems. Without motion the only cues available are visual, supplied by displays of differing complexity and realism depending upon the purpose of the simulation, and tactile, from the forces on the stick and pedals generated by the artificial feel system. Audio cues can be added to represent engine noise, or seat vibration may be used to represent buffet, and so on.

The quality of the visual display determines how much of the missing information can be inferred. A single moving dot and horizon line convey the absolute minimum amount needed for random error tracking assessment but much more additional detail is needed to represent normal flight tasks. Modern facilities can provide extremely powerful visual impressions of true flight, as discussed further in §5.3.3.

5.3.2 Motion simulators

Moving base simulators attempt to provide sufficient motion input to the pilot's vestibular system to present the "feel" of real flight. The available displacements are very limited so that the accelerations and rates have to be washed out, introducing in their turn effects which can destroy the illusion of larger motions. However, modern airliner simulators with six-axis motion
have reached such a degree of sophistication that it is now acceptable in some countries for airliner type conversion training to be carried out entirely in them. These succeed because the linear and angular accelerations simulated are quite limited. Pilot proficiency maintenance and practice emergency drills are now almost exclusively carried out in motion simulators, at far lower cost and without the high risk involved in using actual aircraft.

Motion simulators are almost never used in the design of combat aircraft. The nature of the motions of such aircraft, such as 360 degree rolls or turning at high g, make ground-based representation physically impossible. Even the cues for the rolling motion for quite small bank angles cannot be represented accurately. A steady platform bank angle leaves the pilot falling sideways out of the seat, so the platform attitude has to be washed out until level in order to mimic the absence of a specific side force in the co-ordinated turn. As the pilot levels the wings, the platform has to roll in the opposite direction and then wash out to level again. This introduces two entirely spurious rolling motions which can be tolerated only if the roll is relatively ponderous.

A simple roll to 90 degrees may be begun and stopped by roll accelerations of over 500 deg/sec², producing as much as 1g laterally one way and then the other at the pilot's head, which is normally displaced well above the rolling axis. This can be enough to hit it on the canopy side if the pilot is not braced for it. Large platforms can only produce a small fraction of such acceleration, but even if they could manage it, it would be impossible to rotate to the limiting bank and return to level without producing the most confusing cues to the pilot, who is extremely sensitive to roll acceleration sensed in the ear canals and by the lateral accelerations at the head.

5.3.3 Visual displays

Head-down instrument displays, or head up displays (HUD) projected optically to appear in front of the pilot, can also supply total attitude, speed, altitude, flight path, heading, plan position or map location, and so on. It is possible to complete a mission from take off to landing with these displays alone, sometimes aided by director commands, and fixed base simulation can represent such flight very accurately. However, nearly all simulators supplement these displays with a representation of the outside world.

In the past, the outside world was usually displayed in a single TV screen. The early "contact analogue" used lines and squares etc to represent an earth/sky background, but its limited cues were a considerable restriction on the accuracy with which an aircraft could be assessed. High quality pictures could be obtained from moving belt landscape models viewed through a miniature TV camera with motor-driven lens, but the available scenery was limited to small areas.

Multi-screen computer generated displays became widely used with three projectors to continue the display area round to each side of the pilot. These provide far greater spatial cues and enable the pilot's peripheral vision to contribute to his handling assessment. The effect of this is most notable in the roll axis. It was a well known result of many simulations that pilots perceived the roll response as being more sluggish than in flight. This factor, largely caused by the missing vestibular sensing of roll acceleration, is greatly reduced by the restoration of the visual peripheral sensing so well adapted to the detection of movement. The visual illusion of motion can be so strong that the lack of inertial sensations may cause discomfort or even vertigo. The improved display dynamics relieve another symptom from earlier simulations, a tendency to small roll oscillations (which did not occur in flight) when trying to maintain a steady bank angle.

A factor which strongly influences the pilot's perception is the fidelity of the response of the display to the simulated motion. The earlier TV or analogue displays had quite good dynamics.

The human pilot
With modern Computer Generated Images or CGI, quite small delays caused by the refresh or update rate for millions of pixels being moved about the screen can cause non-real handling difficulties. Inadequate texture, the fine background detail of a scene, is a common cause for the lack of success in simulating landings in most simulators. In real flight a wealth of information on height and descent rate in the last stages of landing is obtained from the suddenly increasing perception of the ground texture. With the large computing power now available, it is possible to project excellent texture on to the scene immediately ahead of the aircraft resulting in remarkable improvements in the simulated landing performance.

5.3.4 In-flight simulators

In-flight simulation may also be carried out in an aircraft equipped with a variable stability flight control system that can be programmed to simulate the dynamics of any other aircraft. The long history of such simulation is detailed in Breuhaus [1991], with descriptions of the techniques used. Calspan at Buffalo, USA, have been the leaders in this field since 1948, though many other aircraft are listed in Table 2-3. Total fidelity is unlikely, as usually only the rotational response characteristics are accurately simulated. This leads to errors in true speed and in the simulated wing lift slope and loading, which connect the attitude and flight path responses and the ratio of pitch rate to normal acceleration. To match the lift and sideforce characteristics requires the generation of direct forces through the use of lift flaps, which are unlikely to provide more than rather modest force increments. While a small number of in-flight simulators are equipped with such direct force generators, most are not.

Despite the lack of total fidelity, in-flight simulation is well proven and high reliance can be placed on its results. The pilot is now in the real flight environment and can perceive a genuine relationship between motion and visual cues. The stress of real flight is there, so that a tendency to PIO is more easily uncovered. Airborne simulation is almost invariably required by the USAF before first flights of new military prototypes. Handling qualities experiments based on generic response characteristics have been of great use, and are the only source of reliable data. The quantity of such data has been rather sparse though fortunately adequate, and it has been essential for the author's study of high order handling behaviour leading to new design methods for fly by wire control laws. Some examples are Harper [1955], Chalk [1958, 1966], Neal and Smith [1970], Boothe et al [1974] and Smith [1978], in which a wide range of dynamics was systematically explored.

5.3.5 Use of simulators in the design process

In the process of control law assessment on a simulator, the designer and the pilots must be familiar with its strengths and faults. A difficulty with a new aircraft design is that the pilots cannot calibrate the simulation by a comparison with flight. Effects solely due to simulation must be inferred from cross reference to the handling of other aircraft with which they are familiar. An aircraft will usually seem harder to handle in the simulator than it actually is in flight, and it will take long and dedicated effort to arrive at a realistic evaluation prior to flight.

This can present a very real difficulty when pilots are presented with an unfamiliar characteristic that may stand out in the simulator but which proves to be insignificant in flight. One example is the pitch down that occurs in a pitch rate manoeuvre demand system when bank is applied without the necessary pitch compensation to maintain level flight, which is more pronounced than in a conventional aircraft. In the latter, the downwards acceleration is proportional to the shortfall in vertical lift component with increasing bank angle. In a PRD system this effect is augmented by the nose up pitch rate generated by the horizontal component of lift, to which the system reacts by a nose down pitch rate command. The pilot's pitch correction is in-
stinctive and automatic in flight, but has to be learned in the simulator.

Simulation has a well proven and essential part to play in the design of FBW control laws. It strings together all the individual aspects of handling which have been subjected to a proper analytical design process into the complete response behaviour to which the pilot reacts. It is also particularly well suited to optimisation of the design for non-linear responses or time-dependent changes in flight condition, such as large amplitude manoeuvring or during take-off. For example, providing a straightforward pilot control technique for takeoff rotation and climb away may require repeated time response calculations with gradual control law adjustments.

The experience with many FBW aircraft has been taken in the past to prove that the handling problems from which they have suffered cannot be predicted in ground simulation. The author is certain that most of these problems could have been found by looking for them specifically. In turn, this necessitates an understanding of such problems, and the ability to identify their presence by analysis of a design and then to rectify them before flight. The apparently mysterious PIO can usually be identified in a fixed base simulator, although the exact circumstances in which PIO might arise cannot always be predicted.

Nevertheless, it is a misuse of simulation to attempt the primary control laws design definition by relying solely on a pilot to "optimise" the whole design. Such a policy is unreliable and potentially dangerous. Simulation must always follow the analytical design work, and any changes introduced as a result of simulation must be performed analytically. Only in this way can the designer maintain the complete understanding of the system necessary for characteristics that are both optimum and safe.

A most important aspect of simulation is to achieve mutual trust and understanding between the design and simulation engineers and the pilots. All of these disciplines may have valid differences of opinion that must be reconciled. Success in this is beyond price, since failure renders the simulation of little value.

5.4 Summary

This chapter has discussed several aspects of the significance of human pilot physiology and cognition to the provision of satisfactory or preferably optimum aircraft handling qualities by designers of flight control systems. These are the motion and visual sense organs through which the pilot perceives the motion and control of an aircraft, the ability to operate in a linear servolike manner in accordance with a generalised describing function model of pilot behaviour, and the ability to interpret handling assessments in the non-real environment of a flight simulator to the real in-flight case.

The most important development, initiated in the 1950s and 1960s, was to understand pilots in terms of mathematical models that adequately replicate their observed behaviour. This has developed over the years to a very advanced level in optimal models. All these models were used to analyse pilot behaviour in association with the given aircraft dynamics of the time. The more recent development of fly by wire technology has changed the situation to one where the aircraft dynamics can be shaped to a considerable extent to suit the pilot. For this purpose the earliest and simplest pilot models suffice, leading to the ultimate simplicity in desirable dynamics. This is the $K/s$ - like response applicable to a wide range of controlled variables primarily including attitude and flight path, as discussed in Chapters 7 and 10.
a) Compensatory attitude tracking loop, with pilot minimising $\theta_e$

$Y_p =$ pilot describing function (Eq. 5-1) \hspace{1cm} Y_c =$ aircraft transfer function

$\theta_c =$ desired pitch attitude \hspace{1cm} $\theta_e =$ attitude tracking error \hspace{1cm} $\theta =$ aircraft attitude

$K_p =$ pilot gain \hspace{1cm} $\tau =$ pilot time delay \hspace{1cm} $K_c =$ aircraft gain \hspace{1cm} $\omega_c =$ crossover frequency

Note: Generally the pilot tries to achieve $Y_p Y_c = \frac{\omega_c e^{-\tau}}{s}$

This is known as the "crossover" model

The pilot prefers minimum equalisation where $Y_p \approx K_p e^{-\tau}$

and so the preferred aircraft dynamics $Y_c \approx \frac{K_c}{s}$

Remnant is a usually small portion of pilot's activity not linearly correlated with the forcing function $\theta_c$

b) Gain and phase of the crossover model

Figure 5-1 Compensatory attitude tracking loop and crossover model
Figure 5-2 Idealised time history of a K/s or crossover model

Figure 5-3 Identification of pilot equalisation in the experiment of Hall [1958, 1963]

1. Aircraft transfer function numerator $T_{02} = 0.6$ in all cases
2. "Remnant" is pilot activity not linearly correlated with the forcing function
Figure 5-4 Nichols plot of open loop response (rectilinear grid) for derivation of closed loop response $\theta/\theta_c$ (curved grid) (No pilot phase equalisation included, i.e. $Y_p = K_p$)

Figure 5-5 Typical closed loop pitch attitude tracking performance that can be derived from the Nichols open loop plot
Available pilot gain and crossover frequency limited only by delay. Large gain and phase margins and good closed loop tracking are easily achieved.

Figure 5-6 The basic crossover model in Nichols plot form

Figure 5-7 Typical variation in pilot ratings vs controlled element gain
Chapter 6
Aircraft design parameters and control hardware Influence on handling qualities

6.1 Introduction

The control stick forces and displacements desired for good handling have always been directly influenced by developments in aircraft design and control system hardware. The range of tools available to designers to shape a number of handling qualities have included aerodynamic means such as tabs, mechanical means such as springs and bobweights, artificial feel means such as q-feel or gear-change devices, and "active control systems" driven by computers. A comprehensive survey of stick and feel system design is given in Gibson and Hess [1996], with lessons learned from 1900 to the present day and descriptions of the interactions of modern system dynamics with the human pilot from a closed loop perspective. This chapter deals primarily with some handling qualities effects associated with powered actuation systems, control circuit design, and artificial feel devices.

The four most important aerodynamically-related basic parameters whose desired qualities determine the overall design of the control and feel system are related to the neutral and manoeuvre points as follows [Gerlach 1983]:

Static stability (stick fixed): "angle of attack stability, stick fixed", \( (C_{ma})_{\text{fixed}} < 0 \)
  related to: neutral point stick fixed
  related to "stick displacement stability", \( d\delta_e/dV > 0 \)
Static stability (stick free): "angle of attack stability, stick free", \( (C_{ma})_{\text{free}} < 0 \)
  related to: neutral point stick free
  related to "stick force stability", \( dF_e/dV > 0 \)
Manoeuvre stability (stick fixed):
  related to: manoeuvre point stick fixed
  related to: "stick displacement per g", \( d\delta_e/dn < 0 \)
Manoeuvre stability (stick free):
  related to: manoeuvre point stick free
  related to: "stick force per g", \( dF_e/dn < 0 \)

Most of the following is related to stick forces, since these are generally found to be of most importance to the pilot. However, stick displacements and their relation to the forces are of some significance, and these are also considered.

As is noted in Chapter 2, the difficulties in handling the pitch-unstable Wright aircraft in the first decade of flight were compounded in the 1908 two-seater by the non-intuitive design of the lateral inceptor. While the European concepts of stability and control were completely different and also illogical, their inceptor designs very quickly evolved into the familiar types of today, that is a stick or wheel pitch-roll control column and a rudder bar. Indeed, the first stick-type column was used on a completely unsuccessful aircraft, the Esnault-Pelterie REP, in 1907. The Bleriot designs, with a control stick and rudder bar, were widely used in training schools before

\[9\] In this context, “circuit” has nothing to do with electronic or hydraulic circuits, but is a standard industry nomenclature for the mechanical system of rods, cables etc. connecting the stick to the actuators or control surfaces.
1914, strongly influencing the standard layout for aircraft in the 1914-1918 war and afterwards. Breguet introduced the standard wheel control in 1911. [Walsh 1976, Oppel 1986, Coombs 1990]

There was no further fundamental change in inceptor design for over half a century, until the advent of the fly by wire era that led to changes in concept. It was no longer desirable to maintain the large travels which had been previously necessary to cater for their direct relationship to the control surfaces and to reduce the control forces to acceptable levels. Occasional problems with the new designs in their turn led to new insights into the influence of inceptors on handling qualities, though the knowledge of the subject remains largely empirical to this day.

Given sufficient stability through proper aerodynamic configuration design, handling qualities were influenced primarily by the detailed design of control surface aerodynamic balance and trim devices. Their development was pursued intensively during World War 2 to cope with the rapidly increasing power, size and performance of military combat aircraft. Various aerodynamicic and mechanical means to manipulate control forces are described in Gibson and Hess [1996]. It became increasingly difficult to achieve consistent and satisfactory results, and these devices represented a large overhead in the flight testing of many designs because of the repeated adjustments that were often necessary, even during production flight testing.

In the post-World War 2 period, further increases in aircraft performance led to the rapid development of hydraulically boosted and fully powered controls (although the earliest example of boosted controls had been seen in 1935). The response of the actuators introduced a new factor in the equations of motion and had some unexpected effects on the feel qualities. It became necessary to replace the effect of the control surface aerodynamic hinge moments with artificially generated “feel” forces, such as stick force/deflection gradients proportional to dynamic pressure as in q-feel. The role of the aerodynamicist in providing good handling qualities began to be supplemented by, and was eventually almost subordinated to, the new flight control engineer.

Relatively slow transports such as turbo-props and some of the smaller jets continue to use unpowered aerodynamically balanced controls. Recently, however, it has become necessary to use a power control for the elevator of the twin turboprop SAAB 2000 to overcome pitch control problems. Much earlier, a fully powered elevator replaced the aerodynamic servo elevator of the original British Aircraft Corporation 1-11 twin jet airliner after loss of pitch control in deep stalls was encountered.

6.2 Conventional aircraft

An important expression to a pilot of the stability of an aircraft is through the stick forces that must be applied to control it. Stick free manœuvre stability can be simply expressed as the stick force per g. In conventional aircraft with unpowered controls, this does not vary much within a reasonably moderate range of flight envelope, at constant loading and centre of gravity [Gerlach, 1983, Chapters 4 - 6]. This arises from the level flight variation of aircraft lift coefficient inversely proportional to dynamic pressure together with the fundamentally linear relationship between elevator angle and lift coefficient. The elevator angle per g is inversely proportional to dynamic pressure, and with a constant hinge moment coefficient the elevator hinge moment per g and the stick force per g are constant. Stick free static stability is expressed as the stick force to change of speed, which is inversely proportional to the initial trim speed, while the elevator angle to change speed varies inversely with the cube of the trim speed.
The elevator angles and stick forces are given in principle as follows:

\[ \frac{d\delta_e}{dn} = \frac{1}{C_{mb}} \frac{W}{\frac{1}{2}\rho V^2 S} \cdot H_m \]  
(6.1)

\[ \frac{dF_s}{dn} = \frac{d\delta_e}{ds_e} \cdot \frac{S_e \bar{e}_e}{S} \cdot \frac{W \cdot C_{hb}}{C_{mb}} \cdot H'_m \]  
(6.2)

\[ \delta_e = \frac{1}{C_{mb}} \left( C_{m\delta} + \frac{W}{\frac{1}{2}\rho V^2 S} \cdot H_n \right) \]  
(6.3)

\[ \frac{d\delta_e}{dV} = \frac{4W}{\rho V^3 S} \cdot \frac{1}{C_{mb}} \cdot H_n \]  
(6.4)

\[ \frac{dF_s}{dV} = -2 \frac{d\delta_e}{ds_e} \cdot \frac{S_e \bar{e}_e \cdot W}{S} \cdot \frac{C_{hb}}{C_{mb}} \cdot H_n \cdot \frac{1}{V_{trim}} \]  
(6.5)

where
\( H_m \) = stick fixed manoeuvre margin
\( H'_m \) = stick free manoeuvre margin
\( H_n \) = stick fixed static margin
\( H'_n \) = stick free static margin
\( \delta_e \) = elevator angle
\( s_e \) = stick deflection
\( F_s \) = stick force
\( S_e \) = elevator area
\( \bar{e}_e \) = elevator mean chord
\( C_{mb} \) = pitching moment about centre of gravity due to elevator
\( C_{hb} \) = elevator hinge moment about hinge line

Towards the end of the second World War, aircraft flight and loading envelopes began to expand sufficiently to cause wide variations in stick force per g. Attempts were made by NACA to minimise this with a form of artificial feel. This was to be done by extremely close elevator balancing to achieve zero aerodynamic hinge moments and by the addition of a bobweight, that is a mass attached to the stick or control circuit applying a force to the stick proportional to normal acceleration. The stick force was intended to come from the bobweight alone, and would therefore be independent of wing loading, centre of gravity position, Mach number etc. Practically constant stick force per g was successfully demonstrated, but overall the attempt failed.

It was found that although the forces in steady turns were satisfactory the dynamic feel qualities were very unpleasant in rapid manoeuvres [Phillips, 1946]. There would be no stick force during the initial control input to indicate to the pilot how much control had been used, and it was impossible to predict the magnitude of the resulting manoeuvre. This led to a requirement that the stick force during a manoeuvre onset should not be less than the force in the subsequent steady manoeuvre [Anon, 1969]. Another way of expressing this is that the displacement of the stick should not lead the force. A historical review and analysis of bobweight dynamics with unpowered controls is found in Chalk et al [1969].

A bobweight system will always have been adopted to improve the stick free manoeuvre margin expressed in stick force per g terms. However, it also has an influence on the stick free static margin, expressed as stick force versus speed change. By exerting a constant forward force on
the stick, the bobweight forces the elevator downwards when flown hands off as the speed is reduced from an initial trimmed condition. The aircraft will therefore pitch down more positively than with no bobweight fitted. To counteract this, the pilot must apply an aft stick force additional to that generated by the normal aerodynamic hinge moment variations as the elevator angle is altered to maintain level flight. The static margin effect can be eliminated by the use of a balancing spring that counteracts the bobweight force in level flight. Problems caused by bobweights in non-level or inverted flight are discussed in Gibson and Hess [1997].

The downspring is a means of augmenting the stick free static stability alone. This is a long low stiffness or "slack spring" that pulls the stick forward with an effectively constant force, working in the manner described above. Both mechanical feel devices have continued to be used in many non-powered control aircraft as a palliative for aerodynamic inadequacies in stick free manoeuvre or static stability. Goldberg [1953] presents the theory of these devices. It is shown that excessive reliance on a downspring for static stability enhancement can lead to unsatisfactory stick free dynamic stability. Eshelby [1975] discusses the effects of a stiff spring.

6.2.1 Power control actuation
A primitive early version of hydraulic power actuation was the power boosted control, in which most of the aerodynamic hinge moment was carried by the actuator with a small proportion still applied by the pilot's direct inceptor effort [Gibson and Hess, 1997]. Originally introduced in some large piston engined aircraft such as the Douglas XB-19 of 1935, the Boeing 307 of 1937, and the Lockheed Constellation, the system was also used on some of the later World War 2 combat types and on the early jet fighters. Handling qualities were improved by the reduction in stick forces, though small amplitude control behaviour was sometimes adversely affected by the need to move the output physically before the power boost was activated. The more extreme flight conditions that became more common could not be satisfactorily alleviated, however, and by the early 1950s full power actuation was becoming the norm.

Figure 6-1 illustrates the basic flow valve, piston and cylinder of a fully powered hydraulic control actuator. Movement of the pilot's inceptor, connected to the valve by the control circuit linkages, initiates fluid flow under pressure into and out of the actuator. The actuator velocity is proportional to the opening of the valve ports, and hence it follows slightly behind the input motion with a small error proportional to the velocity. After the input motion stops, the actuator body continues to move until the valve is closed, at which the positional error is zero. (The valve actuating linkage will take other forms if the cylinder body is anchored to the airframe structure and the piston shaft forms the output, but the functionality is the same.)

While the pilot is isolated from the direct influence of the aerodynamic hinge moments, there are two particular consequences of importance which influence the handling qualities and stick forces and which were not present in the previous era. This led to the introduction of the control engineer alongside the aerodynamicist and to an ultimate merging of the two disciplines in the design of flight control and handling qualities.

The first consequence was the introduction of what are effectively pure lag dynamics between the stick and the control surface position. Early actuators had quite low bandwidths with high phase lags, which sometimes caused significant handling degradation [Finberg, 1963]. This was remedied by reducing the effective lag time constants to 0.05 seconds or less, typically by increasing the valve port areas to reduce the velocity error, that is the amount of valve deflection required to produce a given actuator rate.

Aircraft design parameters and control hardware influence
The second consequence was the introduction of non-linearities and valve force problems. It was extremely difficult to achieve perfect matching of the moving spool valve edges with the ports delivering the fluid. This resulted in a non-linear small amplitude performance degradation with adverse effects on precision control. Valve friction, caused by extremely small clearances in which contamination particles in the fluid could silt up, resulted in a tendency to hold the valve fixed. This could produce unwanted control movements [Phillips, 1957], back-driving the stick through the feedback linkage connecting the control circuit, valve and actuator position, Figure 6-2(a). Such problems were eventually overcome but the highest quality engineering standards were required to achieve satisfactory handling.

The hydraulic flow exerts a force on the spool in the closure direction, proportional to flow rate and therefore to actuator velocity. This is reflected at the stick as apparent viscous damping, Figure 6-2(b). Special spool shapes were developed to minimise this effect. Sometimes a pair of valve centring springs was added to alleviate friction effects, Figure 6-2(c). These springs added to the effective velocity damping force, and were used in some designs for the purpose of improving control circuit damping. The control circuit could also be preloaded against the feel system centring force by the use of unequal springs or a single spring, to minimise backlash.

Transport aircraft had much less stringent flight control system requirements than military combat aircraft, and lagged behind in the adoption of advances in powered control. Early large jet transports such as the Boeing 707 and Douglas DC-8 used aerodynamically balanced control surfaces, but it was extremely difficult to achieve satisfactory and consistent control hinge moments and stick forces, or even to achieve them from one airframe to the next. This was due to their size and high cruising Mach number, and to the difficulty of maintaining adequate manufacturing control over tolerances in the control surface shrouds and balance elements. However, the world’s first jet airliner, the de Havilland Comet, used fully powered controls.

6.2.2 Control circuit characteristics
Of all aircraft designs ever built, the majority have had cable control circuits which continued to be used with the advent of power controls. They have low inertia and backlash, and can be easily routed through small spaces and tortuous routes, but they are prone to considerable friction, increasing at every pulley or fairlead. On civil aircraft the problem is doubled because twin cable circuits are usual to achieve the required reliability. Assessments of military aircraft up to the 1960s found that friction caused most of the serious control difficulties encountered in many [Lang and Dickinson, 1961]. The importance of friction diminishes with heavier stick forces and with reduced manoeuvring requirements. The de Havilland Comet, noted for high friction levels, could be acceptably controlled by the pitch trimmer for most of its essentially non-manoeuvring flight. As a general rule, however, control characteristics were always improved as friction was reduced, and there was no evidence of a minimum satisfactory level.

In fighter aircraft, with generally low stick forces and a requirement for precision control, the most successful control circuit designs used push rod and lever mechanisms. Some simplification was achieved in the earlier Mach 2 English Electric Lightning, Figure 6-3, by running much of the rod system in rollers (including some curved rods). By close attention to detail, the friction was reduced to a minimum of only 2 or 3 Newtons of stick force to initiate control motion, which was considered to be excellent. The necessary investment in design was not always well rewarded. Anderson [1979] describes the VFW VAK-191B V/STOL (vertical take-off and short landing) aircraft in which the mechanical control system, intended for reversionary use if the primary electrical controls failed, had so much friction that it could not be used safely.
If control circuits are not run along the neutral axes of the major airframe components, bending of the structure under load changes their relative lengths and can introduce spurious commands to the control surface actuators. In one example this reduced the apparent stick fixed manoeuvre margin to zero [Kraft et al 1958], i.e. the stick position during the manoeuvre was the same as in level flight although a finite control deflection was being applied. Mechanical compensation for this effect is possible, for example by alternating the direction of motion of successive rods between the suspending levers. Cable control runs may include many automatic tensioning devices which allow the cables to change in length without changing the tension. Airframe heating in the Concord supersonic airliner causes a significant change in fuselage length and the mechanical circuit is declutched while the FBW system controls the aircraft.

6.2.3 Artificial feel for powered flight control systems

Pitch feel

The earliest artificial feel device for powered control systems was the simple centring spring. The resulting stick force was proportional to the elevator deflection. This could not normally be used as the only source of feel in the pitch axis. The stick force per g is obtained by factoring Eq. (6.1) by the stick feel spring stiffness gradient:

\[ \text{Stick force per g} = \frac{k}{C_{mb}} \frac{W}{\frac{1}{2} \rho V^2 S} H_m \]  

where \( k \) = stick feel spring stiffness gradient per unit elevator deflection

The stick force per g is inversely proportional to speed squared. Unlike the constant values in conventional aircraft, it is either excessively heavy at low speed or excessively light at high speed, or both if the speed range is high enough.

A similar difficulty arises with static stability. The stick force to change speed is obtained by factoring Eq. (6.5) by the stick feel spring stiffness gradient:

\[ \frac{dF_e}{dV} = \frac{4kW}{\rho V^3 S} \frac{1}{C_{mb}} H_n \]  

The result is a stick force to change speed that is inversely proportional to the trim speed cubed, rather than inversely with speed in conventional aircraft.

Simple spring feel has been used occasionally, as in the Lockheed SR-71 reconnaissance aircraft despite its extremely wide speed range to Mach 3. However, its operations were essentially non-maneuvring and confined to a relatively small dynamic pressure range. It used a non-linear gearing to provide heavy forces at neutral stick positions together with a large authority triplicated stability augmentation system for improved response dynamics. The de Havilland Comet used a gear change with a centring spring to alter the feel between take off and cruise conditions [Gibson and Hess, 1997].

The earlier problems found with unpowered controls arose again when bobweights were added to spring feel to achieve a more uniform stick force per g. The combination of a fairly heavy bobweight and a light feel spring could in principle provide stick forces with a much reduced dependence on the elevator angle per g and its related factors. Such a combination usually results in a rather low natural oscillation frequency of the control circuit. With the stick free, this mode becomes coupled through the response feedback of normal acceleration to the airframe short period to produce a new stick free mode with typically very low damping. An example was the North American F-86E Sabre fighter, in which a fully powered elevator and tailplane
system replaced the unsatisfactory power boosted elevator with trimming tail of the F-86A, one of the earliest jet fighters. The handling of the F-86E was described by test pilots of the Royal Air Force [Dickinson, 1953] as “spongy” at low speeds to “very sensitive indeed” at 400 knots, the pilot tending inadvertently to prolong any short period pitching in attempts to control it.

Several aircraft were destroyed by catastrophic pilot induced oscillation. The association of this with bobweights is discussed in Chalk et al [1969] and Neal [1971], with an analysis of bob-weight effects on the dynamics of aircraft with powered controls. Bobweights are still in use today in aircraft with spring feel such as the Hawk trainer with its simple control system, in the Lockheed F-104 Starfighter, and in the McDonnell F-4 Phantom where the bobweight is used with a rudimentary q-feel system. The latter type was prohibited from operation at high dynamic pressures with its stability augmentation system inoperative. These bobweight systems date from earlier technology, and although the method has been made to work sufficiently well, its use in a future fly by wire system seems improbable.

When flown hands off, the control surfaces of an aircraft with powered controls are not free to float under the influence of aerodynamic hinge moments. The stick fixed and stick free manoeuvre stabilities are therefore the same, unless a bobweight is used to augment the latter. The stick fixed and stick free static stabilities are also basically the same, but the latter can only be augmented by a downspring (or bobweight) if the artificial feel force gradients vary with speed. There is usually less difficulty with stick free stability than in aircraft with unpowered controls, where the stick free margins are almost always less than the stick fixed margins. The author does not know of any aircraft with powered controls that uses a downspring. The Mach trimmer device mentioned in §2.3.1 was an alternative sometimes used in the transonic cruise condition.

By the 1960s, q-feel devices were in use for the pitch axis in a considerable number of U.K. military aircraft designs. A q-feel device simulates the stick forces caused by dynamic pressure-dependent hinge moment effects of conventional aerodynamic controls. Service assessments in the U.K. [Lang and Dickinson, 1961] had established that they tended to have far fewer handling problems, a point also made by Chalk et al [1969]. The artificial hinge moments can also be made to vary in ways independent of the dynamic pressure, however, as discussed in Gibson and Hess [1997].

Static instability in the transonic region is unavoidable, due to aft migration of the aerodynamic centre, causing a reversal in the direction of stick movement and force to trim speed. This could be made moderate and unobjectionable by good aerodynamic design as typified by the English Electric Lightning in Figure 6-4 [Dickinson 1968]. Manoeuvre stability increased in the transonic and supersonic regime due to the transonic aft shift in manoeuvre point, Figure 6-5 [ibid], the tail angle per g departing from the normal inverse relationship with dynamic pressure.

To prevent the generation of excessive supersonic stick forces, the Lighting q-feel gradients varied only with altitude above Mach 0.9 in a manner similar to the tail angle per g, Figure 6-6. An additional mechanical feel spring (functioning also as a back-up should the q-feel system fail) biased the stick force/displacement gradients upwards at low dynamic pressure, where a low stick force per g desired for easy manoeuvre at up to 8 or 9 g at high subsonic speeds would result in low maximum forces due to the small levels of g at the stall limit. This stick force per g trend was proposed by Gibson [1978] as a desirable specification, Figure 6-7. The format of this figure is the same as the specifications such as Anon [1969]. The n/α scale, in units of normal acceleration per radian angle of attack, is equivalent to a dynamic pressure scale taking account of wing loading and lift slope (§3.2.2).
Many civil aircraft have used q-feel, e.g. the Lockheed C-5A and the Boeing series from the 727 onwards. They must cater for a wide centre of gravity range, which would normally result in a wide range of stick force per g. This has often been alleviated by modifying the nominal q-feel stick force gradients as a function of the tailplane trim position. The gradients are reduced as the tail setting becomes more negative with increasingly forward centre of gravity, to counter the corresponding increase in elevator angle per g [Gibson and Hess, 1997]. This method works well because the cruise operating condition is confined to quite narrow limits of airspeed.

Roll and yaw feel
Most aircraft, both civil and military, have used simple spring feel in the roll axis, often with an authority restriction or gear change at higher speeds. In the Lightning, this was effected by the raised undercarriage door, halving the aileron travel with wheels up. Additionally, a non-linear gearing reduced the control sensitivity around the neutral stick position. In the British Aircraft Corporation TSR-2, a complex hydraulic/electro-mechanical gearing device reduced the roll authority as dynamic pressure increased, increased it when the flaps were lowered, and provided an aileron-rudder interconnect to improve roll co-ordination [Gibson and Hess, 1997]. Such an interconnect has been used on many fighter designs, though never on civil aircraft.

Similarly, spring feel with simple restriction or gear change has been used in the majority of rudder feel systems. In the Lightning, the pedal forces were provided by q-feel with additional mechanical springs for reliability, adjusted to satisfy the airframe strength requirements based on pedal forces of specified magnitude with step or cyclic inputs. To prevent uncomfortably high pedal forces at low speeds, the hydraulic component was switched out with the undercarriage lowered, leaving only the mechanical springs to provide the feel force. In the TSR-2, reliability was achieved by duplicated feel units, avoiding excessively large low speed forces.

Criteria for stick force gradients
There are no absolutely clear guides to the most satisfactory gradients of stick force or displacement per g. Specifications require more than a minimum stick force per g to avoid over-control due to light forces and less than a maximum to avoid difficulty in reaching maximum manoeuvre levels, but they are not related otherwise to task performance. Very stiff feel with as little as two millimetres displacement per g was sometimes chosen by pilots in early Calspan flight experiments, while the author has found values of about 10 millimetres per g have been very acceptable to pilots. Proposed optimum relationships between displacement and force are given by Gerlach [1983] in his Figures 6.3 and 6.4 (from Wimpenny [1954] and Harper [1955] respectively), but these are based on insufficient data to be considered reliable.

The only experiment known to the author in which a wide range of displacement and force gradients was tested is that of A’Harrah [1964]. This found that pilot opinion was influenced more by the stiffness gradient, ie N/cm, rather than by stick force or displacement per g in a comprehensive simulation using a “g chair”. The range of 5 to 45 N/cm was generally satisfactory. For a given stick force per g, the effect of gust induced disturbances through the arm mass was relatively smaller with lower stick stiffness, and a reduced range of 12.5 to 25 N/cm was most satisfactory in low altitude high speed flight. This is certainly in agreement with the author’s experience. Abramovitz et al [1954] found that pilots could perform tasks satisfactorily with a wide range of stick force and displacement gradients.

Figure 6-8 illustrates the characteristic hysteresis effect of friction on feel. Within the friction band, the stick free position is indeterminate, and it is difficult to establish the correct trim
point. With low static stability, this can even lead to stick-free speed divergence similar to static instability. A positive trimming point can be achieved by the addition of a spring breakout force sufficient to separate the hysteresis loops, but if the breakout force is too large relative to the subsequent stick force variations needed for manoeuvres, precision control is very difficult.

Fig 6-9 illustrates an effect created by long control circuit lengths and a feel unit located at the rear of the aircraft. The circuit and its mountings comprise an additional stiff spring in series with the feel unit. This increases the stick deflections by an amount depending on the feel unit stiffness, but does not alter the stick force versus control surface angle. In this example the gradient of stick force against stick displacement is approximately halved at high speed relative to the nominal case with a rigid control circuit. In the absence of an actual pitch control circuit between the cockpit and the tailplane in the fly by wire Boeing 777, the characteristic of circuit flexibility was replicated by a stiff spring in the cockpit-located control wheel and feel system. This was intended to maintain the "feel" of traditional Boeing aircraft handling qualities. The first example of this technique was actually tested successfully in the Mitsubishi Zero fighter of the second World War [Horikoshi, 1965], with a "flexible stick" concept.

### 6.2.4 Non-linear gearing

Pitch control sensitivity was often given close attention in combat aircraft in addition to satisfactory stick forces provided by a good feel system design. With the stick linearly connected to the all-moving tail or elevator, the displacement per g was proportional to the tail or elevator angle per g. Since this is basically inversely proportional to speed squared, it could vary by as much as 25:1 over a potential speed range of 5:1 or more. Also, in combat aircraft the stick will be well forward when flying at high speeds because the tail is trimmed near its positive limit, and will be well aft at low speeds, especially with high lift flaps extended which often increases the trim angle range even further. The typical total stick travel of 250 to 300 mm of earlier combat aircraft imposed a large range of stick positions to be held in level flight, and it was difficult to achieve a comfortable arrangement.

A non-linear gearing was often used to alter the gearing ratio between the stick and the tail at different stick positions. By this means it was possible to maintain a desired minimum stick displacement per g at high speed without too large a value at low speed, and to bring the stick nearer to the pilot in the high speed conditions. Figure 6-10 sketches one such gearing. While only moderate changes in gearing are possible, the design requires only a simple four-bar linkages and has proved to be very effective. The monotonic gearing slope is relatively insensitive to the tail angle trim variations due to mass and aerodynamic effects of external weapon loads, CG movement due to fuel usage, and transonic pitching moments.

Figure 6-11 sketches a much more extreme arrangement, which is intended in principle to maintain a more nearly constant stick displacement per g over a wide speed range. The curve attempts to follow the variation of tail angle per g as the trim varies with CG and lift coefficient, so that only a simple spring feel should be needed. The point of inflection of the gearing curve has to be placed close to the high speed flight trim conditions, but here the non-monotonic slope peaks at a high value and changes magnitude very rapidly with small variations in stick position. Even quite small elevator trim angle differences from the nominal design value can cause large departures from the intended gearing slopes and force characteristics. The tail angle per g departs considerably from an inverse dynamic pressure relationship at transonic and supersonic speeds (Figure 6-5 is an example). Coupled with possible variations in tail trim angle, this can result in extremely heavy stick force per g and/or a wide variation in stick force per g gradient in a traverse from low to high g as the stick moves across the inflection point.
Problems that have occurred in development testing with the Figure 6-11 gearing type include dangerously low or excessively high stick forces with certain store configurations, and large stick movements to maintain level flight caused by small changes in the tail setting with reheat selection. In another case the static stability in the landing approach, as apparent in the trim stick displacement per knot, decreased with forward CG movement due to the extreme nonlinear curve at the aft stick positions. The mechanism may comprise a collection of swinging linkages which can introduce undesirable dynamic stick forces [Assadourian 1958], or it may contain a cam mechanism which can introduce undesirable static breakout forces. Despite such problems, this type of gearing has been successful on a number of subsonic operation aircraft such as the British Aerospace Hawk and the Sepecat Jaguar. In the latter a variable feel unit was still necessary, with a 2:1 range of force gradients.

6.2.5 Trim methods
To an aerodynamicist, “trim” may mean the balancing of an aircraft state by appropriate choice of control surface settings to achieve a zero pitching, rolling or yawing moment overall. To a pilot, it is simply the condition of the control system in which flight can be maintained with the hands off the controls, hence in which a stable equilibrium is obtained. The pitch handling is affected by the method of trimming, either parallel or series.

In parallel trim, Figure 6-12(a), an out of trim force is removed without altering the stick or the control surface positions. This is done by adjusting the datum of the feel unit relative to the stick so that the force to hold the stick decreases to zero. When frequent changes of trim are required, typical of combat aircraft, this method is greatly preferred because the stick does not have to be moved twice for each new condition, once to achieve it and once to trim it. There is also a direct feedback to the pilot of the control surface position.

In series trim, Figure 6-12(b), the feel unit datum is constant relative to the stick, which has a fixed neutral position. The relationship between the stick and the control surface is changed by altering the length of the circuit. The stick must be moved back to the neutral position as the trimming takes place to prevent a change in the control setting. The method is much less common and is often disliked by pilots of combat aircraft.

Because of the wide CG range needed for most transports and the pitching moments of their high lift flaps, the commonest method of control is through an elevator mounted on a tailplane whose setting on the fuselage can be altered by a trim actuator. This is series type trim, and it is generally acceptable to transport pilots with little need for its frequent use. It lends itself particularly well to “flying on the trimmer”. A few designs use the “slab” tail, which is an all-moving tail without a separate elevator, in which case trimming is of the parallel type. Smaller transport aircraft more generally trim by tabs on aerodynamically balanced controls, which is parallel trimming in principle. There is actually a small element of series trim in that the elevator needs to be re-adjusted when the tab setting is altered, but this effect is usually rather small and may not be noticed. It differs from the usual series trim in that the stick displacement must be increased further as the trim is applied because the tab deflection reduces the elevator lift force.

6.2.6 Inceptor dynamics
While inertia weights have often been added to obtain desired pitch stick force per g, the entire control circuit is also an inertial mass which responds to normal, longitudinal, lateral and rotational accelerations. If these have undesirable or even unacceptable results, inertial counterbalance must be applied. Carrier aircraft represent an extreme case of the necessity for longitudinal
balance, because of the very high acceleration of the catapult launch with the pilot's hands kept off the stick to prevent inadvertent inputs.

Obviously, a cable circuit does not have significant inertial effects, but all control columns, pedals and throttles may do so. Push rods are often oriented essentially vertically, acting as a bobweight, where they transfer from the cockpit floor upwards to the spine or upper fuselage region, or downwards back to the tail surface level. They should at least be arranged or otherwise counterbalanced so as not to cause adverse effects on the stick force per g. Vertical accelerations resulting from pitch acceleration can also introduce unwanted effects, so that different sections of the circuit which cancel the normal acceleration effect may still require balancing action to cancel unwanted pitching acceleration effects.

Longitudinal acceleration of 1 g or more is achievable in some combat aircraft with very high thrust to weight ratio, and the total longitudinal component of high normal acceleration at a large angle of attack can be much greater. This causes all the push rods as well as the stick to "fall" to the rear of the aircraft. Three balance weights were added on the Lightning, one on the rudder pedals, one to balance the pitch rods, and one to balance the stick separately which was offset to eliminate normal acceleration coupling on the slightly aft-cranked stick top. Similar balance is usually desirable in most high performance and agile aircraft, but is to be found in large transport aircraft also. Thomas [1991] mentions FAR certified aircraft which had to be restrained from pitching when accelerating, decelerating, or climbing, and some would pitch up to the stall or into a dive if power was applied stick free.

Sometimes an extreme effect may occur. In the Sepecat Jaguar with its non-linear pitch gearing, the longitudinal component of acceleration acting on the mass of the pitch rods in manoeuvres produced the stick force effect shown in curve (a) of Figure 6-13 from flight test records at one point in the flight envelope. The gradient fell from satisfactory to less than half the specified minimum as the g increased. The addition of inertial longitudinal balance effected the remarkable improvement shown in curve (b).

Roll acceleration reaches values several times greater than pitch acceleration. Its most significant effect is the lateral specific force it exerts on the stick grip and pilot arm mass combination if these are situated away from the roll axis. It effectively acts as a form of negative feedback on sticks which lie above the rolling axis, which is fortunately usually the case, tending to reduce the pilot's input action. In conventional aircraft, this is not known to have caused any handling problem. It is a factor in the roll ratchet phenomenon experienced in a number of fly by wire aircraft, as noted in Chapter 9. Inertial counterbalancing is difficult because it has to be applied at the same displacement from the rolling axis as the stick grip. However, Norton [1995] gives the example of the Bell Textron V-22 tilt-rotor aircraft where counterbalancing was part of the solution to a complicated roll-pilot inertia coupling where the roll axis was above the cockpit.

The amount of inertia and friction of the stick and control circuits is often considerable and acts as a significant filter to the pilot's inputs. Provided that the stick free natural oscillation frequency of the system is sufficient, additional viscous damping does not appear to be essential, but it can make a conventional stick feel better. Because the stick and feel system lag dynamics are required to be counted as part of the overall control system lag by specifications such as MIL-F-8785C, additional stick damping was sometimes deliberately omitted. DiFranco [1968] gives an early example of research into this subject, and further discussion is given in Gibson and Hess [1997]. Results are inconclusive, though majority opinion seems to accept that the pilot is capable of compensating for most or all of any extra lag effect if it is not too extreme.
6.2.7 Augmentation

The augmentation system of most early military aircraft was usually very limited in authority. This protected against the risk of failures where the system went hard-over to the limit. Often the handling improvements were essential operationally, but the Lightning with an excellent aerodynamic design and a control system of high quality had been tested throughout its Mach 2 flight envelope without autostabilisation for its first four years. At first its pilots believed auto-stabilisers to be unnecessary, until these were installed and the benefits demonstrated. Then, as always, pilots raised the standard of handling qualities needed to satisfy their requirements.

The pilot’s notes for all such aircraft contained many restrictions to the control inputs and permitted manoeuvres. In the typically “inertially slender” types of the period, that is with low inertia in roll compared to pitch and yaw, rapid rolling at zero or negative g could lead to an autorotational departure, with very high roll rates, large sideslip, high positive g but negative pitch rate [Day, 1997]. If recovery was possible, it was with a nose up control input, completely contrary to the instinctive nose down input to reduce the g. Limitations on the amount of lateral control input had to be voluntarily respected by the pilots, an unreliable method that usually required large safety margins.

A blend of the old and the new began to take place with a few aircraft such as the SR-71 noted earlier. The authority range of the augmentation was extended to provide more complete enhancement of the handling qualities, and this required the introduction of multiple lane systems for safety in an extension of autostabiliser technology towards the fly by wire era. The fin area of the TSR-2 was minimised to reduce the turbulence response and airframe loading in the primary task region of 600 knots at sea level, giving directional instability at its Mach 2 limit. To restore stability, a large authority triplex yaw autostabiliser was designed using the analogue technology then current. While this was not proved in flight because the project was cancelled early in the flight test programme, it provided invaluable design and development experience.

6.3 Fly by wire aircraft

Aircraft such as the Lightning succeeded with very simple controls. More complex aerodynamics or the increasingly capable control technologies in recent years often led to a more elaborate mechanical system. Even where there was a long continuity of experience in advanced aircraft, new problems frequently came to light. In fly by wire aircraft, most of the elaboration lies in the control law block diagram. Figure 4-1 is only a most basic example of such a control law, but as noted further in Chapter 8, Hodgkinson [1982] describes a pitch control law (for the McDonnell Douglas F-18) that contained over 89 elements. Nevertheless, good mechanical design practice is no less important than it ever was.

6.3.1 Actuation

The required linear actuation dynamics are determined by the usually severe stability and performance needs of the flight control system. The pilot is also completely isolated from the mechanical aspects of valve friction and flow forces. The adverse impact of small amplitude actuator characteristics on handling qualities with primarily mechanical control systems is no longer experienced, therefore.

Large amplitude effects have become extremely significant, however. When there is a mechanical connection to the actuator valve, the pilot is physically aware of an actuator rate limit because the stick is prevented from moving faster when the valve has reached the end of its stroke.
In this case the stick position is always closely related to the position of the actuator. Actuator acceleration is controlled by the valve rate, again a direct function of the pilot’s stick activity. These connections are absent in fly by wire actuation, and the pilot has no physical indication of actuator limit exceedance. It is easy for the pilot to drive the actuation into rate saturation.

Unless extremely carefully considered in the design process, these non-linearities may have a profound effect on the closed loop stability of the aircraft, the control system and the pilot. As the aircraft dynamics may depend critically on the loop closure and may be severely degraded if actuator limits are exceeded, the handling qualities can be catastrophically altered without warning, a common cause of pilot induced oscillations. While these will stop when the pilot ceases control, as explained in Chapters 8 and 9, in the worst case a self-sustaining oscillation may occur even with no further pilot inputs.

6.3.2 Pilots’ inceptors

With the ability to pick off pilot stick inputs of either force or displacement by electrical means, engineers may be tempted by the apparent simplicity of a rigid or “force” stick. The absence of stick motion acting effectively as a filter between pilot and command allows higher precision in certain tracking tasks. Similarly, the use of force pick-offs on a stick with conventional displacements offers a more lag-free interface between the pilot and the task. Such conceptions do not accurately represent the constraints of real flight. In several cases, abrupt, inadvertent, or turbulence-induced stick inputs degraded the handling so severely that a change to displacement sensors was necessary, as in the McDonnell Douglas F-18 [Walker 1982].

The General Dynamics F-16 has been the only production aircraft equipped with a rigid side stick. Although most pilots adapted to its use, it proved difficult to optimise as an interface. For some flight tasks the absence of stick motion was a hindrance to good control. Also, because there was no physical indication of the limits, pilots often applied twice as much as the maximum signal force, which was very tiring in sustained manoeuvres. Small aft and lateral deflections were introduced, approximately 4-5 mm in pitch and ±3 mm in roll [Garland 1978]. This was just enough to give a sense of movement and to enable the maximum travel limits to be felt.

"Wrist action" sticks, that is with a very short pivot length and significant angular motion, suffer from cross-talk between the pitch and roll inputs unless the pivot axis is skewed to be in line with the fore-arm. This makes them generally unsuitable for central mounting, as the large skew needed can lead to a roll input being applied when only pitch is intended. This was a concern for the Shuttle Orbiter which has such sticks skewed by 18 degrees, but its limited manoeuvre requirements have not shown a problem in the flight environment [Gilbert, 1983]. Wrist inceptors can work effectively in the side location, where experiments have shown that there is a certain minimum displacement and maximum stiffness for optimum function [Black, 1979].

Arm action sticks, pivoted further from the hand, can also work well at the side. In the Lockheed YF-22, travels of 13 mm aft and 6 mm forwards were used successfully [Dornheim, 1992]. Arm action is generally essential in the centre position, with a substantial length between the pivot and grip to reduce the angular motions. It is also less tiring in sustained manoeuvres where large forces have to be applied. The wrist muscle system is not well suited to this [Myers et al, 1987]. Another example is the original small centre stick in the SAAB Gripen, which caused muscular fatigue as well as oversensitivity [Knotts et al, 1990]. A travel of 6 cm from neutral to the aft stop typifies a reasonable choice for a combat aircraft, with about half this for lateral travel, though “local tradition” usually determines the choice. In the Grumman X-29, this led to
an initial design with an aft travel of 20 centimetres, which was found to give sluggish handling. With the stick travel halved and the feel force stiffness doubled to maintain unchanged stick force per g, a far more satisfactory pitch control resulted [Anon, 1991].

Rudder pedals have always retained substantial motion. Pedal position feedback to the pilot through sufficient deflection avoids the potential problem of command discrimination where force sensing is used, as pilots may apply considerable pressure to both pedals simultaneously. Where the FCS performs the rudder coordination in flight for the pilot, the most stringent control demand on the pedals is probably in nose wheel steering at high speed under hard braking, which has notably unstable dynamics. Conventional pedal travels were typically ±7.5 to ± 10 centimetres. but successfully. Gibson [1995a] discusses the successful stability augmentation of steering systems that has enabled several fly by wire aircraft to use only about ±3.5 centimetres of pedal travel without difficulty.

Dual pilot aircraft, e.g. trainer, transport and some combat types, present some difficulty in the matter of the ability of one pilot to override the other if the sticks are not interconnected to move together. They are separate on the Airbus FBW types, but in the Boeing 777 conventional wheel inceptors are retained with the usual mechanical interconnection. In small aircraft this can be difficult, but they are connected by active torque drives in the Shuttle Orbiter.

Throttle controls have retained their traditional displacements on FBW aircraft. Typical motions are some 18 to 23 centimetres. The manner in which pilots manipulate thrust makes it impossible to contemplate the use of a rigid throttle. In combat they wish to be able to select maximum or minimum thrust instantly and cannot wait for the finite time it must take for a demand to integrate to the limit. In other situations, desired thrust changes are selected by known pre-learned displacement inputs, whereas it would be distracting to have to watch a display to learn when the command had been fulfilled.

6.3.3 Trimming
Trimming in fly by wire aircraft will usually be of the series type, the trim signals being computed and added to the stick commands. This avoids the mechanical complexity of a trim actuator on the otherwise simple FBW stick. As most manoeuvre demand systems require little or no trimming, except at low speeds, the usual pilot dislike of this type is minimised. Parallel trim will be feasible in the future when actively controlled torque spring sticks come into use, though it seems inappropriate where the stick travel is small.

Autopilots conventionally trimmed the stick and throttle positions automatically so that on reversion to piloted control they are in the correct position and there is no transient disturbance. The same principle is easily followed by software commands in a FBW system where there is no trim actuator. It is more difficult to deal with the throttles, which unlike the stick may be positioned anywhere over most of their travel at different times. On the FBW Airbus versions the throttles do not move in autopilot modes, a feature not viewed with much favour by many pilots. To avoid this an autotrim following actuator must be added, which has the advantage in a dual throttle aircraft under direct pilot control that the passive throttle can also be driven to follow the active one without any additional mechanical connection.

6.3.4 Feel and inceptor dynamics
It is certainly possible and usual to dispense with variable feel in fly by wire aircraft. The total stick travel can also be much reduced from conventional sticks of the past, as its position and
travel do not represent the actual surface positions explicitly. The chosen stiffness and maximum desired stick force will determine the total travel, because the maximum stick signal can be made to represent the maximum allowable command in any situation, giving protection from over stressing or stalling for example. Values around 14 N/cm, with perhaps half this gradient and maximum force for lateral control, have been very satisfactory. For transport aircraft with lower manoeuvre limits, higher forces are traditional and will be used even where limiting protection systems are used in a FBW aircraft.

The absence of a control circuit attached to the stick in a FBW aircraft, with its own spring force and typically small inertia, results in a high oscillatory natural frequency. This is almost certain to require up to about 50% to 70% of critical damping to be added for satisfactory operation. Such centre and side sticks of similar and related design in the British Aerospace EAP and the Lockheed YF-22 [Loschke, 1995] were reported to provide a transparent pilot interface with the aircraft. In the Lockheed F-117 pre-flight simulation, the desired quality of handling could not be achieved until an experienced test pilot realised that stick damping was required [ibid]. In the SAAB Gripen, lack of stick damping was a contributory factor to its initial problems.

6.4 Summary

The achievement of good handling qualities has always depended on a variety of mechanical design aspects as well as the basic aerodynamic and inertial properties of aircraft. At first the search was for appropriate pilot's inceptors, settling quite early to the universally adopted stick and pedals in various guises. Tuning of the handling by means of aerodynamic tabs and simple spring devices followed. Jet propulsion brought radical changes to aircraft design necessitating the use of hydraulic power actuation, artificial feel devices and stability augmentation. The role of the mechanical controls engineer assumed an equal importance to that of the aerodynamicist. It was possible to manipulate stick characteristics to a very large extent by mechanical means under difficult and diverse flight conditions of speed, Mach, centre of gravity, under severe design constraints of long control circuits, large accelerations, structural deformations, temperature changes, and so on.

The advent of fly by wire systems has replaced most of the mechanical complexity associated with good handling with computer control law complexity, requiring new skills once again. Despite this, the many lessons reviewed in this chapter need to be remembered to prevent avoidable handling problems and deficiencies being re-introduced by a failure to appreciate the importance of mechanical control properties and the needs of the pilot's physiology.
(a) Motion of input circuit opens valve, actuator cylinder follows after.

(b) Input circuit stationary, cylinder motion stops when valves close

Figure 6-1 Principle of hydraulic actuator control

(a) Friction tends to hold the valve open, moving the actuator and pulling the valve linkage and the entire control circuit with it until stopped by pilot or feel force resistance.

(b) Flow-induced force acts to centralise the valve, proportional to flow velocity and actuator rate. Equivalent to viscous damping in the control circuit.

(c) Friction can be counteracted by valve centralising springs, adding to effect of viscous damping. Use of single valve spring loads out circuit backlash between actuator and feel spring.

Figure 6-2 Power actuation effects on control circuit feel
Figure 6.3 English Electric Lightning flight controls
Conventional Mach 2 system from the 1960s
Figure 6-4 Typical tail angles to trim of the English Electric Lightning, from [Dickinson, 1968]

Figure 6-5 Typical tail angles per g of the English Electric Lightning, from [Dickinson, 1968]

Figure 6-6 Typical artificial feel force gradients dependent on dynamic (pitot-static) pressure. Cut-off to altitude variation only above M 0.9 and biassed by constant additional spring

Figure 6-7 Desired stick force per g increasing at low speeds. Taken from [Gibson 1978]
Figure 6-8 Control circuit friction characteristics

Figure 6-9 Effect of control circuit stretch on column to elevator angle ratio with q-feel unit at rear of aircraft (Boeing 747).
Figure 6-10 Non-linear stick to tail gearing (monotonic gradient)

Figure 6-11 Non-linear stick to tail gearing (non-monotonic gradients)

Aircraft design parameters and control hardware influence
Stick and surface positions retain constant relationship

Stick position remains in fixed position after trimming

(b) Series trimming

Figure 6-12 Parallel and series trimming methods

1.0 Normal acceleration [g]

\[ F_e \text{ [lbs]} \]

(a) Forces with unbalanced circuit
(b) Forces with longitudinal circuit balance added

(a) Minimum specified gradient [lbs/g]
(b) Maximum specified gradient [lbs/g]

Pull

Figure 6-13 Correction of stick force per g by addition of longitudinal balance mass to the control circuit in a feel system with non-linear gearing
Chapter 7
The development of new criteria from old data

7.1 Introduction

In this chapter, shortcomings are described in the state of the art of handling qualities and criteria that the author found when entering this field in the 1970s. This is followed by a description of the author's discovery of new metrics and understanding of the connections between aircraft dynamic response qualities and pilots' opinion, opening up a much wider scope than was available before for the design of good handling qualities in fly by wire aircraft.

A new handling qualities methodology was created by the author during the 1970s onwards, initially from studies of available flight research data, and then refined by direct experience in the development of the Panavia Tornado strike fighter, the BAe FBW Jaguar and EAP research aircraft and the Eurofighter 2000 fighter projects. The Tornado flight control system contains elements of both traditional mechanical and earlier analogue fly by wire technologies, building on much that was tried and tested. The FBW Jaguar, EAP and Eurofighter depend completely on new digital FCS with no connection to the past. Without such an evolution [Gibson op.cit.], these aircraft could not have been so successful.

The process began after a landing accident caused by a pilot induced oscillation in a prototype Tornado in 1976 [Gibson 1978]. There were no suitable criteria or requirements available at that time to explain the problem. The author identified a specific area in the dynamic attitude behaviour as the root cause, and defined a solution that has stood the test of time (discussed in Chapter 8). Although the Tornado has a fly by wire control system, its control laws lack the integrator function of later superaugmented systems (see Chapter 4), and retain a direct piloted connection with the control surface actuators. However, the laws contain additional feedforward and feedback filters and gains that rendered meaningless the conventional pitch handling requirements in the official specifications of the time [Anon, 1968]. The PIO was eliminated and the pitch handling at higher speeds was refined by scheduled variations of the previously constant feedforward filters, to reshape the pilot-controlled attitude frequency response.

Superaugmentation often had unforeseen effects on both the short term and long term response characteristics. Berry [1982] has been quoted on this subject in Chapter 1. As the response characteristics are dominated by the flight control system and the aerodynamic parameters tend to have greatly reduced influence, the simple modal parameters equivalent to the conventional frequency and damping are usually absent. There may be several poles and zeros clustered quite close to frequencies that are typical of the short period mode of conventional aircraft, and it is often impossible to decide which one should be taken to quantify the handling. Even if there is only one response mode with a suitable "short period" frequency, it may not be in angle of attack, the parameter conventionally operated on by pilots' control inputs. It will be in the parameter controlled by the maneouvre demand system, for example pitch rate.

The handling criterion specified for the superaugmented BAe FBW Jaguar was that its handling should be as acceptable as that of the standard Jaguar [Daley, 1984]. The initial PRD (pitch rate demand) control law design provided a fast well damped pitch mode at frequencies typical of the short period response, and was expected to be very satisfactory. When assessed in the first simulation, it was judged to be sluggish and lacking in maneouvability, although the attitude was well controlled. The cause was immediately identified in the slow initial angle of attack re-
sponse. This was quickened by a stick command feedforward path directly to the pitch control actuators to generate the desired initial pitch acceleration and rate response to the stick inputs. Lacking a formalised methodology at that time to optimise the handling of such a system, the selection of the modified laws was made on the fixed base simulator.

After these experiences, it was clear to the author that it had become of the utmost importance to be able to quantify desirable characteristics in ways other than by the traditional expressions of the short period mode parameters. The formal specifications for the short period, an area dominant in the pilot's appreciation of an aircraft's handling qualities, were based primarily on its frequency and damping. These are associated with the flight path response through control of the angle of attack, and lack detailed definition of the attitude response which dominates the pilot's actual perception of aircraft control. A search was made through the available literature in papers, journals and reports for results of handling qualities flight research. Only in-flight data was considered to be suitable, and this was mostly provided by variable stability aircraft studies such as those of Calspan as early as the 1950s. Other similar studies had been conducted in the 1960s specifically to prepare for a new specification [Anon, 1969]. Ashkenas [1984] lists 241 references mostly from this period, which was particularly rich in handling qualities research.

The importance of attitude dynamics
The physical parameters most closely controlled by and of interest to the pilot in the short term are the angular attitudes or rates and, generally more loosely and in the longer term, the flight path. Usually one or the other is paramount in a given flight task, though the limits on flight path responsiveness will be set by overriding limits on attitude behaviour. This is because the pilot usually perceives and exercises the overall control of the aircraft through an inner attitude loop. It is applied either to achieve precision control of attitude itself, for example in pointing the fuselage in a tracking task, or within an outer loop in a flight path or other displacement-related task, inferred indirectly from the attitude or directly from display or ground based cues. Figure 7-1 illustrates the general principle.

However, although an automatic system can operate two closed loops simultaneously, a pilot must generally divide attention between the tasks. For much of a flight the task is rather routine and the pilot can operate in an "open loop" manner. The generally predictable situations permit pilots to employ learned, pre-programmed control inputs (as does any car driver). They will mostly be concerned with directing the flight to a desired end point along a path, which may be very loosely defined. Other tasks will demand high precision. In combat the flight path may be altered coarsely and vigorously to achieve a position from which to exercise precision closed loop attitude control for weapon aiming. Precision control of flight path will be necessary to perform the landing, though the manner of piloted loop closure is somewhat complex.

Because the design of control laws to provide good handling cannot yet be reliably performed by incorporating optimal or advanced pilot closed loop theory, practical criteria are needed to define design limits on a variety of individual open loop aircraft response characteristics. The author was able to extract much new handling qualities information from the source material for those criteria that had remained hidden previously. The significant attitude parameters in were found to be the attitude dropback, pitch rate overshoot ratio, time to the first pitch rate peak, bandwidth and sensitivity, defined and discussed in this chapter. Their relationships are explicitely connected to the classical short period by the parameter $T_{a2}$ mentioned in Chapter 3, §3.2.2, but it made no appearance in formal handling qualities specifications until 1987. It is the inverse of the "lift parameter" $L_a$ given in Eq. (7.2) below.
This newly extracted information became the basis from which the author's methodology was developed. The low order response characteristics of conventional aircraft associated with satisfactory or optimum task performance and pilot opinion were identified to formulate criteria that could be applied to the design of flight control systems of any higher order.

From the outset, the author set specific goals for the criteria and design methodology:

a) They would have to be expressed in both frequency and time domains, as neither on its own give a sufficiently complete design definition.\textsuperscript{10}

b) They should in particular define limits on the shape of the responses in these domains without reference to numerical mode parameters or mathematical formula, so that they could be applied to any controlled response however high its order, and independent of the type of model available, i.e. parametric or non-parametric.

c) They should not depend on the assumption of any explicit pilot model parameters, as it is very difficult to apply these reliably.

d) They should enable the control design engineer to synthesise directly the handling characteristics desired for specific tasks as perceived visually or physically by pilots.

e) Their meaning should be equally clear to both pilots and control design engineers.

To satisfy these goals, the principles of Occam's Razor were followed, that "\textit{No more things should be presumed to exist than are absolutely necessary}" [Anon, 1979]. Put more directly, the fewest possible assumptions are to be made in explaining a thing, such as how the pilot's likes can be related to the characteristics of a response. Paradoxically, this eventually led to a greater number of handling parameters in the author's methodology than is found in formal or "official" documents, but each one clearly identifies a particular facet of handling that is usually lost within a "catch-all" such as the frequency, damping or so-called time delay of a response mode.

The time domain has often been considered unsuitable for handling qualities analysis. Partly this is because it is almost impossible for a test pilot either to obtain satisfactory responses in flight due to the difficulty in providing pure inputs typically in step form, or to be able to set up sufficiently steady and trimmed initial conditions to prevent contamination of the response. Even with the modern capability to input pure response forms by signal injection into the flight control system, it is still not really possible to obtain the necessary initial condition quality. To paraphrase Hoh [1988b], "\textit{In the time domain the region of crossover tends to be suppressed into the origin of the time response, which shows mainly the mid to low frequency characteristics. This makes the time response more appropriate to lower frequency tasks such as pursuit tracking and flight path control, while precision tracking with aircraft attitude is best specified with frequency response criteria}".

The author's experience has been that the contrary is true. The methods described in Chapter 10 for optimising the handling design of fly by wire aircraft, for which the desired response characteristics can be both closely specified and usually achieved, rely heavily on the time domain qualities examined in §7.4. Satisfactory behaviour in the crossover region is very visible in a $K_s$-like time response, while the high frequency characteristics associated with the angular acceleration, inadequately identified in other criteria known to the author, are easily quantified in the time domain. Although frequency responses are also employed in the author's methods, their

\textsuperscript{10}To a mathematical analyst, of course, the time domain differential equations and the frequency domain Bode, Nyquist and Nichols plots all describe the same system dynamics and so should be equivalent. It is not so for the pilot-centred methodology developed by the author, which depends on the graphical visualisation of "what the pilot sees" to design the handling qualities of a control law set.
use is principally to ensure the absence of high order degradation and pilot induced oscillation tendencies - and even these are detectable in the time response (Chapters 8 and 9).

An initial set of the author's criteria was in use by 1981 to develop the FBW Jaguar flight control laws in its later highly destabilised configurations. In flight the aircraft demonstrated exceptionally crisp and precise control, and it was much liked by the pilots. Although acknowledged to be near the upper limits of responsiveness, its handling was greatly superior to the standard Jaguar. It is thought to have been the first aircraft to enter flight trials with control laws designed specifically to ensure freedom from PIO, in which aim the criteria were successful.

The material formed the basis of an AGARD paper [Gibson, 1982]. It seems to have been widely regarded as seminal and parts became generally known as the "Gibson Criteria" discussed below in this and the following chapters. They have been referred to in many papers up to the present day [e.g. several in Tischler, 1996]. The criteria were refined and developed as discussed in the later Gibson papers. These papers provide much of the material in this thesis, particularly Chapters 7 to 10. Some discussion of this was incorporated as additional background material in the current military handling qualities requirements [Anon, 1987]. It was an inestimable benefit that their validity could be tested during the design of a further three fly by wire aircraft, the EAP [Hartley, 1988], the ASTOVL (Advanced Short Take-Off and Vertical Landing) research VAAC Harrier [Shanks et al., 1994] and Eurofighter 2000 [Smith, 1997]. The author's evolving criteria formed the only high order handling specifications for these aircraft, or in the latter case a principal part of them.

7.2 Contemporary high order criteria

In attendance at the AGARD conferences of 1978 and 1982 to present papers describing successful methods of dealing with high order handling qualities design in analogue and digital control systems, the author found a general conviction that the problem of "digital delays" was so serious that digital computing could not be successfully used in fly by wire aircraft. Despite the many high order handling criteria that had been offered, discussed, dismissed or otherwise criticised in a ferment of new but unresolved ideas, there seemed to be few ideas as to how to get rid of these troublesome delays.

In the USA, dissatisfaction with formal criteria for conventional aircraft was already felt in the late 1960s. Such high order effects as existed at that time were related mostly to the influence of actuator and feel system dynamics at higher frequencies, which did not impact greatly on the basic lower frequency response characteristics. As many new fly by wire aircraft began to display serious and unexpected handling difficulties during the 1970s and early 1980s, it became widely believed that these were "generic problems associated with high performance aircraft having sophisticated flight control systems" according to Berry [1982]. Berry [1983] also defined "large time delays" as one of the dominant characteristics of superaugmented aircraft.

All this may well have been true of many aircraft of the time. The author discovered that the problems were not generic but man-made, and could be readily prevented (Chapters 8 and 9).

7.2.1 Low order equivalent systems

The possibility of representing these effects in the time domain by time delays was investigated by DiFranco [1968] and by Neal/Smith [1970]. This idea of equivalent models was developed into the "Low Order Equivalent System", or LOES, in Hodgkinson et al [1976], by matching
complex high order frequency response dynamics with classical low order dynamics and a time delay to derive equivalent short period frequency and damping mode parameters. LOES was introduced as a formal requirement in the USAF specifications [Anon, 1980].

The frequency responses of attitude and normal acceleration at the instantaneous centre of rotation of the actual aircraft are matched as closely as possible with the simple two degree of freedom approximations with an added time delay, in the frequency range of 1 to 10 radians per second. The resulting equivalent short period frequency and damping are then compared with the standard requirements to assess the general handling, though these parameters strictly define only the flight path qualities. Phugoid matching is seldom performed, the concept of a high order phugoid having little meaning, and no time delay would be used.

Many discussions and papers of the time on this (and other criteria) are found in Crombie and Moorhouse [1979], Fuller and Potts [1982], A’Harrah et al [1978], Smith and Bailey [1982], and Moorhouse and Woodcock [1982]. These all throw much light on the handling problems. The author found LOES unattractive. It is not itself a handling qualities criterion, but merely provides a model that can then be assessed against the older specifications for conventional aircraft. There is a wide variety of possible conventional response shapes that satisfy these older specifications but do not actually guarantee good or optimum handling. Although the process depends on matching both the attitude and normal acceleration, the resulting short period parameters are related only to flight path control. Astonishingly, some LOES practitioners match only the attitude response, an exceedingly unreliable means to quantify flight path parameters.

The method may have difficulty in matching responses of different types, such as a heavily augmented pitch rate demand (PRD) with command filtering as noted earlier. PRD systems may have conventional attitude but unconventional normal acceleration responses that LOES cannot accommodate. There is a view that this difficulty is to be taken as a justification to discourage non-classical response types, but that would be an undesirable restriction since the latter have demonstrated significant superiority for some purposes. A simple check on the validity of the LOES method for specific cases is to calculate the time responses of the derived equivalent model. If these are close to the high order model response, particularly the flight path angle, then the method is valid.

LOES does not quantify the all-important characteristics of the attitude response in the inner loop through which the pilot conducts much or most of a flight. Subsuming the detailed characteristics of the actual time and frequency responses that directly influence the pilot’s perception of handling qualities into a different set of dynamics, LOES does not permit an explanation of these influences. As described in A’Harrah et al [1978], the proposal to allow the process to generate whatever value of $T_{52}$ gave the best equivalent match could result in unfathomable parameters with no physical meaning. This was a strong motivation for the author, along with some other members of the conference audience [Chalk, 1979], to reject the idea completely.

Also, except in research experiments where they were deliberately increased, sometimes to extreme values, the “digital delays” which caused such widespread concern in the early 1980s were never actually large enough to cause major problems. Quite similar problems would have occurred in an analogue implementation without any delays at all, the fault lying mainly in the control law structure and dynamics. The pilot induced oscillations demonstrated so often in modern fly by wire aircraft result from a number of characteristics, mostly caused by undesirable lag as discussed in Chapter 8. The idea that such effects can be simulated by the different time delay characteristic has always seemed unrealistic to the author.

*The development of new criteria from old data*
Many unconventional but legitimate response characteristics of superaugmented aircraft often cannot be represented by an equivalent low order system [Hoh, 1996, Marchand, 1993]. Usually it can only match a gain peak by a low damping ratio, even if the response is highly damped. It may have difficulty in finding a mode frequency match in overdamped PRD systems. Reconstruction of a transient response from a LOES match sometimes bears little resemblance to the original, a highly significant failing. LOES appears to be applied most successfully where the characteristics are quasi-conventional with small "time delays". A manoeuvre demand system can be designed specifically to resemble the classical response in order that the classical specifications can be followed, but in doing so some benefits may be lost.

7.2.2 Neal and Smith
Some contemporary criteria did attempt to address the analysis of actual characteristics directly. One proposal that assisted the author in the search for a new methodology was that known universally as "Neal and Smith" [1970], described more recently in Anon [1991]. Making no assumptions about the order of response, it quantified the handling by the lag and lead equalisation necessarily applied in the crossover pilot model (Chapter 5) to satisfy pitch attitude tracking performance indicators, specified as the closed loop droop, resonance and bandwidth discussed further in Chapter 8. As it required assumptions to be made about these factors and the pilot model that were not yet clearly established, it was felt to be unsatisfactory as a synthesis tool.

The Tornado control laws already contained command path feedforward filters identical in form to the usually assumed pilot model equalisation. It was obviously more effective to utilise these directly to produce $K/s$-like aircraft response characteristics for good closed loop tracking without pilot equalisation. Illustrated in Gibson [1978], this philosophy has been the basis for the development of the author's methodology.

7.2.3 "Ralph Smith"
Another criterion set useful to the author was that of Smith and Geddes [1979]. Smith's "No tracking" hypothesis, referred to in §5.2.2, amounted to providing a sufficiently $K/s$-like aircraft response so that the pilot needed little conscious tracking effort. Several measures of handling qualities were proposed that depended on the response shapes in both the time and frequency domains of the pitch attitude response. There was no assumption about the order of the response and no reference to the modal parameters of short period frequency and damping. The ideas led the author to study the correlation of attitude and flight path time response shapes with pilot opinion in the studies noted above, from which characteristic $K/s$-like limits in the time domain could be derived and synthesised by filtering more directly than by Smiths's criteria.

7.3 Expanding the information data base

The natural frequency and the relative damping ratio of the short period motion define the classical short term angle of attack response to control or turbulence inputs. The natural frequency is related to the stiffness/inertia ratio and so also to the pitch acceleration characteristics, and should not be confused with the damped frequency which seldom appears in handling qualities work. A clear relationship to the attitude response is not identified, however.

7.3.1 Time response parameters
The first major revision to the US military handling qualities specifications [8785B], Anon 1969] defined upper and lower limits on the natural frequency to prevent excessively abrupt or sluggish response, outlined in Figure 7-2(a). This was based on Bihrl [1966], a study of preci-
sion approach path control. This proposed that the pilot’s ability to predict a flight path response was related to the ratio of the initial pitching acceleration to the steady state normal g following a step control input. This ratio, the Control Anticipation Parameter or CAP, connects the short period natural frequency, flight condition, wing loading, lift slope and manoeuvre margin.

The approximations derived by Bihrl [1966] and also given by Chalk [1969] are, for zero tail lift effect:

\[
\text{CAP} = \frac{\dot{q}}{n_{st}} \approx \frac{\omega_{sp}^2 W}{L_{\alpha}} \approx \frac{\omega_{sp}^2}{(n/\alpha)}
\]  
(7.1)

where \( L_{\alpha} = \frac{(\rho V^2 SC_{La})}{1/T_m} \)  
(7.2)

and \( \dot{q} \) = initial pitching acceleration
\( \omega_{sp} \) = short period natural frequency
\( n_{st} \) = steady state normal acceleration
\( n/\alpha \) = normal acceleration per radian angle of attack
\( W \) = weight
\( S \) = wing area
\( C_{La} \) = wing lift slope

The damping was specified separately. In the most recent specification standards [Anon 1987], the same frequency limits were explicitly expressed as CAP values, combined with the relative damping as in Figure 7-2(b). The frequency boundaries in both forms of the criterion are directly associated with the classical aircraft short period response. Lines of constant CAP are equivalent to lines of constant manoeuvre margin, as was shown by Bihrl. A proof is also given in Cook [1997].

Thumbprint plots are lines enclosing regions of similar pilot comments on the short period frequency and damping plane. At first these regions were labelled “best tested”, “good”, “fair” or “poor”, before the now-standard Cooper-Harper pilot rating scheme was developed. They were generated in a number of in-flight variable stability research programs of the 1950s and 1960s, the majority being performed by Calspan at Buffalo, USA. The results formed much of the basis for the short period specifications in Figure 7-2. The early examples in Figure 7-3 show only the lines enclosing “best tested” areas.

It is seen in Figure 7-3(a) that the selection of short period frequency for good handling was different in each case, although there was good agreement on damping limits. The CAP parameter was intended to improve the correlation, but as shown in Figure 7-3(b) this was not very well achieved. It did provide a more logical structure for the selection of suitable natural frequencies, although it was soon apparent that its upper limits could be far too high and it was certainly possible to obtain good handling at values below the lower limits.

CAP has been widely misunderstood, and is often thought to be a measure of quality of the attitude response. It seems to be generally forgotten that its origin was a measure of predictability of the landing approach of carrier aircraft, a task dominated by flight path control. The initial pitch rotation acceleration is that of the angle of attack, or AoA, the parameter mainly defined

\[11\] The A’Harrah results were from a ground based “g chair” simulation of low altitude high speed flight, included because they appear to have been thoroughly validated by flight experience. They caused great controversy at the time, probably because of a misinterpretation of their significance to attitude control. It appears that attitude was available only as a very minor cue, and the experiment was actually dominated by a height control task as discussed in Chapter 9.

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by the short period mode that leads directly to the normal acceleration and the corresponding flight path changes. Attitude is the continuous summation of the AoA and the flight path angle responses. The attitude response characteristics can be hugely different for a constant CAP.

**Generic time response parameters**

*When the pilot moves the stick, something happens. The question to be answered is why the pilot likes or dislikes the result. A time response history is an excellent graphical form in which to present and study handling information, and the simplest input is a step. Although a pure step input cannot be applied in practice, in calculations it ensures that only the aircraft characteristics are studied without the confusion caused by the introduction of filtered inputs in an attempt to represent a realistic pilot action. The author introduced this analysis concept in [Gibson, 1982].*

The generic result of a step control input is shown in Figure 7-4. The responses represent three arbitrary aircraft with the same short period mode frequency and damping, but with different wing loading and lift slope and so different values of $T_{\theta_2}$, for which see §3.2.2 and Eq. (7.2) By choosing appropriate speeds and magnitude of control input, each can have an identical flight path angle time response. The initial pitch rate transient is primarily the angle of attack (AoA) rate, different in each case because the AoA reaches different steady values in the same overall response time. As the AoA rate decays to zero, the flight path angle rate increases to its steady value, forming also the steady pitch rate which is the same in each case. The initial pitch rate may therefore overshoot the steady value by a large, small or zero amount depending on the AoA needed to generate the flight path angle rate. The corresponding attitude response, which is formed by the addition of the AoA and flight path angle, has therefore quite a different character in each case.

As discussed in Chapter 3, the phugoid motion is neglected. A non-classical feature which has been added is a small lag in the onset of pitch acceleration, a factor of profound importance in many handling problems of recent years which is discussed in Chapter 8. This lag is unavoidable when even the best quality actuation is used to move the control surfaces, but as will be seen its effect has sometimes been exacerbated by the adoption of command lag filters.

In Chapter 3, Figure 3-3, a number of time response relationships between attitude, flight path angle, pitch rate, flight path angle rate, g and velocity were given, neglecting tail incremental control lift effects. Figure 7-5 adds the new time response metrics of attitude dropback ratio and flight path time delay with their relationship to the classical short period parameters. The path time delay $t_\gamma$, so-named because it notionally represents the path angle response by an initial delay followed by a constant slope, is defined by the time constant of the second order short period motion oscillation in Eq. 3.8, hence

$$t_\gamma = 2\zeta_{sp}/\omega_{sp}$$  \hspace{1cm} (7.3)

$T_{\theta_2}$ is effectively the time lag constant between attitude $\theta$ and flight path $\gamma$, hence

$$T_{\theta_2} = DB/q_{ss} = 2\zeta_{sp}/\omega_{sp}$$  \hspace{1cm} (7.4)

or $DB/q_{ss} = (T_{\theta_2} - 2\zeta_{sp}/\omega_{sp})$  \hspace{1cm} (7.5)

Since these quantities are also physically measurable in calculated responses, complex "high order" systems can be compared directly by this method. It is more difficult to set up the steady conditions to obtain the characteristics in flight. A test record from the Delft University of Technology Citation in Figure 7-6 shows that with care this could be done reasonably well (but
note that there appears to be a time shift between the attitude and AoA traces which should initially be nearly identical). However, as the methodology is intended for design, difficulties in its application to flight test assessment are not relevant.

One feature should be particularly noted. When the attitude dropback is zero, the attitude time response tracks along the "K/s" line after the initial pitch rate transient is complete. The significance of this in terms of the crossover model for precision closed loop attitude tracking was noted in Chapter 5. The attitude and flight path responses cannot be separately adjusted unless direct lift control is employed. A shorter flight path delay increases the attitude dropback ratio, and reducing the dropback ratio increases the flight path delay. In the classical response type, these would result respectively from a combination of an increase or decrease in the short period frequency, and a decrease or increase in damping.

This form of analysis was applied by the author initially to data from the Calspan LAHOS, or Landing Approach High Order Systems, [Smith, 1978] and "Fighter flying qualities" flight experiments [Boothe et al, 1974]. These simulated a number of nominal short period modes and added a variety of lag or lead filters to the stick command to simulate high order effects. Although the tasks included precision tracking, no performance data were measured, in accordance with the normal practice for handling qualities evaluations. These are always based on pilot ratings and comment, because skilled pilots are able to maintain good tracking scores even with poor handling, although at the expense of increased workload and poor opinion. The time responses for each case were calculated and the metrics in Figures 7-3 and 7-4 were obtained.

Because most cases were degraded by the high order simulation, only a rather small number of satisfactory cases was found. Based on these, the following findings were given by the author in Gibson [1982], and they became widely known as part of the "Gibson Criteria":

(i) The pitch short period characteristics of primary importance to good handling are: pitch attitude dropback; pitch rate overshoot and time to first peak; flight path time delay; overall response time to reach steady conditions.

(ii) Negative attitude dropback (i.e. overshoot) was associated with sluggish, unpredictable response both in flight path control and in tracking.

(iii) Attitude dropback ratio from 0 to about 0.25 seconds was excellent for fine tracking and was associated with comments typified by "The nose follows the stick".

(iv) Increasing attitude dropback led to abrupt response and "bobbling", from "slight tendency" to "continuous oscillations" in tracking tasks. Sometimes this was called pilot induced oscillation (PIO) but did not cause concern for safety.

(v) Attitude dropback had little effect within the range tested upon gross manoeuvring without target, landing approach or flight refuelling, provided it was not negative.

(vi) CAP up to 3.6 rad/sec²/g was satisfactory for gross manoeuvring without a target, but was unsatisfactory above 2 rad/sec²/g for the landing approach, above 1.0 rad/sec²/g for fine tracking, and below 0.28 rad/sec²/g for any task.

(vii) The pitch rate overshoot seemed to qualify the dropback behaviour, a ratio greater than 0.0 making a dropback ratio as small as 0.25 seconds unacceptable.

(viii) Small values of flight path delay were associated with excellent flight refuelling control, but were not essential for good gross manoeuvring and did not on their own ensure predictable behaviour.

The development of new criteria from old data
These observations provided much new insight into the detail of the dynamic response characteristics that had always influenced pilot opinion. However, their real utility was associated with the ability to shape the handling responses of fly by wire aircraft in a way not possible in previous aircraft types, instead of having to accept the results of aerodynamic and inertial properties. A discussion of the author's techniques in this regard is given in Chapter 10.

Others have also adopted the author's principle, notably Bland et al [1987] who describe how the pitch attitude tracking of the S/MTD F-15 Technology Demonstrator was greatly improved by pre-filtering the pitch command signals to reduce the pitch rate overshoot and attitude drop-back. The normal acceleration response was thereby slowed somewhat, with lengthened flight path time delay as expected from Figure 7-5, but this compromise was acceptable to the pilots.

7.3.2 Frequency response parameters

The author's second approach was to determine limiting boundaries of good handling on Nichols plots. This was done simply by plotting the open loop attitude to stick force frequency responses of the same Calspan research data from which the time response criteria above were extracted [Smith, 1978 and Boothe et al., 1974], and drawing enclosing boundaries around them. Because the author's aim was to develop criteria for the design of good piloted closed loop handling qualities in fly by wire aircraft in which specified characteristics could be obtained, only the very small number of cases with the best Level 1 piloted closed loop qualities were included. No boundaries for Level 2 or Level 3 responses were derived. Separate boundaries for the landing approach and for "up-and-away" flight are shown in Figure 7-7 [Gibson, 1982].

The boundaries are actually shape templates relative to specified control points on the Nichols plot responses. Their amplitudes are not absolute but relative to these control points. For landing approach cases, the 0 dB line of the template is moved over the -120° phase angle point, and for up-and-away cases it is moved over the 0-3 Hz point. The templates can be drawn on transparencies and moved up or down over the actual Nichols plot for a rapid manual evaluation, but they are also readily drawn automatically on computed printouts.

Although they were derived with no pre-conceived theoretical expectations simply as empirical shapes wrapped closely around certain given frequency responses, nevertheless the templates well represent the desirable features of the aircraft response for the crossover model described in Chapter 5. That is, there is a region of $K/\omega$-like response at low frequencies, contained between the boundary lines "1" and "2". At higher frequencies beyond the crossover, boundaries "4" and "5" ensure adequate attenuation to maintain a reasonable closed loop gain margin. An additional boundary "3" is included to permit the presence of limited amounts of attitude drop-back, which always cause the response plot to divert to the right in a kink or "knee" before moving leftward and downward. This dropback allowance is larger for the landing approach, where control of flight path is the dominant task, than for up-and-away flight where precision attitude control is usually more significant. (See findings (iii), (iv) and (v) in §7.3.1 above.)

Obtained as they were from such a limited data base, these results were simplistic. Since the boundary lines are straight, naturally curved response lines could cut through them locally, and this sometimes led to arguments about whether this violated the "requirement" or not (they would not). In the dropback allowance region, it was impossible to tell whether the dropback was acceptable or not, since its actual value cannot be deduced from the frequency response. The boundary line "4" is unnecessarily restrictive, as there is no inherent reason from closed loop considerations why a response to its right should not be acceptable.
It also became clear to the author that the up-and-away boundaries, obtained from data in a small speed range, did not adequately represent the unavoidable natural variations in the attitude frequency response "shape" over a wide range of speed and altitude. These are indicated in the generic sketches in Figure 7-8. In a fly by wire aircraft it is possible to force the piloted attitude response to follow a narrowly defined shape, but it may not be desirable. As suggested in Figure 7-4, a large dropout ratio can be a consequence simply of the need for a large rapidly applied angle of attack, for example at high altitude, to generate a fast normal acceleration response. This was also discussed in Chapter 4, §4.3 and Figure 4-2.

As a result, the frequency response boundaries given in Figure 7-7 have not played much part in the author's subsequent design methods. Because design optimisation of pitch handling qualities proved to be very satisfactorily accomplished in the time domain, use of frequency domain criteria became largely confined to ensuring the prevention of pilot induced oscillations by elimination of excessive high order dynamics, as will be discussed in Chapter 8, and to adjusting the pitch sensitivity discussed in §7.6 below. Nevertheless, these boundaries did succeed in drawing attention to the importance of frequency response shapes, and they have been quoted in references such as Blight et al [1996] and Kendall [1996] where control laws were satisfactorily re-designed to take account of them them.

7.4 Extending the time response criteria to enhance the "thumbprint" plots

A considerable amount of data was also published in earlier literature, usually giving only pilot ratings without comment, but they are also valuable because they covered a wide range of classical behaviour with no added high order effects. Instead of requiring a calculation of each individual time and frequency response for analysis, the author's new handling qualities could be obtained directly from the modal short period and lift parameters. The latter, \( T_{\theta_2} \), was seldom noted in early experiments but it can be approximated with sufficient accuracy from the features of the aircraft (e.g. from "Janes All the World's Aircraft"). With the wing loading and the estimated lift slope from the planform, \( n/\alpha \) is obtained by \( C_{L_{\alpha}}/C_L^{12} \), and then \( T_{\theta_2} = V/g(n/\alpha) \) from Chapter 3 §3.2.2.

New forms of presentation were needed for such analysis, however. The author investigated the possible addition of the Figure 7-4 metrics to many of the proposed criteria in the research data leading up to publication of Anon [1969], none of which contained the attitude parameter \( T_{\theta_2} \) crucial to the specification of attitude handling. The most widely useful format proved to be the original thumbprint of the type illustrated in Figure 7-3. Figure 7-9(a), one of the earliest published thumbprints [Newell and Campbell, 1954], was the first to which the author applied this treatment. Note that despite the constant stick force per g the pilot comments span an extreme range of attitude and flight path sensitivity, a subject discussed further in §7-6. Figure 7-9(a) is unable to illuminate the reasons for this. Figure 7-9(b) shows how well the pilot's comments can be explained by revealing attitude and flight path information hidden behind the traditional metrics of short period frequency and damping alone. These results were for a constant \( T_{\theta_2} = 1.0 \).

The "linear PIO" line represents a boundary first given in Ashkenas et al [1964], below which piloted closed loop instability was likely. It was considered somewhat academic since it could not be crossed or even approached closely by reasonable aerodynamic variations. However, it was crossed here and in many other variable stability aircraft flight experiments by artificially

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12 This simplified derivation, neglecting the elevator lift effects, arises from equating the lift coefficient \( C_L \) to 1g and dividing it into the lift slope \( C_{L_{\alpha}} \) to obtain \( n/\alpha \) in g per radian units.
created combinations of frequency and damping through feedback control means. These experiments provided an unremarked pointer to later PIO problems introduced by improper design of fly by wire control systems, some of which might have been avoided by "remembering the past" (Chapter 2). It will be discussed further in Chapter 8.

**Frequency units**

The short period frequency has been expressed in Herz units in Figures 5-3, 7-3(a) and 7-9, rather than in the conventional radians per second. The early experimental data from the U.S.A. were always expressed in Herz. It is the author's view that this measure, based on complete cycles, is the natural choice when considering response characteristics in terms of piloted handling where no mathematical manipulation is involved, as is usually the case. Herz units are still used in most of the author's design methodology and appear as appropriate in the rest of this document. With the publication of Anon [1969], use of radians per second was standardised in formal criteria. The additional frequency scales in Figure 7-9(b) have been added for the reader's convenience. One scale is in radians per second. The other is in units of \( \omega_p T_{Q2} \) radians, numerically identical in this case because \( T_{Q2} \) had the value of 1.0 in this example.

**7.4.1 The introduction of \( T_{Q2} \) in formal short period criteria**

The influence of \( T_{Q2} \) as a handling parameter had been discussed in the 1960s, e.g. by Mclauer et al [1960]. There it was noted that its value had always been favourable to aid short period attitude control (in essence, it was typically similar to the flight path time delay \( t_y \), hence ensuring \( K/s \)-like dynamics by Eq. 7.4), but in future aircraft its changed values would drastically alter the relationship between load factor and pitching responses and require modified pilot behaviour. At that time, there was little coverage in flight or simulation, and considerable effort would be needed to investigate it. Other researchers also proposed the inclusion of \( T_{Q2} \) in handling criteria. The compilers of the revision to the handling qualities specification Anon [1969] were unconvinced by such evidence and excluded it in favour of flight path-dominated criteria alone.

Figure 7-10 shows a thumbprint plot by Shomber and Gertson [1965] from research in the above period. This result was from simulation of the landing approach of a large transport aircraft, and is for a fixed \( L_q = 0.5 \). Instead of the usual frequency and damping format, the thumbprint used \( L_q/\omega_{sp} \) (or \( 1/\omega_{sp} T_{Q2} \)) to replace \( \omega_{sp} \). The author found that this format was ideally suited to the addition of all the Figure 7-4 metrics, excluding the pitch acceleration peak lag \( t_y \) which is a high order feature that cannot be related to classical response forms. In addition, the "linear PIO" boundary mentioned above in §7.4 and Figure 7-9(b) is shown.

This format is not completely universal, however. The contours of some of the response metrics take values that are fixed. These are \( q_m/q \), zero dropout, and linear PIO, shown by solid lines, which are all proportional functions of \( \omega_{sp} T_{Q2} \). The other metrics of \( t_y \), non-zero dropout and \( t_y \) shown by dashed lines take values proportional to \( T_{Q2} \). Conveniently, this allows the dashed lines to be re-labelled with appropriately scaled values without being otherwise altered.

It can be seen that the line enclosing the satisfactory pilot ratings (better than 3.5) is bounded quite definitively by differing response characteristics in different places around the boundary. Starting at the top, the limit is excessive \( t_y \) (time to the first pitch rate peak). Moving clockwise, the next limit is excessive \( t_y \) (sluggish flight path) and/or attitude overshoot (negative drop-back), then a too short \( t_y \) and too large \( q_m/q \) (pitch rate overshoot), then the linear PIO region and finally a minimum short period damping level.
A major proposal for revision of military aircraft handling qualities specifications was undertaken by Hoh et al [1982], which formed the basis of Anon [1987]. By this time the importance of attitude parameters to handling qualities was more widely recognised, and the new criterion shown in Figure 7-11 was included. This is actually the inverted form of Figure 7-10.

The author's initial assessment of this new criterion included the addition of the linear PIO and zero dropback lines (shown dashed). This indicated that the proposed lower Level 1 limits to $\omega_{p}T_{\theta 2}$ were approximate at best (as many criteria including the author's own often are). The author also added the thumbprints of best tested results from two in-flight variable stability experiments, a naval carrier approach task [Eney, 1969] and an early Calspan fighter task [Harper, 1955], indicated previously in Figure 7-3. It was obvious that the proposed lower boundaries were not well chosen for either the Cat. A (combat) case or the Cat. C (landing) case. The error arises from the effect on the short period frequency, at a constant value of $\omega_{p}T_{\theta 2}$, for the wide possible range of the parameter $T_{\theta 2}$. The latter varies from less than 0.5 seconds at high sea level speeds to 3 or even 4 seconds at high altitudes and low airspeeds, allowing wide variations in the minimum frequency which is significant in its own right as a flight path parameter.

With a typical $T_{\theta 2}$ of around 2 seconds for many aircraft in the Cat. C landing cases, as in the Eney experiment, precise flight path control would be difficult at the minimum short period frequency of 0.7 rad/sec allowed by the suggested lower limit. This clearly ought to be higher for such aircraft. The author's own interpretation of the data given in Hoh et al [1982], from which the Figure 7-11 proposed limits were derived, suggests strongly that a more accurate lower limit for Cat. C would be a $\omega_{p}T_{\theta 2}$ of about 2.0. As discussed later in this chapter, an even higher limit of 3.0 would be necessary to ensure a minimum $\omega_{p}$ of 1.5 rad/sec for precision flight path control. On the other hand, the Harper result shows that the Cat. A lower limit was set too high for the reference aircraft.

Figure 7-12 shows the author's time response metric lines formulated in the Figure 7-11 format, drawn for a $T_{\theta 2}$ of 1.0. As noted above, the solid lines show the fixed values of pitch rate overshoot, zero dropback and linear PIO. Values of the dashed lines of peak pitch rate time, non-zero dropback ratio and flight path time delay are obtained for other $T_{\theta 2}$ by proportional scaling.

Figure 7-13 shows the Eney [1969] results in the same format. It can be seen that the Level 1 region is bounded by lengthening $t_q$ and should not extend downwards to the original lower criterion limit given in Figure 7-11. There were insufficient data to complete a closed boundary but at the right there is a hint of a limitation at approximately zero dropback associated with a $t_q$ of 2.0 seconds, which is certainly rather long for a task requiring precision path control. The upper limit coincides with a short $t_q$ and large $q_{m}/q$. The data point at the location "K" is of particular interest, with Level 3 pilot ratings of 7 and 8, or "Considerable pilot compensation is required for control". It lies within the Level 1 region of the Harper [1955] result, with a $T_{\theta 2}$ of only 0.6, in Figure 7-11.

It is clear that the criterion as given in Hoh et al [1982] cannot be relied on as originally presented because it neglects the strong and variable attitude effects discussed here. It requires consideration of the additional Gibson time response metrics before a proper appreciation of the handling can be obtained from this general format. The metric lines show that the limits to good handling are then quite well defined. Similar analyses of other available experimental results were given in Gibson [1986], which resulted in the generic handling boundaries in Figure 7-14 associated with satisfactory values of the time response features shown in Figure 7-4.

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Smith and Geddes [1979] proposed satisfactory upper and lower limit values to $t_q$ of 0.9 and 0.2 seconds. The author modified these in his own early work to 0.9 to 0.3 seconds for up-and-away flight and 1.5 to 0.5 seconds for the landing case, based on study of results from Boothe et al [1974] and Smith [1978]. This appears to explain the tendency for the thumbprints to be located further down the figure as $T_{92}$ decreases. For a given aircraft, $T_{92}$ decreases with speed while $\omega_{up}$ increases, their product remaining roughly constant. The $t_q$ values associated with a given level of $\omega_{up}T_{92}$ therefore decrease with speed. To reduce this variation, $\omega_{up}$ has to decrease at a rate greater than inversely proportional to speed. This is not an aerodynamically natural feature, but in the experiments the pilots had a wide choice of dynamics from which to choose the best, a luxury not given to pilots of production aircraft. Another interpretation is that for aircraft with increased wing loading or reduced lift slope, and hence with increased $T_{92}$, a lower $\omega_{up}$ is desirable at a given speed to prevent excessively short $t_q$ with a large dropback ratio.

However, the thumbprints should never penetrate below the "linear PIO" line, nor much below the zero dropback line. The latter result will be conditioned by whether flight path or attitude is the more important parameter to the pilot in any given task.

The consistency of the parameter values around the thumbprint boundaries is as good as can be expected in handling qualities experiments (considered excellent if the opinions for each case vary by no more than one rating point in either direction, a total of two thirds of a Level range). They agree qualitatively with the findings in §7.3.1 from Gibson [1982]. It must be realised that the criterion format is applicable only to classical low order response dynamics, and then only if the metric values are inserted appropriate to the correct $T_{92}$. A high order fly by wire aircraft response has no pre-ordained relationship between the various metrics and any nominal frequency and damping modes, but it will always exhibit the time response features of Figure 7-4.

### 7.5 Frequency and time domain criteria: Bandwidth, a general responsiveness measure

The concept of bandwidth is well known in servomechanism and electronic devices. It can also be applied to aircraft response qualities, but it is defined in a different manner. Whatever the task, aircraft must be sufficiently responsive, yet not excessively so, for the pilot to be able to couple with it in the task performance. This can be measured both by the dynamic qualities - the "shape" of the response - and by the gain or amplitude of the response to any control input. If it responds too quickly or if the response is too great, the pilot will complain of overcontrol tendencies and of PIO. If it is too slow or the response is too small, the pilot will complain of sluggish control and may overdrive the aircraft in a form of slow PIO.

#### 7.5.1 Manual attitude control

The author proposed a bandwidth concept for pitch attitude control in the landing approach [Gibson, 1982], based on the premise that open loop attitude dynamics due to stick force that were useful to the pilot were confined to frequencies without excessive phase lag. The limiting frequency was taken at a phase angle of -120°, the author's definition of bandwidth frequency. Study of the LAHOS data [Smith, 1978] suggested a Level 1 range from 0.2 to 0.5 Hz.

The same bandwidth concept was also developed as a handling parameter in the AFTI YF-16 decoupled direct force control experiments [Hoh et al, 1982a]. This proposed criteria for independent control of the six degrees of freedom, for which conventional metrics were unsuitable. It was extended to pitch attitude control due to stick force in normal flight in the "Bandwidth Criterion" [Hoh et al 1982, 1982a]. Figure 7-15 defines bandwidth as the lower of a nominal
“open loop gain or phase margin” frequency. Figure 7-16 shows the proposed limits (combined with the phase delay \( \tau_p \), a measure of additional higher order control system phase lags sketched in Figure 7-15 and discussed in Chapter 8). Note that the classical \( \theta(s)/\eta(s) \) transfer function does not exceed a phase lag of 180 degrees. In this case the gain margin cannot strictly be defined. If the phase lag exceeds 180° even by only small amounts, due to actuator dynamics for example, the gain bandwidth is defined but is almost always higher than the phase bandwidth. The latter is then taken as the bandwidth.

While there is an analogy with the customary servomechanism gain and phase stability margins of 6 dB and 45° (or 60°) respectively, it cannot be taken too far. It is best to quote Hoh et al [1982]: “Strictly speaking, bandwidth in pitch involves \( \theta/\theta_C \), a closed loop describing function of pilot/vehicle response. Here, however, there is no assumption about a pilot model, and ‘bandwidth’ is specified in terms of the aircraft-alone gain and phase margins. … this describes the pilot’s ability to double his gain or to add a time delay or phase lag without causing an instability.” They can be thought of in a simpler way as defining the range of open loop aircraft response frequencies which the pilot can use easily in performing a task without encountering closed loop stability problems. As will be discussed in Chapter 8, the lack of any reference to the absolute pilot closed loop gain is a serious deficiency preventing the criterion from being used as a stability margin measure. The overall gain is a major contributory feature in PIO.

The length of “shelf” in the pitch attitude Bode amplitude plot (the distance between the numerator and short period break points in Figure 7-15) is \( \omega_p T_{02} \). It can be seen from the expression for dropback in Figure 7-5 that to achieve zero dropback (the “\( K/s \)” value), the shelf width must equal \( 2 \omega_p \) exactly. A shelf wider than this results in dropback. An increasing short period frequency increases the bandwidth, and for constant \( \omega_p T_{02} \) the nature of the dropback becomes more and more abrupt and unsatisfactory. A high \( \omega_p \) and a wide shelf create excessive attitude dropback and pitch rate overshoot effects, significant time domain characteristics readily identified qualitatively in the frequency domain. The corollary is that if a time response has large dropback and pitch rate overshoot, the frequency response will have a wide Bode shelf or its equivalent “knee” in a Nichols plot, which impact directly and unfavourably on the crossover region dynamics. If the time response has near zero dropback and low pitch rate overshoot, it is a certainty that the frequency response exhibits desirable \( K/s \)-like crossover characteristics.

This close connection between frequency domain and time domain criteria is illustrated by the example reported by Quinn [1984], as shown in Figure 7-17, and discussed by Gibson [1991a]. The intention of this test pilot school exercise was to see if changes to the stick forces and gradients, with a short period frequency at the upper allowable limit of CAP and hence with a high bandwidth, could change the pilot ratings from satisfactory Level 1 to unsatisfactory Level 2. Despite the nominal Level 1 conditions (according to Figure 7-3), every result was Level 2 because precision attitude control was impossible, even when the short period frequency was reduced substantially. The table shows excellent flight path but extremely poor attitude response characteristics. These results could certainly have been predicted from the large and rapid attitude dropback in the time response alone.

At the other end of the scale, low bandwidth may or may not be a sign of sluggish handling. Much lower bandwidth limits were given (without comment) in Mitchell et al [1990], which are shown in Figure 7-16. In a comprehensive tracking experiment by Wilson et al [1990], all the results including many with very satisfactory Level 1 pilot ratings of 2 to 3 lay well below the nominal Level 1 bandwidth of the Bandwidth Criterion in the region categorized as Level 2, Figure 7-16. (To avoid confusion with other possible bandwidth definitions, the Hoh criterion...
discussed above is always referred to as the "Bandwidth Criterion", the style by which it is universally known and recognised in the handling qualities community.)

It was shown by Gibson [1991a] that those internal elements discussed earlier, i.e. measures of attitude, pitch rate and flight path behaviour, could have more influence on the tracking performance than the bandwidth. This is illustrated by the author's piloted simulation experiment examples from Gibson [1985b] in the Nichols plot of Figure 7-18. Tracking performance was measured by the proportion of time within a small tracking error during many test runs, in each of which the target had to be re-acquired and tracked several times. The attitude droopback of the nominal Case A was eliminated in Case B by reducing $\omega_p$ and the use of a stick pre-filter, and in Case C by a pre-filter alone. The tracking performance of Case C was markedly improved relative to Case A despite a reduction in the bandwidth to less than the Bandwidth Criterion minimum for Level 1 combat handling in Figure 7-16. However, its performance was much better than Case B, undoubtedly because of its substantially higher bandwidth.

It may be noted in passing that Case B and C had effectively the same flight path bandwidth, expressed as the flight path time delay, yet their attitude bandwidths differed substantially. This phenomenon, impossible in conventional aircraft dynamics where both bandwidths are directly influenced by the short period dynamics, is a result of using a pre-filter technique to enhance the pitch handling. The subject will be addressed again in Chapter 10.

As usual, results from a limited flight regime are misleading. A fixed range of bandwidths is not a natural feature of conventional aircraft. It varies both with speed approximately in proportion to $\omega_p$ and with altitude, typified generically in Figure 7-19 [Gibson 1990]. Considerable variation in bandwidth is therefore natural if the speed range is wide, and no simple conclusion about bandwidth limits can be extended from one flight condition to another. "Islands" of bandwidth seem to resemble the short period frequency and damping thumbprints, inconsistent between different experiments but characterised by consistent internal elements. It is also likely that when these elements are directly shaped to a non-classical form for optimum results (as the author demonstrated in the Figure 7-18 work and will discuss further in Chapter 10), upper bandwidth limits as suggested in Figure 7-16 can be removed, the tracking performance improving with increasing bandwidth if the droopback ratio is kept small.

Notably, the recent version of the Bandwidth Criterion [Mitchell et al, 1994b, Hoh and Mitchell, 1996], proposed for incorporation in updated handling qualities requirements [Anon, 1987], differs substantially from the original, and only specifies a minimum and much lower value. It introduces the attitude droopback and pitch rate overshoot ratios as major factors in acceptable bandwidth, in accordance with the author's work. This is an example of a criterion that was originally from the frequency domain being enhanced by the addition of time domain metrics.

### 7.5.2 Manual flight path control

In the same manner as for attitude control, §7.5.1, there are both time and frequency domain criteria for control of the flight path, e.g. Hoh [1988a, b] and Gibson [1990, 1991a]. An early proposal by the author to link the two domains was given in Figure 3 of Gibson [1982]. This converted the specified short period frequency and damping limits in Figure 7-2 [from Anon 1969] into the maximum time for the normal acceleration response to reach steady conditions, if well damped, or the minimum time to first pass through the final value with low damping, in a step time response. This was derived empirically from standard plots of second order time responses, where for example the time to reach a steady value is nearly $9/\omega_n$ for a damping ratio of 1.3.
For the minimum $\omega^2_{sp}/(n/\alpha)$ of 0-28 for Cat. A (combat) in Figure 7-2, this time is given approximately as $16-4/\sqrt{(n/\alpha)}$.

For modern fly by wire systems, the response time for minimum damping is of little interest, as the response is inherently highly damped. Figure 7-20 shows the maximum Level 1 times for Category A (combat) tasks. For the heavily over-damped case of $\xi = 1-3$, the times are clearly excessive at low values of $n/\alpha$, i.e. low speeds. A modified maximum limit for $\xi = 1-0$ (for which the time to reach a steady value is about $6/\omega_n$) is indicated. Maximum time limit values represent minimum values of short period frequency at lower speeds. As discussed further below, the latter are important to maintain adequately precise control of the flight path. Ashkenas [1971] noted that satisfactory limits of response time appeared to be about 2 seconds for attitude tasks and 4 seconds for flight path tasks. Given that the likely primary tasks are attitude control at high speeds (large $n_{2g}$) and flight path at low speeds (low $n_{2g}$), this observation matches the modified Figure 7-20 limits rather well.

The author also introduced the flight path delay parameter $t_\gamma$ (§7.3.1, Figures 7-4 and 7-5) in [Gibson, 1982]. Although the flight path is controlled through the short period dynamics of the normal acceleration response, $t_\gamma$ gives a more direct physical expression of the flight path response in the time domain, and it became the author's preferred method of quantifying the short period response for fly by wire aircraft. In any given conventional aircraft, the distance covered to establish a steady rate of change of path angle is roughly constant. This results from the fact that the short period frequency and hence the path delay time are inversely proportional to speed. As the speed decreases, the time taken to cover this distance becomes too long, requiring some minimum in short period frequency or maximum in path time delay to be specified.

The formal specifications of short period frequency and damping limits illustrated in Figure 7-2 can be transformed into a specification for flight path time delay as follows.

$$\text{Since the limiting values of } \frac{\omega^2_{sp}}{(n/\alpha)} = c$$

(7.6)

(where $c$ is a specified constant)

then $\omega_{sp} = \sqrt{c(n/\alpha)}$

(7.7)

By definition (7.3) $t_\gamma = 2\tau_{sp}/\omega_{sp}$

(7.8)

hence $t_\gamma = 2\tau_{sp}/\sqrt{c(n/\alpha)}$

(7.9)

The resulting flight path time delay limits derived from Figure 7-2b are shown in Figure 7-21. The upper boundaries combine the lowest permitted frequency and the maximum damping at corner “A” for Cat. A (combat). The lower boundaries combine the maximum permitted frequency and the lowest damping at corner “B”. Similar boundaries are derived for Cat. C (landing approach) by the same method. The excessively large maximum values of path time delay that result at low speeds were evidently not considered for either Category A or Category C when the original specification was written. However, the extreme combinations were seldom tested in flight. Experiments from which frequency limits might have been established would generally use a nominal damping ratio of around 0-7 to 0-8, at which the path time delay would be approximately 1-5 to 2 seconds.

The consequences are shown graphically by the time and frequency response in Figure 7-22 of point “X” in Figure 7-21. The time response only reaches the nominal asymptote after some 10
seconds, far beyond the pilot’s predictive power, giving an appearance of an acceleration control \( (K/s^2) \). In the frequency response, the 0.85 radians per second natural frequency lies in the middle of the range desired for good path control. The flight path angle phase relative to control inputs is -180 degrees at this point, coupled with a gain slope of -40 dB per decade which represents a \( K/s^2 \) acceleration response in the region of interest. An example from Boothe et al [1974] with excellent path control for the flight refuelling task is also shown for contrast. This has clearly \( K/s \)-like characteristics in both the time and frequency responses in the range needed by the pilot. The initial time lag due to the short period mode negligibly affects the path response at the significant frequencies and is sufficiently small to cause no piloting problem.

The flight path bandwidth, defined similarly to the attitude Bandwidth Criterion, Figure 7-15, is the frequency where either the phase angle is -135° or where the gain is twice that at -180° phase angle. The flight path bandwidth is much lower than the attitude bandwidth, flight path being essentially equal to attitude lagged by the lift parameter \( T_{d1} \) (Eq. 3.7). As the flight path response phase angle is -180° at the short period natural frequency \( \omega_{sp} \) (Eq. 3.8), the flight path bandwidth is lower than \( \omega_{sp} \), unlike the attitude bandwidth which is higher.

Figure 7-23 shows that the bandwidth and time delay parameters are closely equivalent for well damped responses. The minimum frequency domain bandwidths of 0.6 and 0.8 radians per second for normal and precision landing control given by Hoh [1988a, 1988b] agree well with the corresponding independently derived time domain path delay limits of 1.5 and 10 seconds suggested in Gibson [1982 to 1991a]. The considerable region of nominal Level 1 short period behaviour which lies outside these satisfactory delay or bandwidth limits is clear, extending to include the extreme point “X” in Figures 7-20 and 7-21. It can be concluded that for normal damping levels of around 0.7 to 1.0, the minimum short period frequency for landing should be more like 1.5 to 2 radians per second than the 0.85 actually permitted e.g. [Anon, 1987], to achieve a sufficiently responsive control of flight path.

### 7.5.3 Some special cases of manual flight control

#### Carrier landings

The special difficulties of path control in carrier landing approaches was previously mentioned in Chapter 3, §3.2.4, with regard to operation “on the back side of the drag curve”. In this regime a piloted closed loop path instability occurs if the pilot attempts to control flight path with the stick instead of with the throttle, that is the aircraft descends instead of climbing when the stick is moved aft. Control of speed through the AoA remains normal.

Extremely precise path control is required to hit a landing area, perhaps only some 35 metres long and often heaving up and down. It is carried out with a typical approach speed of 1.08 Vs, which gives an exceptionally low stall margin and a very limited capability for manoeuvre. The approach speed is well below the minimum drag speed, described as “on the back side of the drag curve” because the drag increase as the speed decreases. Raising the nose initially reduces the descent path angle, but the speed decreases, the drag and hence the thrust required increases and the aircraft begins to sink more rapidly than before. In the worst “low and slow” case, full power may be needed merely to maintain the speed, and a pitch down becomes the only option left to increase the speed again to reduce the drag before the descent can be checked. This is unfortunate if the ground or sea is very close.

Naval aircraft usually depend on a fast engine thrust response for approach path control, but two criteria, by Smith and Geddes [1979] and Patterson [1967], have been stated for carrier landing
approaches by pitch control alone for if this is feasible. Figure 7-24 shows that the two requirements can just be met with a path delay of 1.5 seconds and a pitch rate of 2 degrees per second. Since the latter typically represents a 0.25 g increment, it is too close to the stall. A smaller pitch rate requires a path delay of 1.0 second or less, as suggested earlier for precise control.

Large aircraft

Very large aircraft tend to have a manoeuvre margin that is significantly larger than the static margin, due to the long tail length and the correspondingly large pitch damping moments. As a result their short period frequencies at approach speeds will not be as low as might be expected from their large pitch inertia. Even so, they are likely to lie at the bottom of frequency limit boundaries such as those in Figures 7-2 and 7-3. Large aircraft are almost expected to be sluggish by default, but although for most landings this causes no problems, poor handling may be most evident when it is least wanted in difficult landing conditions. However, there is an ameliorating effect, arising from the location of the cockpit at a considerable distance ahead of the centre of gravity, giving the pilot a substantial phase advance in the cockpit flight path response.

This is illustrated in Figure 7-25 for a nominal short period frequency of 0.2 Hz with the cockpit located from 3 metres behind to 30 metres ahead of the CG. No elevator lift effects are included, as discussed in §7.2.1. This effect moves the instantaneous centre of rotation on the application of a sharp control input from the centre of gravity to the “centre of percussion” some distance ahead of the CG, the shorter the tail arm the greater this distance. A cockpit located aft of the centre of percussion, as on the Shuttle Orbiter, will initially sink on application of a nose up control input. For the extreme forward cockpits the flight path angle, which can be considered to represent vertical velocity, has an uncomfortable hesitation in the response. A limit on this effect is suggested in Mooij [1984] which is equivalent to specifying that there should be no “flight path angle dropback”.

A limitation of the Bandwidth Criterion is exposed here. For sufficiently forward cockpit locations, the phase lag is always less than 135° except at high frequencies well outside the range of interest to a pilot. A phase angle bandwidth cannot be defined realistically by this criterion. This is not realistic, and results from the over-simplified assumption of equating an artificial “open loop phase margin” with a closed loop crossover frequency.

Ground effect can also make the landing flare easier to perform in large aircraft. The Boeing 747 requires no more than a small nose up pitch angle change for a soft landing, according to Davies, and performs an acceptable landing with no control action at all. The early landings of the Concord prototype were very heavy, because the pilot’s reaction with up elevator to the pitch-down due to ground effect slammed the wheels on to the ground. It was soon found that the ground effect would cushion the landing without pilot action.

7.6 Sensitivity

Sensitivity in aircraft handling terms is a reference to the responsiveness of an aircraft relative to the inputs from the pilot. If the sensitivity is too high, the pilot will be unable to control with precision because the aircraft will over-react to small inputs. If it is too low, the pilot must apply large physical motions or efforts to the stick to initiate a sufficiently rapid response and precision will again be diminished. With satisfactory sensitivity, the aircraft follows demands predictably and positively within a natural-seeming range of pilot effort.
Even with satisfactory response dynamics, excessive sensitivity can lead to pilot induced oscillations if the pilot is unable to operate with a low enough gain. As already noted in Chapter 5 (§5.2.8), it was found by McRuer [1960] that the gain the pilot must select to perform closed loop tasks satisfactorily has a dominant influence on pilot opinion. Figure 5-7 indicated the manner in which opinion worsens as the pilot gain is forced to increase or decrease from an optimum level. This result was based on pitch attitude control tasks, but no definition of appropriate values was adopted in subsequent requirements [e.g. Anon 1969, 1980, 1987].

Contrary to the traditional acceptance of stick force per g as a principal measure of pitch control sensitivity, the author concludes that if it plays any part it is only a minor one, for the reasons discussed below. Attitude-based characteristics seem to define sensitivity best, where the best measure found by the author is the dynamic pitch rate to stick force amplitude ratio around the crossover frequency.

7.6.1 Stick force per g

Beyond an upper limit the control forces to reach maximum "g" are simply too great, even when applied with both hands. Below a lower limit, it may be too easy to over-stress the aircraft. In higher speed flight conditions where the maximum positive g limits of an aircraft can be reached, it is typically specified that the stick force at the limit g should be between 56 lbs and 21 lbs [Anon, 1987]. For an aircraft with an 8g limit the corresponding stick force per g is between 8 and 3 lbs/g. At lower speeds this is permitted to increase, though the maximum stick force does not necessarily increase because the manoeuvre capability decreases with reducing airspeed until eventually the aircraft reaches the stall speed in level flight.

In aircraft without powered controls, stick force per g is nominally invariant with airspeed because the elevator angle per g varies inversely and the elevator hinge moment varies directly with airspeed squared (assuming a rigid aircraft and no Mach effects). In the landing approach at around 1.3 times the stall speed, an aircraft with a nominal 3 lbs per g would notionally require the application of just over 1.5 lbs stick force to reach the stall, negligible compared with the minimum 21 lb force specified to protect against breaking the aircraft at higher speeds. However, the stick deflection per g is much greater than at high speed, and there is normally also a warning of impending stall. These factors may be why pilots have never objected.

With the introduction of powered controls and artificial feel, it became quite usual for the stick force per g to increase with decreasing airspeed. A typical range of approach speed stick forces common to a number of combat aircraft types with this trend was shown in Figure 6-7 from the author's proposal [Gibson, 1978] for such an increase to be formally adopted. These forces ranged from about 8 up to about 20 lbs/g. Such an increase results for example from use of the modified q-feel characteristics as indicated in Chapter 6 and Figure 6-6, where it can be generated either by the use of an additional reversionary mechanical feel spring or by appropriate scheduling of the hydraulic q-feel pressure with dynamic pressure. A similar characteristic is provided by the simple spring and bobweight feel system, due to the constant spring force element which is proportional to the elevator angle per g. These wide variations from the traditional norm have been found to be entirely acceptable to pilots. However, even with the highest force gradients mentioned above, the small manoeuvre levels normally associated with the approach and landing flare require the application of no more than about 4 lbs or so of stick force.

While high stick forces were used in the past to discourage over-stressing the aircraft, the manoeuvre limiting that can be employed in a fly by wire system makes them unnecessary. Low
forces tailored for comfortable single-handed application rather than for protection are then possible. In the author's experience, when pilots of combat aircraft are relieved of the responsibility for structural protection, they consider that even the lower value of 21 lbs previously required for protection of the g limit is heavier than desirable for sustained manoeuvres in the region of maximum g because it is an unnecessary additional physical burden. With the typical constant stick feel gradient system of aircraft with manoeuvre demand control, the stick deflections required at low speeds do not increase as markedly as in conventional aircraft, but an increase in stick force per g compensates for this. Again, this further departure from tradition has been found to be acceptable to pilots.

Evidence has also been available for some 40 years in the variable stability research results of the 1950s era. The author drew attention in §7.4 and Figure 7-9 to the finding that despite a constant stick force per g the pilots perceived the handling to range from dangerously light and responsive to very sluggish and heavy as the short period dynamics were varied. The quite remarkable implications of this were not followed up anywhere at the time or afterwards. Already noted in Chapter 2, the following observations from the leading source of experimental handling qualities data, Calspan, are supported by the author's own observations and conclusions:

"Pilot comments indicated that the primary concern of the pilot was his ability to quickly change pitch attitude in acquiring a target and precisely controlling pitch during tracking maneuvers. Normal acceleration control was certainly of concern to the pilot, but this response was more in the sense of a measure of maneuver magnitude and as the maneuver limiting factor." [Boothe et al, 1974]

7.6.2 Pitch attitude-related sensitivity measures

Attitude

The author's study of the LAHOS landing approach research [Smith, 1978] led to the proposed pitch sensitivity criterion shown in Figure 7-26 [Gibson, 1982]. The pilots chose the stick command gain that they considered most suitable for the landing task, with response dynamics that ranged from excellent to dangerous. The stick feel force gradient remained constant. The resulting stick forces ranged from 6 to 60 lbs/g for all cases and from 14 to 38 lbs/g for Level 1 cases, suggesting that its absolute value could not be of direct importance to the pilot.

The author found that sensitivity was well correlated by the attitude gain (in degrees per lb) at the bandwidth frequency defined at -120° phase angle (§7.5.1). This is close to the piloted crossover frequency, at which the aircraft gain is significant for closed loop control (Chapter 5, §5.2.5). The chosen attitude gains varied considerably but in a consistent way with bandwidth, and the variation at any particular bandwidth was small. An exception was a set of poorly damped cases in which the attitude gain was higher, but here it is likely that this reflected the ability of pilots to tolerate the large response resonance, close to the bandwidth frequency.

The criterion was successfully used in the design of the EAP control laws, which gave excellent handling in the landing approach. The format was taken up by Hoh et al [1987], adapted to the more widely used bandwidth defined at -135° phase angle. Further discussion is given in Hoh [1988a].

Pitch acceleration

CAP, the Control Anticipation Parameter, is often cited as an attitude sensitivity parameter for time responses, expressed in its form of the ratio of initial pitch acceleration to steady state
normal acceleration. However, though the upper ranges of CAP up to the maximum allowable 3-6 rad/sec^2/g (Figure 7-2b) are associated with excellent manoeuvrability due to the high short period frequency, they give poor attitude precision due to oversensitivity as discussed in §7.4.1.

After experience with the FBW Jaguar, which was certainly crisp and which was found by two pilots to be somewhat over-responsive, the author’s internal handling qualities design guides for the EAP [Gibson, 1985a, 1987a] set an upper limit on CAP of 0-44 for up-and-away flight, and 1-22 for the landing approach initially but reduced to 0-7 subsequently. Pitch attitude tracking simulation experiments conducted in support of these guides at the time [Gibson, 1985b, 1987b] obtained excellent tracking performance with CAP of only 0-14 to 0-19. This contrasts with the formal minimum Level 1 CAP of 0-28 specified in Figure 7-2b for combat tasks. In flight the EAP exhibited exceptional qualities exemplified by these pilot comments extracted from a flight report on combat assessment:

*Immediate, accurate and easily controllable response; transitioning from hard or maximum rate manoeuvres to fine tracking was exceptionally easy with excellent control response; settling times into a fine tracking situation were negligible.*

While the EAP utilised a variable dynamic stick command pre-filter that provided precision attitude tracking for small inputs but a quickened response for large inputs, a technique that will be discussed in Chapter 10, the CAP for large inputs was seldom more than doubled. The pilot comments above show that CAP values at the lower extremes of the formally required range were entirely sufficient for good handling in a highly manoeuvrable aircraft.

In the flight control laws of the Lockheed Martin F-22 fighter, currently under development, the design CAP values chosen were 0-35 rad/sec^2/g for high speed flight and 1-0 for the landing approach [Harris and Black, 1996]. These values are remarkably similar to those adopted in the EAP design, although they were independently arrived at. In both cases low CAP values were found to be a consequence of satisfying similar non-parametric attitude response metrics.

Gautrey [1998] describes simulation experiments on the handling qualities of fly by wire transport aircraft with a variety of manoeuvre demand control laws, primarily in the low speed landing approach phase. All were designed to achieve a CAP of 0-6 rad/sec^2/g, on the basis of Gautrey’s research into the requirements for precision landings of large aircraft. While this CAP is lower than in the EAP and F-22 landing approach examples, it represents a flight path response that is considerable faster than is found in conventional large aircraft. The simulation pilots found that the landing control qualities were very satisfactory. Mooij [1984] proposed a maximum value for large and transport type aircraft of 0-45 rad/sec^2/g.

The above examples are a reminder that CAP was derived as a flight path control metric, not an attitude one. In the time domain, CAP defines only the initial angle of attack acceleration, but does not predict the subsequent attitude and rate dynamics that dominate the pilot’s behaviour and opinion as discussed above. In the frequency domain, CAP is unrelated to the attitude response characteristics in the crossover frequency range near the bandwidth frequency and cannot predict the closed loop tracking sensitivity. CAP can only reflect the flight path response in classical dynamics, and in a fly by wire aircraft its value can be altered substantially by a high frequency lead pre-filter while leaving the flight path response only slightly modified.

If CAP does have any relevance as an attitude sensitivity metric, it is not to be found in the range of values shown in Figure 7-2b that are given formally in Anon [1987]. In fact, CAP does not involve the element of stick force gain necessary to define sensitivity. The relevant measure
of pitch acceleration is its amplitude per unit stick force. This depends upon the short period frequency but also independently upon the pitch feel force gradient. In Gibson [1990], a result of analysis of the LAHOS experimental data was presented showing a variation of the pitch acceleration frequency response to stick force gain with the attitude bandwidth frequency that was as consistent as the attitude gain results in Figure 7-26. In this case the acceleration gain increased with bandwidth. However, no use was made of this result in the author's design criteria.

In the time domain, it is impossible to apply the pitch control in a pure step and in practice the acceleration will be substantially less than that implied by the CAP value. In pitch tracking tasks where typical attitude corrections may only be a few milliradians, or less than one quarter of a degree, the normal acceleration variations are so small that the associated pitch acceleration is most unlikely to be detectable by the pilot. This is especially so with the very low values of CAP found optimum as discussed above. A step input response upper limit of 1.0 deg/sec²/Newton, or about 4.4 deg/sec²/lb, was stipulated for the landing approach in Gibson [1985b, 1987b]. This was empirically based on experience with the FBW Jaguar, but it was never a deciding factor in the design of the EAP control laws. Gauvreau [1998] states that for transport aircraft the optimum pitch acceleration is between 0.4 and 0.7 deg/sec²/lb. This is about 6 to 11 times less than the author's proposal for combat aircraft, but this mostly reflects the much heavier stick force per g employed on transport aircraft, coupled with their usually somewhat lower CAP.

The author concludes that pitch acceleration does not appear to provide a satisfactory basis for a pitch control sensitivity design measure.

**Pitch rate**

The "Pitch Rate Sensitivity Criterion" by Sturmer [1986] proposed upper and lower gain boundaries on pitch rate frequency response Nichols plots for the landing approach, based on the LAHOS results of Smith [1978]. Sturmer's criterion was amended slightly and extended to combat flight conditions by Wilson et al [1990], shown in Figure 7-27. Over a range of frequencies from zero to that at which the pitch rate phase angle is close to -45°, these gain limits are essentially constant. The frequency at this phase angle is the pitch attitude bandwidth (phase angle -135°), and the pitch rate gain sensitivity limits at this point are therefore independent of the bandwidth. The author simplified the design process by using only the specific limit values at the bandwidth frequency [Gibson, 1995a].

Figure 7-28 shows the generic relationships of the gain at the bandwidth frequency of the response to stick force of pitch attitude and pitch acceleration, given the constant pitch rate gain limits of Figure 7-27. The pitch attitude gain varies inversely with bandwidth frequency, while the acceleration gain varies in proportion to it. The plot for attitude resembles very closely the empirical attitude result in Figure 7-26, and the plot for acceleration also replicates quite well the shape of the result given in Gibson [1990]. This should be expected as they are all derived from the same LAHOS data.

Smith and Geddes [1979] noted that attitude is perceived visually as the integration of a controlled rate output. It seems logical to accept that the well known pilot preference for rate-like control for many purposes, expressed by the nearly universal \( K/s \) optimum for closed loop control, makes sense of a constant range of satisfactory rate gains in the crossover region - i.e. close to the pitch attitude bandwidth frequency. The author's design criterion obtained from this is to ensure that the rate gain at the attitude bandwidth frequency lies within the proposed limits.
The author extended this sensitivity criterion by the empirical observation that, at least for well-damped highly augmented fly-by-wire response types, there is a reasonably good correspondence between the pitch rate gain at the bandwidth frequency and the gain of the initial peak of the pitch rate time response. The limits of the former in Figure 7-27 translate into values of 1.1 deg/sec/lb and 0.56 deg/sec/lb. Suggested empirical limits for the time response peak are 1.1 deg/sec/lb and 0.6 deg/sec/lb, though an exact match should not be expected and the values are indicative rather than to be precisely applied.

Figure 7-29 shows how the pitch rate sensitivity gain can readily be derived directly on the pitch attitude plot from the fact that, in simple harmonic motion, the peak rate equals the peak amplitude times frequency. Hence the addition of the frequency in radians per second (in dB) to the attitude gain point gives the pitch rate gain. The response examples here are from Figure 7-18, taken from the author's simulation [Gibson 1985b]. The tabulated peak pitch rate time response gains are very similar in Cases B and C to their frequency response gains. The match for Case A is less exact but it is still well above the suggested upper limit. In fact, all the cases in Gibson [1985b, 1987b] with the best tracking performance satisfy the pitch rate sensitivity criteria, a fact discovered only in retrospect. On the other hand, the correct sensitivity does not guarantee excellent tracking.

The examples in Figure 7-30, from Boothe et al [1974], further illustrate the usefulness of the time response pitch rate peak gain. Cases 2D and 8A have a large dropout ratio, contrary to precise attitude tracking requirements, yet 2D with almost 50% larger dropout than Case 8A was rated considerably better though it was somewhat oversensitive. The reason appears to be the pitch rate sensitivity, which almost satisfies the proposed maximum in Case 2D but greatly exceeds it in Case 8A. Case 8D was rated as excellent. It represents the $K/s$ ideal almost perfectly with zero dropback, and its pitch rate peak sensitivity is in the middle of the desired range. From this open loop time response it can be deduced that the closed loop crossover characteristics of 8D are ideal, as was found in actual flight.

The author has had good experience in practical application of the pitch rate sensitivity criteria to the Eurofighter design. Before-flight simulation of the preliminary control laws for the early flights, a very basic set designed for reversionary purposes rather than for normal Service operation, indicated some pitch over-sensitivity. Modification of the control law to satisfy the frequency response sensitivity criterion led to excellent pilot opinions. The following quotations from the report of an RAF test pilot on his first flight in the early stages of flight development, still using a reversionary control law set, are presumably unbiased by company loyalty:

Took no getting used to; much better than any other jet I've flown; approach path deadbeat; my gentlest landing ever; landing remarkably easy; a joy from start to finish.

The above considerations indicate that sensitivity expressed through stick force per g limitations is related primarily to traditional structural protection and is not truly a general measure of handling quality sensitivity. The optimum adjustment of stick force characteristics seems ideally to be related to attitude-based measures, particularly the pitch rate gain at the bandwidth/crossover frequency.

No such criteria of any sort are available for transport or civil aircraft in general, however. It is likely that the simpler tasks and typically heavier stick forces make them unnecessary.
7.7 Summary

Mathematical processes are heavily involved in the design of aircraft, though they will often be in the form of computer numerical calculations due to the extremely complex models that can be necessary. If exact mathematical models of human pilot behaviour were available, it would be possible to use entirely mathematical means to obtain good handling qualities. Since there are no such pilot models, reliance is placed on pilot opinion and comments, which are given in time and frequency response terms and metrics.

This Chapter has presented the author’s core discoveries of new metrics for pitch handling qualities. Derived from conventional aircraft dynamics, they explain the confusing contradictions in pilot opinion in many past variable stability aircraft research experiments. They offer a coherent range of parameters that constrain the satisfactory area of handling on the \( \omega_{pr} - \zeta_{pr} \) plane, for long the primary handling parameters for conventional aircraft, to an area that varies with the aircraft parameter \( T_{92} \). The parameters are the attitude dropback ratio, time to the first pitch rate peak, pitch rate overshoot ratio with associated sensitivity gain, and also a flight path angle delay independent of the attitude characteristics. The "shape" of both time and frequency domain responses are found to provide important aids to the recognition and quantification of satisfactory handling dynamics, including the desirable crossover model characteristics.

Because the metrics can be identified either graphically from output response plots or by computer processing, they can be applied equally readily to the design of high order response types as will be discussed in Chapter 10. Requiring no mathematics for their application and providing a direct visual representation of the response characteristics, they are ideally suited to fostering the all-important dialogue between control law design engineers and pilots, the lack of which has caused many mistakes in the past.
Figure 7-1  Piloted closed loop control of flight path with an attitude control inner loop, taken from Myers et al, 1987

1. The attitude loop is continuously monitored and closed to correct path errors.
2. The flight path loop is broken when attitude is the dominant task
Figure 7-2 Short period requirements for Category A (combat) tasks

The development of new criteria from old data

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Figure 7-3 Experimental "Best tested areas" from several in-flight and ground simulations
Figure 7-4 Generic short period pitch time responses
(2 degree of freedom; no incremental elevator load effects)
Steady pitch rate \( \dot{q}_{ss} = \dot{\gamma} \)

\[ t_\gamma = \frac{2\zeta_{sp}}{\omega_{sp}} \]

\[ DB/q_{ss} = (T_{\theta_2} - 2\zeta_{sp}/\omega_{sp}) \]

Figure 7-5 Detailed attitude/flight path relationships

Figure 7-6 Flight demonstration of attitude/flight path relationship
Figure 7-7 Early "Gibson frequency response criteria" templates [1982].
(Boundaries 1 to 5 enclose optimum response shapes)
Effect on pitch rate gain of increasing speed at constant altitude

Effect on $1/T_{\theta 2}$ with increasing altitude at constant true speed, due to increasing angle of attack required

For constant stick force per g and manoeuvre margin:

\[ \omega_{\theta} \propto V \]
\[ 1/T_{\theta 2} \propto V \]
\[ \omega \theta T_{\theta 2} = \text{constant} \]
\[ q_{\theta} \propto 1/V \]

(a) Bode plot gain asymptotes

Figure 7-8 Generic trends in pitch attitude frequency responses with flight condition.

No unique attitude frequency response "shape" can be found to provide satisfactory handling at all flight conditions over all the frequency range. However, the crossover and K/s-like characteristics are visible.
Figure 7-9 Enhancement of thumbprint plot format by additional time response metrics.
Configuration with constant $T_{a2} = 1.0$ from [Newell and Campbell, 1954] with constant stick forces of 66 lb/g.
Figure 10  Short period criterion proposal, enhanced by Gibson attitude and path metrics
Based on configuration with constant $L_a = 0.5$ from [Shomber and Gertson, 1965]
1. Lines of pitch rate overshoot, zero dropback and linear PIO are valid for all $L_a$.
2. Lines of dropback ratio, path delay and peak pitch rate time valid only for $L_a = 0.5$.

Figure 7-11 Specification short period boundary proposals from [Hoh et al, 1982]
The flight experiment results, linear PIO and dropback boundaries have been added by the author. These suggest that:
1. The proposed Cat. C (landing) lower boundary should be raised
2. The proposed Cat. A (combat) lower boundary should be lowered
3. The corners of these lower boundaries fail to take account of the effects of the linear PIO boundary and the zero dropback line.
Figure 7-12 The "Gibson" attitude and flight path short period parameters

Note on linear PIO line: Normal aerodynamic configurations are confined to the area above this boundary line. The area below the boundary is possible only by control feedback methods (Chapter 8).
Figure 7-13 Carrier landing handling qualities [Eney 1969], showing how the boundaries of good handling in past flight experiments can be identified in terms of the Fig.7-4 metrics.

Figure 7-14 Generic boundary limits for the short period thumbprints, describing in general terms the time response characteristics that set bounds on the limits of good handling.
Figure 7-15 Definitions of attitude bandwidth and phase delay

Gibson [1982]: Bandwidth = $\omega_{l(120)}$
Hoh et al [1982]: Bandwidth is lesser of $\omega_{BW\text{gain}}$ and $\omega_{BW\text{phase}}$

Figure 7-16 Discrepancies in Bandwidth Criterion results
Figure 7-17 Typical effects of high CAP on pitch tracking, from [Quinn et al, 1984]

![Diagram showing test points and conditions selected at upper Level 1 frequency limit.]

Figure 7-18 Effects of bandwidth and dropback on tracking [Gibson, 1985b]

1. Cases B and C are modified from Case A to achieve zero dropback.
2. Case B has reduced short period frequency and command path pre-filter.
3. Case C has only a command path pre-filter (see Chapter 10)
Figure 7-19 Generic trends in attitude bandwidth variations of conventional aircraft. The range of values is far greater than the proposed limits of the Bandwidth Criterion, Figure 7-16.

Figure 7-20 Alternative specification of minimum "bandwidth" for Cat. A (combat) based on proposal in Gibson [1982].

Limits on response time in Figure 7-4 (time to steady state):
1. based on Figure 7-2 minimum Level 1 short period frequency
2. based on more appropriate limits for satisfactory handling
Figure 7-21 Conversion of short period requirements to path delay

Figure 7-22 Flight path control: excessively low bandwidth results in extremely sluggish flight path response

1. Case 2D is a flight refuelling condition with excellent path control.
2. Point "X" is the permitted Level 1 minimum frequency/maximum damping for Cat. C (landing approach).
Figure 7-23  Equivalence of flight path bandwidth $\omega_{\Delta y}$ and flight path time delay $t_y$

Figure 7-24  Equivalence of 1.5 secs. flight path time delay to other criteria.
   by use of pitch control alone
2. Patterson [1967] requires climb through an approach path parallel to and 50 feet
   higher than the original path by use of pitch control in not more than 5 secs.
Figure 7-25 Aircraft length effects providing an apparent phase advance in flight path response at the cockpit.
Figure 7-26 Proposed pitch attitude sensitivity criterion for landing approach [Gibson, 1982], derived from LAHOS flight experiment [Smith, 1978].

Figure 7-27 Proposed pitch rate sensitivity criterion [Gibson, 1995a]
Figure 7-28 Generic correlation of sensitivity parameters

Figure 7-29 Identification of pitch rate sensitivity gain directly from attitude plots, with examples from Gibson [1985b] shown in Figure 7-18.
Pilot comments:

2D: One minor objection, a tendency to be a little bit oversensitive, during tracking.

8A: Stick forces were quite satisfactory for manoeuvring, probably too light for tracking, bobbling the nose almost continuously. It needs extensive improvement.

8D: Liked the lack of PIO during tracking. Predictable in tracking. Nose moved at the rate looked for; it felt good.

Figure 7-30 Pitch time response sensitivity based on peak pitch rate gain, with examples from [Boothe et al, 1974]. Pilot comments are consistent with proposed transient pitch rate sensitivity limits between 1.1 and 0.6 deg/sec/lb (this thesis).
Chapter 8
Linear, low and high order pilot induced attitude oscillations

8.1 Introduction

Early film of the Wright Flyer shows continuous overcontrolling in pitch. Jex and Culick [1985] estimated that its highly unstable airframe, with a time to double amplitude of 0-6 seconds, was controllable if the pilot closed a tight attitude loop, with a neutrally damped attitude oscillation. This was a pilot induced oscillation or PIO, which is defined in Anon [1987] as a sustained or uncontrollable oscillation resulting from efforts of the pilot to control the aircraft. PIO is caused by an unstable or neutrally damped closed loop coupling between the aircraft dynamics and the pilot. The Flyer was unusual in that, at the pilot gain value for stable closed loop control, either increased or reduced pilot gain would have resulted in loss of control. This is the only example known to the author of a PIO condition that could be said to be necessary for flight. However, it was also a special case that was open-loop unstable. All other cases have been open-loop stable, but PIO remained an unmitigated nuisance at best and a catastrophe at worst.

Anderson [1993] states that in 1908, 80% of the licensed Flyer pilots were killed in accidents. No conventional aircraft with unpowered controls has subsequently ever experienced such a severe problem, but PIO continued to occur as aircraft design advanced. The study of PIO events and possibilities by Ashkenas and McRuer [1964] listed a number of causes related to mechanical and aerodynamic deficiencies found up to that time:

- Hysteresis in the stick versus elevator position, caused by control linkage friction with inadequate centring forces, resulted in low frequency speed and climb oscillations typical of control linkage friction with inadequate centring forces.
- Actuator valve friction plus compliant control linkage resulted in large oscillations at the short period frequency. The valve is held fixed at random positions, producing actuator motion with the input point locked to it until sufficient resistance is generated in the linkage to free the valve.
- Localised unstable slopes within the overall pitching moment curve led to moderate-period oscillations of varying amplitudes (depending on the extent and nature of the kink) during manoeuvres near the angle of attack on which the kink was centred.
- Transonic separation over the rudder caused control reversal for small deflections, leading to limit cycles when the rudder was used to damp yaw oscillations. (This would similarly cause limit cycles in a stability augmentation system, but would not constitute a PIO.)

Such problems were eliminated or greatly alleviated on the advent of fly by wire flight control systems. Electrical signalling removed the mechanical connection between the pilot and the power control actuation, leaving only the pilot’s stick and simple feel spring as a potential source of frictional and other deficiencies. The worsening handling qualities associated with the jet era configurations could be suppressed to a large extent by powerful full authority stability augmentation systems, and it became possible to control configurations in which the overall aerodynamic pitching moment is grossly unstable

13 The most extreme example of the latter is the Grumman X-29, which at the worst flight case had an unsaugmented time to double amplitude of about 0-12 seconds. Though successfully stabilised, the margins are too small for practical use in a service environment. Practical limits are exemplified by the FBW Jaguar, EAP and Eurofighter, with double amplitude times in the range of 0-25 to 0-18 seconds.
Table 8-1 shows the usual pilot rating scale for PIO, or PIOR [DiFranco, 1967]. It appears to be somewhat difficult to interpret consistently. According to Duda and Duus [1997] it is highly non-linear, and the main distinction is between PIOR 3 and 4. In both of these, oscillations develop when the pilot closes the loop or initiates abrupt manoeuvres. With PIOR 3 these compromise the performance but can be stopped at some expense to task performance by reducing pilot gain. With PIOR 4 they can be stopped only by freezing the stick and abandoning the task.

8.1.1 Low and high order system PIO

A PIO can range from small undamped oscillations to a grossly unstable divergence, and can occur even where the normal aircraft response is stable and heavily damped. They have occurred from very low phugoid frequencies up to high frequency “ratchet” at several Herz. They have occurred over an extremely wide flight envelope, mostly in pitch control but sometimes in roll control, though they have come to be most notoriously associated with landing and take off. Most PIOs begin only when the pilot attempts tight closed loop control, and will stop instantly if the pilot abandons the task. Unfortunately, in many cases the pilot feels impelled not to abandon the task because of the circumstances, typically when in close proximity to the ground.

Typically there would be no evidence of severe control difficulties for a long period of time, until a sudden and sometimes catastrophic PIO occurred. It can also occur very early in the aircraft’s development, as on the “Flight 0” of the General Dynamics YF-16 prototype where a high speed taxi run had to be converted into a take off [Hoh et al, 1982], and the PIO on the 6th flight of the prototype SAAB Gripen that was widely shown on television.

In aircraft prior to the use of fly by wire control systems, the basic response dynamics were essentially low order, that is of classical form, despite the addition of actuator dynamics and a certain amount of stability augmentation. PIO generally attracted PIOR ratings no worse than 3, and pilots could retain control by reducing gain without suffering more than unsatisfactory task performance. These could generally be understood by the use of familiar linear piloted closed loop analysis. An exception was the severe PIO type that occasionally resulted from non-linear control saturation, discussed in Chapter 9.

In recent years, PIO has become a serious problem in many fly by wire aircraft, most of which had digital flight control systems. Hodgkinson [1982] gave a pitch rate transfer function example from the digital McDonnell Douglas F/A-18 that was defined by 89 damping, frequency or time constant values instead of the three required for the classical short period approximation. Despite the fact that this excluded direct digital sampling effects, PIO was often attributed to "digital delays", to the extent that some held the view that digital flight control could never be satisfactory. However, all had very high order control law structures, and similar problems were found with analogue systems, e.g. the Panavia Tornado [Gibson 1978]. It was widely felt in the early 1980s that such systems could never be understood properly or result in satisfactory flight control. This was the driver for the LOES concept (Low Order Equivalent System, Chapter 7).

Most typically, these PIOs were characterised by low frequency and large amplitude, with PIOR of 4 or greater. Such PIO always resulted in at least the abandonment of a task or at worst a serious accident. The high order features of the control systems could be characterised by either excessive phase lag or excessive command gain, sometimes both together. The former would cause closed loop pilot-aircraft instability, rather than the simpler low closed loop damping in low order systems, and high command gain often converted the initial instability into a catastrophic event by the dramatic change in aircraft dynamics after rate saturation of the actuators.
Such events have a major impact on the development of the type involved. The F/A-18 in particular required a long period of flight control system redesigns to overcome serious handling deficiencies (including many PIO conditions). This was so protracted that the aircraft was at one time on the brink of cancellation. The Gripen programme was seriously delayed, and suffered a second PIO accident, before the problems were satisfactorily solved.

8.1.2 The "invisible" problem of high order PIO

As the basic causes of high order PIO are not mysterious, it might be wondered why the problem was not solved very quickly. The reason is that until recently, little effort was expended elsewhere to investigate the problem as a whole and to find design criteria to prevent it. The author’s paper with a description of a Tornado PIO and its solution [Gibson, 1978] was greeted with some amazement. Chalk [1979] described it as "a rare example of a type that should be encouraged. (It) is rare not because problems occurred but because the designer was willing to report on the occurrence. In fact, similar problems have been experienced in the YF-17, YF-16, F/A-18 and Space Shuttle". While the latter were dealt with locally, there was no attempt to develop a widely based effort to solve the general problem. The Tornado paper seems to have begun a slow trend to more openness, but this was sometimes grudging. The flight record of the YF-22 PIO in 1992 was altered before publication in Aviation Week [Dornheim 1992], thus concealing the rate limiting of the pitch controls even though this was mentioned in the text. The correct record is given by Harris and Black [1996].

Unwillingness to publicise PIO events had several root causes. Highly skilled pilots may refuse to admit that that they had lost control of an aircraft, even temporarily, and can simply deny that a PIO occurred. There is no evidence unless adequate flight records exist. Management may be tempted to blame the pilot because the serious PIO events are rare, and the redesign and testing required to eliminate the problem would involve considerable expense and time. In fact, no blame attaches to the pilots involved, and no level of experience makes a pilot immune from the possibility if it exists. Even research pilots testing a number of PIO-prone landing configurations have been unable to prevent its initiation [Smith, 1978]. Nothing can prepare a pilot for the sometimes traumatic or terrifying event, described by one pilot as "like an out of body experience" according to McRuer in discussions at the 1994 PIO Workshop [Anon, 1995].

The considerable public notoriety from accidents to the Lockheed YF-22 [Dornheim] and the SAAB Gripen, the McDonnell Douglas MD-11 and Boeing 777 PIOs [Anon, 1997], the PIOs and structural-pilot couplings of the McDonnell Douglas C-17 and the Bell Textron V-22 Osprey [Norton, 1995] spurred the authorities into action. In the past few years a major official effort has been commenced, sponsored by NASA and the USAF, with a number of contracts placed on the leading US research establishments and institutions. This has produced a series of in-depth studies into the whole complex array of causes and circumstances, to date notably Klyde et al [1995], McRuer et al [1995, 1996], Mitchell et al [1994a, 1995, 1996]. Attempts were made to change the description of PIO to "aircraft-pilot coupling" or APC to remove the perceived stigma. This new title has not become widely used and PIO remains the usual phrase.

Most notably, a "Committee on the Effects of Aircraft-Pilot Coupling" was set up by the U.S. National Research Council under the chairmanship of D. T. McRuer, meeting in 1995 and 1996, to evaluate the current state of knowledge about adverse APC and processes that may be used to eliminate it from military and commercial aircraft. It has reported in Anon [1997], with seven pages of findings and recommendations too numerous to summarise here. A significant finding was that most of the severe high order PIO cases had involved actuator rate saturation. Most usefully, it draws together all the known PIO criteria and those currently under development,
and recommends that in the present state of knowledge, as many as possible should be applied to any new design to increase confidence in PIO prevention.

A recent general workshop discussion on PIO is reported in McKay [1994] and Anon [1995]. Very recently, the AIAA held a one-day tutorial short course on PIO [Anon, 1998]. The range of factors contributory to PIO is so great that the following serves only as an introduction.

8.1.3 A path to solution and elimination of high order PIO

Pilot induced oscillations can be successfully analysed if only one can find their essential elements. Unfortunately this has usually been after the event. The author was forcibly introduced to the problem in 1976 with an accident to an early Panavia Tornado prototype [Gibson, 1978]. There was a total absence of literature on the subject, but a solution had to be found quickly. This required an understanding of what was unacceptable in the aircraft dynamics and discovery of what was acceptable. Determination of essential differences from conventional aircraft enabled a successful change to the control laws to be proposed by the author. The elements of the solution were not entirely complete at the time, but the author's subsequent research proved that they were based on sound reasoning and they have not fundamentally altered to this day.

The causes of high order PIO have been many and varied, but they all reduce to the problem of insufficient stability margins in what is often a very simple closed loop. The author has found that the essential aircraft dynamics that lead to it are similar in all available examples whether linear or non-linear, and therefore they can be eliminated by design. The subject has been discussed in all the Gibson references, reporting the author's evolving methodology. The main finding in these references is that the likelihood of high order PIO can be identified from the aircraft response dynamics alone, and that the following characteristics are exhibited:

a) The PIO always occurs at or slightly higher than the frequency where the open loop phase lag reaches 180°, the "PIO frequency".
b) The open loop phase lag reaches 180° at low frequency, typically well under 1-0 Hz.
c) The phase lag increases rapidly with frequency in this region.
d) The open loop response gain at the PIO frequency is large, allowing large amplitude oscillations to be forced within the limits of the control stick travel.
e) Elimination by design of points b, c and d shapes the response dynamics to guarantee freedom from PIO.

8.2 The fundamental low order closed loop tracking problem

The pilot-aircraft closed loop system was discussed in Chapter 5 and illustrated in Figure 5-1. A highly skilled and trained pilot can develop substantial lag and lead equalisation, in effect modifying the aircraft open loop attitude response to a $K/s$-like form to minimise tracking errors. The advantages of the Nichols plot for handling qualities analysis and design were described in §5.2.7. Its format is eminently suitable for examination of the problem because it shows in a powerful visual manner the closed loop characteristics resulting from a given combination of pilot and airframe open loop responses. Bode plots, widely used by others for handling studies, do not have this facility in use and the author uses them only for limited purposes.

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Since only fully augmented aircraft are discussed in the remainder of this thesis, all mention of "aircraft dynamics" refers to the aircraft plus stabilisation system closed loop. Reference to "aircraft open loop dynamics" means those of the augmented closed loop system alone without closure of the pilot-aircraft loop.
The classic attitude response, either in pitch or roll, does not have phase lags greater than 180° (phase angles more negative than -180°) at high frequency, as sketched in Figure 7-15. In the case of roll attitude, this is because of the fundamental structure of the (simplified) dynamics, Eq. 3.13. The (simplified) pitch attitude dynamics, Eq. 3.3, permit greater lag at intermediate frequencies in principle, but not in practice as discussed in §8.3 below. Tracking attitude angle without pilot time delay would therefore give infinite gain margin. In fact, the presence of pilot time delay while tracking random-seeming target motions, the additional phase lag due to any pilot lag equalisation, and the usual presence of actuation dynamics ensure a total phase lag greater than 180° at moderately low frequencies. The arbitrary aircraft response in Chapter 5, Figure 5-4, is typical of many conventional aircraft. Such dynamics did not lead to a general problem of loss of control since the stability margins usually remained adequate.

The "PIO syndrome" and tracking PIO
The arbitrary attitude response example in Figure 5-4 illustrated a typical closed loop problem which is not soluble by pilot gain adjustment alone. If a low enough pilot gain is used to avoid the resonant area, the basic tracking performance is poor, with droop in the closed loop gain at the lower frequencies. If sufficient pilot gain is used to prevent the latter, oscillations ranging from mild to severe can result. These will occur at the frequency where the aircraft plus pilot response plot is tangential to a Nichols closed loop response amplitude curve. Figure 5-5 sketched a Bode plot of the closed loop response. Such a response can be very sensitive to pilot gain, with small variations causing large changes in closed loop amplitude ratio and thus the oscillation amplitude. With unsatisfactory aircraft dynamics, the pilot had also to apply gain and phase equalisation as discussed in Chapter 5.

Figure 8-1 shows a figure taken from Gibson [1978], illustrating the development of the prototype Tornado pitch tracking qualities. During this time the feedback stabilisation gain schedules and filters remained unchanged, and so there was no change in the stick-free dynamics. The only changes made were to the stick command pre-filter, thus modifying the controlled response characteristics. The author's assessment was by a modified Neal and Smith criterion [1970]. This criterion judges the handling by the pilot lead equalisation necessary to achieve a specified closed loop bandwidth of 3 radians/second, defined in the figure, with less than 3 dB droop and minimal closed loop resonance and with a given pilot time delay (see also §8.2.1).

The author assessed the closed loop resonance resulting from a pilot model with only a time delay and a gain selected to achieve zero droop at 0-25 Hz. Plot A is with an early version of the pre-filter, and plot B is with an intermediate version. Though overall control of the aircraft was acceptable, precision attitude tracking was unacceptable due to overcontrolling with A, while B was controllable but was poorly damped, leading to pilot complaints of "tracking PIO" though with no loss of control or safety implications. Plot C shows the final version, which used the first pre-filter that had been specifically designed to the author's new handling methodology, and which was completely successful with excellent tracking. The aim of this pre-filter was to remove the need for pilot equalisation by means of optimising the open loop aircraft response through the command pre-filter, an aim not explicitly addressed in the earlier versions.

Figure 8-2 shows Nichols plots of the Tornado pitch attitude frequency response at the flight condition of Figure 8-1. Plot (a) is the aircraft dynamics without a pre-filter, and curve (b) includes a pilot time delay of 0-15 seconds. The assumed pilot gain of 5-0 lbs/degree is clearly not possible, as there would be a severely divergent oscillation due to the negative gain margin. This could be converted to a stable but highly resonant oscillation if the pilot drops his time delay, as suggested in Ashkenas et al [1964], but in practice the pilot gain would have to be re-
duced by some 60% or the tracking attempt abandoned altogether. Plots (c) and (d) show the effect of the lag/lead pre-filter C, which was \((1 + 0.2s)/(1 + 1.0s)\). This substitute for pilot equalisation with a 5 to 1 attenuation ratio gave excellent following of the closed loop 0 dB line while retaining the original pilot gain level. Pre-filters A and B, not shown here, represented a progression between the response (b) without pre-filter and the response (d) with pre-filter C.

The oscillations that occur are a resonance of the closed loop system, and do not indicate an actual instability. McRuer has called the characteristic the "PIO syndrome". The clue to this is an excessively wide "attitude shelf" (described in §7.5.1). The shelf width at nominally constant amplitude is defined by the range in frequency between the end points, \(1/T_{\theta}\) at the low frequency end and \(\alpha_{tp}\) at the high frequency end. This width is visually obvious directly from the horizontal frequency scale in a Bode plot, but appears less obvious at first sight in a Nichols plot since the horizontal scale is no longer in frequency units. However, it can be identified readily from a low attenuation between the frequency points marked along the response plot\(^{15}\).

"Bobble" PIO

A tracking problem known as "bobble" may be described as PIO by pilots. For this reason it is mentioned here, though in the author's opinion it is not strictly a closed loop stability phenomenon even though it is associated with closed loop tasks. Bobble is a tendency for unexpected small and abrupt pitch attitude excursions. The only record found by the author in the literature is shown in Figure 8-3. This was Case 2-D in Boothe et al [1974]. It was given a Cooper-Harper pilot rating of 2 to 3 and a PIOR of 1 despite a slightly objectionable tendency to oscillate slightly when closely tracking a target. A second example from this reference, Case 8A, was severely affected by bobble. The pilot complained that the attitude control in tight target tracking was very objectionable, though the oscillations were small and non-divergent.

No flight record of the Case 8-A bobble is available to the author, but Figure 8-4 shows its pitch attitude frequency response. Its extreme form is due to the 16 radians/second short period frequency, an unnaturally high value imposed by artificial feedback of angle of attack. Attribution of an arbitrary pilot time delay of 0.22 seconds, equivalent to additional phase lag of 80°/Hz, gives an aircraft plus pilot open loop response with ample closed loop gain and phase margins. Adding a pilot gain of 8 dB, or 2½ lbs/degree of attitude error, moves the combined response as shown with 100° phase margin though only 4 dB gain margin. To become neutrally damped a pilot gain of 12 dB or 4 lbs/deg would be necessary, a gain level that is certainly possible but not likely in the author's experience. The stick force per g selected by the pilot for excellent manoeuvrability was itself only 4 lb/g.

The cause of bobble is to be found in the open loop time response characteristics. Aircraft with bobble problems invariably have excessive attitude dropback and pitch rate overshoot ratios. These features of the two examples here have already been shown in Figure 7-29 from Chapter 7, §7.6.2, in the discussion of pitch sensitivity. A significant part of actual aircraft tracking involves pre-cognitive rather than pure compensatory control, whether the target is in straight or turning flight. Having acquired the overall target in the sight, the pilot may then move the sight to selected parts of the target. At the end of each attempt to move the sight the attitude changes again, requiring further repeated attempts. The test pilot school experiment of Quinn [1984] discussed in Chapter 7, §7.5.1 and Figure 7-17, was certainly spoiled by bobble due to the severe dropback and pitch rate overshoot.

\(^{15}\) Some Nichols plotting routines do not identify the frequency along the response, a deficiency that greatly reduces their usefulness for handling qualities assessments.

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Bobble PIO is another manifestation of an attitude shelf. It appears in the Nichols plot as a knee in the response shape at phase angles typically between -60° and -90°, rather than as an intrusion into the closed loop resonance region as in the "PIO syndrome" noted above. In Figure 8-4 the extremely small attenuation that occurs between 0.5 Hz and 2 Hz contrasts with about -12 to -16 dB in more normal responses. For either problem of bobble or the PIO syndrome, however, the solution is the same in the author's design methods. A properly chosen stick command filter can reshape the attitude frequency and transient responses to eliminate both problems, as discussed in Chapter 10.

8.2.1 Early response shape criteria
Pitch attitude PIO of the kind discussed here is generally found only in combat aircraft, where the aircraft gain may be high because the stick forces are low in order to reduce pilot effort in manoeuvring at very high normal accelerations, in tasks genuinely requiring precise pointing of the airframe towards a non-stationary target. Up to the 1939-1945 war, aircraft had moderate wing loading and moderate or high wing lift slope. Consequently they tended to small or moderate values of $T_{02}$ and hence $\alpha_p T_{02}$ (see §3.2.2, §7.4.1 and Eq. 7.2) With moderate shelf width and negligible dropback, hence little tendency towards the "PIO syndrome", and with high damping, they possessed good $K/s$-like attitude response behaviour. PIO was almost unknown in them. Later swept wing types with increased wing loading, lower lift slopes due to high wing sweep, and higher short period frequencies particularly at supersonic speeds, were more prone to this problem [Gibson, 1990]. Some scale can be applied to the task difficulty by noting that pointing errors of one quarter of a degree while tracking, about 4 milliradians or mils, are towards the edge of satisfactory limits.

Transport and commercial aircraft generally have very much higher stick forces, with a correspondingly low aircraft gain $[\theta(\omega)/F_r(\omega)]$ deg/lb. Their attitude seldom needs to be controlled at a high bandwidth. Although such aircraft have large values of $T_{02}$ typically about 2 seconds in the landing configuration, so that the "shelf" commences at about 0.5 rad/sec, their short period natural frequencies are also typically very low and the shelf is not naturally wide. It is true that attitude is the primary visual cue used by pilots, but where manually controlled flight path precision is necessary it is supplied by painstaking trimming of power and the flight controls. A pitch attitude is often selected after take off for the initial climb, but it does not have to be adjusted continuously on a second by second basis and there is no penalty for some inexactness. Attitude PIO has not been reported in conventional aircraft of these classes, and no criteria were developed for them.

The following criteria methods were derived from experimental results obtained in the Calspan NT-33A variable stability aircraft, reported in Neal and Smith [1970], Boothe et al [1974] for "fighter up-and-away" flight, and in Smith [1978] for fighter landing approach. The data represented conventional aircraft response qualities dominated by aerodynamic and inertial factors, but with some added pseudo-high order effects in the form of command path filters. Generally the criteria attempted to define in different ways how far the aircraft response may stray from the optimum $K/s$-like shape before becoming unsatisfactory or unacceptable, though this was not specifically identified.

Neal and Smith
The "Neal and Smith" [1970] frequency domain criterion was an early attempt to identify the quality of the shape of the closed loop attitude response characteristics. This examines the overall pilot-aircraft closed loop dynamic characteristics through the pilot equalisation required to achieve (amongst other aims) a desired closed loop bandwidth, defined as the frequency where
the closed loop response phase lag is $90^\circ$. (This should not be confused with the bandwidth discussed in §7.5, which is defined by aircraft open loop characteristics alone. However, both formats in effect set an upper limit on the range of useful frequencies at which the pilot can use the airframe/system dynamics.)

This criterion was developed further in Radford et al [1980]. An important feature was the recognition of "flying qualities cliffs", that is closed loop attitude tracking dynamics which seem benign and well behaved, but which diverge with little warning into large resonance or over-control as the pilot attempts to increase the closed loop bandwidth. This sudden change can occur with only small changes in pilot gain and equalisation technique. These effects are not exactly equivalent to the causes of high order PIO that entail a loss of closed loop stability, which can be identified directly from a different part of the frequency domain as discussed later, but they are closely related. Robust absorption of wide variations in pilot technique is essential for good handling free of PIO tendencies, whether low or high order.

A more recent discussion of the criterion can be found in Anon [1991], and its application to an improvement in the Grumman X-29 flight control system is described in Clarke et al [1996]. The criterion was a significant pointer to the author of a pathway to new handling analysis methods, as mentioned in Chapter 7.

R.H. Smith

Another criterion for the pitch attitude response shape is given by Ralph Smith, [1979] (a different Smith). This measures the average attenuation in the aircraft open loop amplitude ratio per octave between 1 and 6 radians per sec. Less than 2 dB per octave indicates the PIO syndrome of an excessively flat shelf. An attenuation rate between 2 and 6 dB per octave indicates an averagely $K/s$-like or rate response that satisfies Smith's "No tracking hypothesis" mentioned earlier in §5.2.5. Attenuation greater than about 6 dB per octave in the above frequency range is associated with a sluggish response tending towards a $K/s^2$-like behaviour, that is to say an acceleration response with excessive phase lag. This can lead to a different form of slow PIO due to "overdriving". The pilot applies excessive control to obtain a more satisfactory initial response, producing an overshoot as the response eventually follows. The control is then reduced, again by too much initially, and so on.

The criterion fails to identify the bobble PIO problem of Case 8A discussed above, giving approval to the attenuation in the criterion frequency range but missing entirely the low attenuation at higher frequency.

The method addresses the instability problem associated with high order control systems rather indirectly, through the attitude response phase angle at a criterion frequency that is determined on the basis of the $K$, $K/s$- or $K/s^2$-like gain slope characteristics. However, it does not consider directly the attitude dynamics in the critical frequency range identified by the author, that is in the region of $180^\circ$ phase lag.

The "Gibson Criterion": first steps to a PIO solution

Observations on a number of transient and frequency response characteristics were derived independently by the author in his early research and published in Gibson [1982], as discussed in Chapter 7 (§7.3.1 and 7.3.2). Although they acquired the label of "The Gibson Criteria" in a number of other authors' papers and continue to be referred to as such to this day, many of them are less a set of precise criteria to be followed in design than observations of a general connec-
tion between physical measures of response characteristics and pilot opinions. The recommended frequency response boundaries in Figure 7-7 were based on the very small number out of all the experimental results that had satisfactory handling. By definition this excludes the possibility of PIO. The remaining boundaries of less satisfactory responses did make reference to the likelihood of PIO. The reference gave another criterion, specifically addressed to the PIO problem, which the author considered to be only partially successful. He has replaced it completely by other criteria which are given below.

In contrast to the views of Berry [1982] quoted in Chapter 1, it was the author's contention that it was indeed possible to identify from existing data the good qualities that should be provided by the dominant flight control laws in the new highly augmented fly by wire aircraft. While most criteria simply attempt to define whether the handling is good, indifferent or bad, the author's intentions have always been to identify good response characteristics that should be provided by control law designers. This was to be achieved, if feedback design alone was insufficient, by command path shaping filters such as those employed in the Tornado design. The PIO solution in the latter together with subsequent further improvements invoked only changes to these filters. The success of the final version was the starting point for the subsequent development of the author's criteria.

A feature of the frequency response shaping limits indicated in Figure 7-7 [Gibson, 1982] is the prevention of the attitude shelf shape to eliminate both the PIO syndrome and the potential for bobble with an emphasis on K/s-like responses at frequencies below an open loop aircraft bandwidth limit. The latter was chosen independently by the author for the landing approach at a phase angle of -120°, but had the same purpose as the contemporary version at -135° due to Hoh et al [1982] in Figure 7-15.

In addition to addressing the basic closed loop handling qualities, however, the boundary limits were extended further beyond the bandwidth limit into the higher frequency region containing characteristics that directly influence the tendency to PIO instability. It was by then clear to the author that high order PIO occurred at a frequency very close to that at which the aircraft open loop attitude phase angle reached -180° and that the pilot contributed negligible phase angle once locked into a PIO, in agreement with the "synchronous pilot" hypothesis of Ashkenas et al [1964]. The aim was to decrease the aircraft attitude response gain in this region relative to the bandwidth limit gain sufficiently to inhibit the possibility of inadvertently generating large PIO oscillations.

While the criterion was successful in identifying crucial features of a PIO prevention strategy, its further evolution by the author is discussed in later sections of this Chapter and in Chapters 9 and 10.

8.3 The low order "linear attitude PIO"

The concept of the low order linear attitude PIO (referred to in Chapter 7) resulted from an attempt by Ashkenas et al [1964] to derive the conditions for a linear pilot-aircraft attitude PIO, assuming conventional dynamics with no actuation or other added dynamics. They proposed that in the sustained PIO situation, the pilot can act synchronously with the oscillation as the target motion is now predictable and not random. In that case the pilot acts without time delay or phase equalisation - the "synchronous pilot", so that only the aircraft dynamics and the pilot gain need to be included. Their analysis of the conditions in which the pilot-aircraft attitude
closed loop could be driven unstable with a sufficiently high pilot gain gives the expression
\[ 2\zeta_s\omega_s - 1/T\theta_2 < 0 \quad (i.e. \quad 2\zeta_s\omega_s T\theta_2 < 1) \]  
(8.1)

Examination of the attitude dynamics of Eq. 3.2 shows that the maximum phase angle created by the numerator at high frequencies is +90°, and the maximum denominator phase lag is 270°. The total phase lag at high frequency can never exceed 180°, therefore. In theory, if the expressions given above become less than 0 or 1 respectively, the total phase lag will exceed 180° at intermediate frequencies. The physical cause is that given a large enough value of 1/Tθ2 relative to 2ζsωs, there is a range of frequencies at which the phase advance from the numerator is insufficient to prevent the denominator phase lag from taking the total lag to more than 180°. In this case a pilot-aircraft open loop response gain margin can be defined, and the closed loop can be destabilised by sufficient pilot gain. An arbitrary example of such attitude dynamics is given in Figure 8-5.

However, Ashkenas [ibid] shows that this is impossible with conventional aircraft aerodynamic configurations. Neglecting incremental elevator lift effects, the expression above can be approximated as
\[ 2\zeta_s\omega_s - 1/T\theta_2 \approx -M_q - M_\alpha < 0 \]  
(8.2)

Since both damping terms are normally negative, this requires one to change sign. No known practical configuration exists where this is possible. It was concluded that for aircraft with negligible control system dynamics, PIOs involving conventional pitch attitude are essentially impossible. This is indeed the practical experience of many decades of conventional aircraft prior to the use of powered controls. However, many experimental simulations achieved it by altering the pitch damping, arbitrarily assigning the necessary aerodynamic value in ground simulations or applying positive (i.e. destabilising) pitch rate feedback in airborne simulations. These therefore represented conditions which could not occur naturally. This was not always understood.

The simulation of this condition by Hall [1958, 1963] was discussed in Chapter 5, where the extreme and non-linear nature of the pilots phase equalisation generated in the low frequency/low damping short period region of Figure 5-3 was noted. It was pointed out by the author [1982] that Hall's dividing boundary between linear and non-linear pilot activity, though pre-dating the linear attitude PIO boundary criterion of Ashkenas, matched it remarkably well. This is shown in Figure 8-6, but note that the boundary is valid only for the simulated 7θ2. A record of severe PIO resulting from attempts to track with such dynamics is illustrated. Hall described the pilot activity as "a rather frantic switching". McRuer [1960] notes that the pilot square wave output distribution indicates highly non-linear behaviour resembling a relay controller, associated with the generation of extreme lead. This can be understood from the K / s²-like acceleration response initiated at very low frequency in Figure 8-5. Note that this non-linear pilot behaviour occurred in a linear aircraft simulation. The contribution of non-linear aircraft dynamics to the PIO problem is discussed in Chapter 9.

On the other hand, the subject of low order linear PIO was specifically examined by DiFranco [1967] where PIO tendencies were found. Eney's carrier aircraft approach simulation [1967], using a North American Navion four seat light aircraft (modified at Princeton University as a variable stability aeroplane) to represent naval jet aircraft, required heavy positive pitch rate feedback to achieve low short period damping, as was acknowledged and explained in the report. With the very large natural heave damping of the Navion represented by a Tθ2 of only 0-5 instead of a more typical 2-0 for naval combat aircraft, the low natural frequency and damping ra-
tio combinations lay well beyond the linear PIO boundary, and therefore did not represent realistic aircraft configurations of the time. The Navion was later modified to use direct lift flaps, so that subsequent simulations could represent naval aircraft $T_{P2}$ characteristics much more accurately, e.g. Miller [1969]. Eney conducted further carrier landing experiments with an actual carrier aircraft, the NF-8D [1969]. These extended almost into the "linear PIO" regime, where the handling was found to be very poor (Figure 7-13).

Although the validity of low order aircraft simulations which crossed this so-called PIO boundary was controversial, they did in fact stumble into problem areas of reduced pilot-aircraft closed loop stability margins caused by excessive aircraft gain and phase lag. Such experiments provided an unremarked pointer to later high order PIO problems introduced by improper design of fly by wire control systems, which might perhaps have been avoided by "remembering the past" (Chapter 2). The attitude response example in Figure 8-5 translates rapidly across to and over the 180° lag line at low frequencies and high amplitude ratios, with reduced stability margins when the pilot closes the loop. These remain a significant problem to this day in high order PIO, which were to appear with dramatic effect in the era of fly by wire aircraft.

8.4 The linear "high order PIO": the author's evolution of an understanding and solution

The 1976 Tornado accident that introduced the author to the problem is illustrated in Figure 8-7 [Gibson, 1978]. The pilot had the impression of a small turbulence disturbance followed by an uncontrollable pitch-up despite the fully forward stick input. The record showed a PIO of gross proportions in which the aircraft was responding to the controls, though out of phase with the stick. As has been the case in every other high order PIO, there was no malfunction. The impact on landing broke the undercarriage, though the aircraft was repaired and resumed test flying.

The "synchronous pilot" of Ashkenas et al. [1964] (§8-3), modelled as a gain element and assumed to apply control in anti-phase to the attitude oscillation, is clearly evident in the fundamental stick input oscillation. With no pilot phase contribution, the closed loop instability naturally occurred at the frequency where the aircraft attitude phase lag to control inputs was around 180 degrees as shown in the record. The concept of the synchronous pure gain pilot model became a powerful tool in the author's discovery of solutions to high order PIO and design criteria to prevent it. Though the pilot actions in PIO were later found to vary from the pure attitude-related gain model, often with highly non-linear behaviour, the fundamental pilot actions are always tightly synchronised to components of the attitude response as discussed in Chapter 9.

It is notable in Figure 8-7 that, as a result of the actuator rate limiting, the oscillations decrease in frequency as the amplitude increases in order to maintain the constant out-of-phase timing. However, the PIO clearly began in linear conditions. As modification of the actuators was out of the question, attention was directed to improving the linear control law design. The author decided to concentrate only on the aircraft dynamics in the 180° phase lag region, as discussed below. No point was seen in trying to develop a theory of transitional pilot behaviour between normal control and the full PIO. This policy of dealing with PIO as a specific problem of aircraft dynamics complete in itself, separately from considerations of general handling qualities, has subsequently proved to be correct and has led to the author's successful design criteria.

Figure 8-8 shows the calculated landing case pitch attitude frequency responses for four different pitch control law versions. A point of concern in the first version was the high gain at the 180° phase lag frequency, so that large oscillations could easily be generated by quite moderate
stick inputs. It had already become clear from other flight experience that the stick command gain at low speeds was too high, as it was excessively easy to saturate the pitch control system. In the complete absence of any other criterion whatever, the policy was adopted that there must be a margin from instability if any pilot again used the same gain as in the accident. The second control law version, with a substantial reduction in stick command gain at low speeds and doubled gain margin, was nearly in a flight cleared status at the time of the accident. It was approved for the continuation of flight testing and the resumed use of augmentation in take off and landing, which had been confined to direct link unaugmented operation after the accident.

The author had serious reservations about this decision because the linear dynamic characteristics of the second version were little changed from the first version. He circulated a written opinion that a PIO would eventually occur with the second version, though probably much less severe due to the increased gain margin. All pilots had agreed that landing in the unaugmented mode was easier, despite its sluggish nature, than in the first full augmentation version. Although the latter had given good, well damped and light control in the landing approach, the touchdown was imprecise and gave pilots the sensation of having to "feel for the ground". The author believed that this was caused by the transient lag in the pitch acceleration, Figure 8-8, though no evidence existed at that time of its significance. In the second version the transient acceleration lag had been scarcely reduced at all, and in both versions the lag was obviously much larger than in the unaugmented case where the only high order effect was the addition of conventional actuator dynamics.

Some pilots were satisfied by the second version, but others sometimes experienced a lack of precision or a small tendency to overcontrol in the landing. Some time later, just as the author had predicted, the incipient PIO shown in Figure 8-9 occurred. The pilot reported that he felt the onset of an out of control situation, that he froze his control inputs to suppress it, and that he was then able to land easily with a satisfactory touchdown. The record shows that shortly before touchdown the pilot initiated an oscillatory stick input similar to that in Figure 8-7 except that it did not diverge. This is actually a commonplace activity known as "stick pumping", widely seen in landing records of many aircraft as discussed below and quite typical of all Tornado landings.

The pilot did not actually succeed in freezing the controls, but only reduced the pumping amplitude and increased the frequency slightly. As expected, there was no repetition of the severe divergence of the first PIO because of the higher gain margin, even though the record indicates a higher pilot gain than the accident pilot had used. As the tailplanes were close to their nominal rate limit, the effective safety margin was unacceptably small. Further use of full augmentation for take off and landing was again prohibited until a final solution was developed.

**Stick pumping characteristics**

The stick pumping phenomenon is frequently seen and is a predictable activity identified by Bihrl [1966] while working at Republic Aircraft in the 1950s. He noted that in the few seconds just before touchdown, pilots would often engage in a rapid pitch control oscillation in phase with pitch acceleration, Figure 8-10. The frequency was typically about 6 to 10 radians per second, well above the short period, where the pitch acceleration had little or no phase lag relative to stick inputs. Due to the high frequency there was little attitude and even less flight path disturbance. The pitch acceleration amplitude was consistently around ±6.5 deg/sec², which Bihrl found to coincide with available data on the threshold of vestibular detection in humans. He concluded that pilots acted this way to generate confidence in pitch control as the speed reduced towards the stall when very precise flight path control was needed for a smooth and safe landing. The activity was also quite subconscious, all pilots being unaware of it.

Linear pilot induced attitude oscillations
The author had used the theory in the Tornado design process to ensure that there was adequate hydraulic pump flow capacity at idle engine rpm in the landing approach, and in fact found in flight records that pilots did subconsciously stick pump as predicted. The initial stick pumping in Figure 8-9 was somewhat larger than usual but was normal for this pilot and within the range of variability between 3 and 9 deg/sec$^2$ seen by the author in flight records$^{16}$. The reduced amplitude and increased frequency of pumping just before touchdown are consistent with the pilot’s description of an incipient PIO despite the apparent absence of clear supporting evidence to a casual observer.

A consistent feature of the stick pumping frequency in conventional aircraft, typically at 8 to 10 rad/sec, is that the attitude oscillation generated at the threshold pitch acceleration is very small, as shown in Figure 8-11. The attitude is usually less than a fifth of a degree peak to peak, and therefore normally unnoticeable. The attitude phase lag to control inputs would be close to 180 degrees. The Tornado stick pumping frequency was about 3 to 4 rad/sec, and at the nominal acceleration level the attitude would be around 2 degrees peak to peak. Pilot variability caused some to use larger pumping amplitudes than others. The most likely trigger for the accident and the subsequent incident seemed to be that the pilots suddenly became aware of the attitude oscillation. They were then presented unexpectedly with a ready-made PIO situation with the attitude already out of phase.

Although the author identified unfavourable stick pumping dynamics as the initiating trigger for the Tornado PIO problem, stick pumping was not a PIO trigger in conventional aircraft. The path to future prevention of PIO by design was clearly to ensure that the attitude dynamics in the stick pumping frequency region were made to favour the subconscious pitch acceleration pumping activity, and not to encourage the possibility of the unstable synchronous pilot-attitude PIO coupling which occurs at similar frequencies. In many PIOs, stick pumping is not evident or appropriate, but all occur near this critical frequency region.

In these records a secondary oscillatory stick input is also evident at about 2½ Hz. A peak at this frequency is often seen in measured pilot transfer functions, associated with the neuromuscular system. It appears to indicate pilot tension in difficult or critical tasks. In Figure 8-9, its amplitude is significantly larger in the second phase, when the pilot was recovering from a perceived incipient PIO, than in the first phase of conventional stick pumping. This high frequency oscillation or "dither" has no obvious functional purpose and plays no part in the author’s PIO theory.

The stick command filter
As the nominal pumping frequency was about the same in both Tornado control law versions, it was obvious to the author that a solution would have to alter the dynamics to increase the pumping frequency and to reduce the overall response gain still further. A feature of the pitch control laws up to this time was an invariant lag/lead stick command pre-filter. Its purpose was to improve the high speed pitch handling, as the response without a pre-filter would otherwise have had the unsatisfactory characteristic shown in Figure 8-2 (plots a and b). Analysis showed that a higher stick pumping frequency and gain margin in the landing case would result from removal of the filter at low speeds.

This led to the third version with a filter in which the lag/lead time constants were scheduled with airspeed to provide a pure gain at low speed. The stick pumping frequency increased to a

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$^{16}$These accelerations are significantly larger than the thresholds given in Hosman and van der Vaart [1978, 1980]. No explanation is offered here, but the tasks and environment were different.
value at which it was thought the attitude oscillation might be sufficiently small to avoid attracting the pilot's attention, though it must be admitted this was rather a guess. The gain margin increased to the same as in the direct link mode, some 10 dB better than with the first pre-filter version, Figure 8-8. The transient pitch acceleration lag was greatly reduced, improving the sense of connection needed by a pilot between control inputs and aircraft response. To obtain a similar PIO divergence would now require the use of a pilot gain some three times that used in the accident.

This solution has stood the test of time, in that no further PIO of this type has occurred since. The experience was the beginning of a new handling qualities methodology which avoided dependence on traditional measures of response characteristics such as frequency, damping, etc. The initial criteria were based on intuition and engineering judgement in the absence of any other data for analysis at the time. Subsequent assessment against increasingly available new data enabled the criteria to be developed, but the author's original ideas have never been fundamentally supplanted after two decades and continue to show excellent agreement.

8.5 Development of new criteria to prevent high order PIO

The first systematic studies of high order handling, that is to say handling with additional dynamic characteristics not present in the conventional response, were the Calspan in-flight variable stability experiments noted in §8.2.1. These studied a range of conventional short period characteristics to which a variety of linear first and second order lag type command filters were added to represent high order effects. It does not appear that these were intended to investigate PIO as such, but they did produce a number of examples. The results in Smith [1978], which became widely known as LAHOS from "Landing Approach High Order Systems", were particularly rich in PIO data for the critical landing task. This is fortunate, because it is in this task that most of the major PIO problems have occurred, and there has been little else available from which to gain a general insight into them until recent times. These data provided most of the basis on which the author's criteria were initially developed.

The author began to examine the Calspan flight data, which had become available only after the Tornado experience. The frequency and transient responses of all the configurations were calculated and their features analysed in comparison with the pilot opinions and comment data. The initial conclusions were reported in Gibson [1982]. These ideas contributed to the outstanding success of the British Aerospace FBW Jaguar research project, fitted with a quadruplex digital flight control system with no mechanical back-up [Smith, 1982, 1997 and Daley, 1984]. Eventually configured to a very high level of aerodynamic instability in pitch, with a time to double amplitude of 0.25 seconds, this aircraft had excellent handling qualities with no PIO tendencies.

Analysis continued further and resulted in the discovery of much new information that had remained unnoticed within old data, as discussed in Chapter 7 and reported initially in Gibson [1986]. This extended the author's earlier findings, summarised together in §8.1.3 above, and resulted in the enhanced criteria used in the equally successful EAP digital FBW unstable canard delta research aircraft that led ultimately to the Eurofighter 2000.

High order phase angle

Study of the LAHOS results in particular, but also the fighter up-and-away results, revealed a consistent pattern of differences between cases free of PIO and those highly prone to it. These are illustrated in Figure 8-12, the response characteristics of one specific LAHOS case with a
progression of additional lag pre-filters in the command path. The pilot rating changed from 2 (excellent) to 10 (loss of control in a PIO) as the added lag increased from zero in the basic case to a high value. The corresponding change in the shape of the frequency response with increasingly adverse high order effects was found to be entirely characteristic, and led to the generalisation in Figure 8-13 published in Gibson [1986].

The "low order" response typifies experimental cases free of PIO. In a classical "low order" response the phase lag never exceeds 180°. Note that here the phase lag does increase beyond 180°, an inevitable result for any aircraft with powered actuators driving the control surfaces. There are practical limits to the maximum gain that the pilot either can or ever wishes to apply, and sufficient attenuation at this crossover ensures that a satisfactory gain margin is maintained. Additional lag dynamics are minimised by good actuator design, and there is no important practical difference in handling from the pure or classical form. It is therefore considered legitimate to include such a response in the low order category.

The "high order" response typifies those cases which had severe PIO problems. Other responses progressed from acceptable to unacceptable PIO as their characteristic shape changed from the low to the high order type. The low order type transforms into the high order type by accumulating additional lag. This has often been due to an attempt to soften or attenuate an excessively abrupt or aggressive behaviour by a simple lag filter. Contrary to these expectations, a lag pre-filter increases the amplitude at the PIO frequency into which the pilot becomes locked, instead of providing the intended response attenuation.

Quantitative measures:
Phase rate and \( \omega_{(-180)} \), the frequency for -180 degrees phase angle

The assumption made in the original Tornado study, that attention should be focussed on the region around -180° attitude phase angle in order to find adverse high order effects, was found to be fully justified. Two quantitative measures were identified initially. These were combined in Figure 8-14, also published in Gibson [1986]. The pilot opinion rating Levels of Table 2-1 have been used in preference to the PIOR scale in table 8-1, since the pilot ratings are invariably dominated by the occurrence and nature of a PIO.

A close empirical correlation was found between the severity or absence of PIO and the phase rate \( \frac{d\phi}{d\omega} \) at \( \omega_{(-180)} \) deg/Hz, the slope of the attitude phase angle versus frequency at the frequency of the -180° phase angle point indicated in Figure 8-13. In a classical attitude response, the phase rate is effectively zero because the phase angle approaches but does not exceed -180°. As increasing amounts of higher order dynamics are added, the phase rate increases.

The frequency \( \omega_{(-180)} \) was also found to be very significant. This was noted in the discussion above on the amplitude of the stick pumping phenomenon and its possible role in triggering a landing PIO. It also has more general significance in indicating the physical possibility of large PIO amplitudes, as for a given pitch acceleration or control power the attitude response is inversely proportional to the frequency squared.

It is to be noted that while high order control laws have been implicated in many adverse effects it is not possible to quantify these effects from the order of the system. Properly designed, as discussed further in Chapter 10, it should be completely feasible to generate response shapes that possess good low order characteristics, that is low phase rates and high "PIO frequencies". This may seem to be an unnecessarily obvious statement, but as noted earlier it was a widely but
wrongly held belief in many quarters that high order control laws necessarily created adverse high order handling characteristics, e.g. Berry [1983].

**Transient acceleration lag**

Also apparent in Figure 8-12 is an increasing lag in the time of the peak acceleration together with a reduction in peak amplitude in a step input transient response as the lag in the command filters is increased. This is again entirely characteristic of increasingly PIO-prone responses, indicating another "shape" to add to the range of non-modal parameters useful for handling analysis. It is immaterial that a pure step input is never applied by a pilot, as the matter of concern is the transfer function qualities of the aircraft alone.

In a conventional aircraft, the angular accelerations coincide with movements of the pilot's controls, which are directly linked to the control surfaces. If there is excessive lag in the connection between the pilot and the control surfaces, the feeling of immediacy of response which a pilot needs for predictable control is lost. The author found that an acceleration peak lagging a step input by more than 0.25 seconds was generally unsatisfactory, and was certainly catastrophic if it was as much as 0.5 seconds. This lag is closely related to the phase rate, which is a simplified representation of the additional high frequency dynamics affecting the transient acceleration.

When tight control of attitude is needed in critical situations, such as landing, the pilot may be induced by lack of immediate response to apply an excessive open loop input, followed by a reversal which sets up an oscillatory overcontrol situation. A simple assessment of the transient lag in the direct path between the pilots input control and the corresponding control surface will serve to give a good indication of the likelihood of PIO tendencies. That is, if a step input is applied at the pilot's stick, the control surface should respond with as near to a step as can be provided through the actuation system, with the amplitude adjusted by the control laws to generate the desired angular acceleration. This simple test is entirely independent of the order of the control law or of any feedback loop design characteristics. Many handling deficiencies can be tolerated if the pilot has sufficiently direct access to the attitude response acceleration.

**8.5.1 Further enhancement of the author's criteria**

Over a number of years, the author's experience with the use in design of the above PIO criteria and the increasingly available evidence from other PIO data led to further enhancements.

**Amplitude limitations**

Freedom from high order PIO can also be assured if that the maximum response amplitude that can be generated by the use of full stick inputs at the PIO frequency (i.e. where the phase angle is -180°) does not exceed a very small value, say one or two degrees. Note that this does not guarantee good general handling qualities, which depend on other factors. It was also noted in Gibson [1982] that no PIO had been found where the attitude response gain at the PIO frequency was less than 0.1 deg/lb, regardless of the handling dynamics. Such limitations were employed in the author's design criteria for some time.

It is not always easy to satisfy the desired amplitude limit especially in the case of reversion to simpler standby-control laws after the occurrence of system faults. The criteria were essentially based on linear characteristics and did not account for non-linear effects such as actuation rate limiting. As a result, difficulties in formal flight test clearance of a new design could arise as there was no criterion to show that it would be safe from PIO if the desired stringent maximum attitude response gain limit was exceeded. To quantify this shortcoming, the author studied the
LAHOS results again. By taking account only of those cases with obviously high order PIO, that is excluding any where the PIO was of the bobble variety, it proved possible to define Levels 1, 2 and 3 amplitude ratio limits as shown in Figure 8-15(a) [Gibson, 1995a].

These amplitude ratio limits are applicable at all amplitudes up to full stop-to-stop inputs. Such a provision has sometimes been rejected as unrealistic, on the grounds that full control amplitudes would not be used in flight. Nothing could be more mistaken. Mitchell and Hoh [1996] list the extreme range of pilot stick forces applied, the amplitude and frequency of motion, and the range of flight conditions in PIO events. Klyde et al [1995] and Anon [1997] provide many examples of previously unpublished PIO records. From these it is clear that no size of aircraft is immune to high order PIO, nor does the size and heaviness of the pilot's controller diminish the probability of extreme control activity. Full stop to stop control inputs have been made in aircraft as large as the Boeing 777 prototype and Airbus A-320, the latter in a lateral PIO lasting some 50 seconds before control was regained.

In a linear system, of course, it does not matter what amplitude of control input is applied, but large control inputs in many flight control systems have caused rapid and dramatic changes in the response dynamics due to saturation effects. The only way to ensure the absence of such effects, or at least that they are satisfactorily restricted, is to construct the control law design to accept full amplitude inputs while retaining at least quasi-linear dynamics, by eliminating or ameliorating the saturation effects that guarantee PIO. This is discussed further in Chapter 9.

The average phase rate

The author eventually found that his original phase rate criterion occasionally gave an unduly pessimistic measure, due to a locally large phase rate at \( \omega_\ell(180) \) that diminished considerably at frequencies higher than the PIO point.

A similar measure, the "phase delay", had been derived for the Bandwidth Criterion in Hoh et al [1982a], noted in §7.5.1 and defined in Figure 7-15. It is derived from the excess phase lag between the PIO frequency and twice that frequency, and so captures a more realistic slice of the excess high order lag effects. Expressed in units of seconds, it was stated to be an alternative to the LOES time delay, and that it was more meaningfully based on phase lag effects which were the true basis of high order handling qualities problems. Agreeing with the latter but disliking any use of the word "delay" in relation to frequency response phase angles, the author adapted it as the average phase rate given in Figure 8-15(b).

The author's definition of average phase rate is:

\[
\text{Average phase rate} = \frac{-\left(\varphi_{2\omega_\ell(180)} + 180\right)}{\omega_\ell(180)} \text{ deg/Hz} \tag{8.3}
\]

where \( \omega_\ell(180) \) is expressed in Hz.

The phase delay definition in Hoh et al [1982] is:

\[
\text{Phase delay } \tau_P = \frac{-\left(\varphi_{2\omega_\ell(180)} + 180\right)}{(57.3 \times 2\omega_\ell(180))} \text{ sec} \tag{8.4}
\]

where \( 2\omega_\ell(180) \) = twice the -180° phase angle frequency and is expressed in radians per second.

Substituting \( 2 \times 2\pi \times \omega_\ell(180) \) in Hz for \( 2\omega_\ell \) in rad/sec in (8.4):
Phase delay $\tau_p = \frac{-(\varphi_{20k-180} + 180)}{(57.3 \times 2 \times 2\pi \times \omega_{k-180})} = \frac{-(\varphi_{20k-180} + 180)}{(720 \times \omega_{k-180})} = \frac{\text{average phase rate}}{720} \quad (8.5)$

The numerical value of the average phase rate of any given response is somewhat less than in the author's original phase rate definition, reflected in the revised boundaries in Figure 8-15(c). In practice the average phase rate and the phase delay are interchangeable. Phase delay was incorporated in the standard US specification [Anon, 1987]. However, as a high order measure it is specified there on its own, which does not account for the importance of the PIO frequency or the response gain.

**Application of the Figure 15 criteria**

The Figure 15 criteria boundaries represent an analysis of a range of response dynamics that is relatively small compared with the numbers of PIO events that have actually occurred. Many of the configurations were flown only once by only one pilot, and the opinion rating attached to it might not be repeated exactly by other pilots. Other configurations might have led eventually to a PIO given enough exposure to more pilots and more difficult flight conditions. Parrag [1998] points out that there is a considerable "grey area" in deciding whether an oscillation should be called a PIO or pilot over-control resulting from unfamiliarity or insufficient adaptation. It is unlikely that the exact boundaries of Level 1, Level 2 and Level 3 PIO qualities could ever be precisely delineated for all possible examples of high order PIO.

Some interpretation is necessary in the meaning of the gain limits, as it can be the case that a response might be classed as Level 2 by its phase rate and frequency, but as Level 1 or Level 3 by the gain criterion. The author would interpret the effect as signifying better or worse PIO characteristics, so that any oscillation would be unlikely to diverge in the Level 1 gain example but would probably be divergent in the Level 3 gain example. The interpretation would be that the response should still be classed as Level 2 in the first case but must be downgraded to Level 3 in the second case. The severity of the high order characteristics is also related to the slope of the response plot across the -180° phase angle line, in the sense that the more shallow the gradient the worse the PIO tendencies. The author would interpret a response that crossed one Level boundary limit into the region above as downgrading the response to the higher (worse) Level.

With three different parameters to be assessed, one of them potentially requiring some interpretation, it cannot be claimed that this criteria set is guaranteed to make absolutely accurate quantification by pilot rating of the PIO tendencies of past configurations a certainty. What is certain is that the further outside the Figure 15 Level 1 limit boundaries that the response of a new design penetrates, the worse its PIO tendencies will be. On the other hand, responses just within the Level 1 limits in all respects are unlikely to experience serious high order PIO, but they still possess some residual high order characteristics. The classical aircraft of old without power control actuation would plot far out of sight to the right on the bottom edge of the phase rate figure, with a response gain equally far out of sight downwards on the gain plot. Between this ideal extreme and the practical reality lies a gradation of increasing high order effects that will eventually lead to PIO tendencies. Except for unavoidable actuation dynamics, these effects are entirely artefacts of, and therefore under the control of, the control law designer.
The author’s adoption of "Level" boundaries carries no official status, but reflects only his own analysis of the experimental data based on pilot comments and ratings according to the "Level" concept. The term "Level" itself formally implies an official acceptance standard of nominated qualities set by the customer. It will be recalled that the definition of Level 1 includes the Cooper-Harper 3 pilot rating (CHR 3) with "some mildly unpleasant deficiencies" (Table 2-1). Today’s designers, with extremely wide powers to produce the response dynamics of choice, are freed from the old restriction of hoping that the pilot can adapt to whatever dynamics fall out from the design process.

The good designer should not simply be content to obtain the minimum standard just within the Level 1 limits. The problem of PIO has been too serious too often for anything but the best efforts to prevent it. Rather than just satisfying some formal standard, the most important issue is how far into the less than perfect region of dynamics the designer may permit the response to enter while retaining an absolute guarantee of freedom from high order PIO. Put in another way, the issue is that the designer should set handling qualities aims equivalent to CHR 2, or better still, CHR1 which is "excellent, highly desirable". By illustrating factors that have been associated with PIO ranging from severe to mild or none at all, the Figure 15 criteria point to the response dynamics to be avoided by the maximum possible margin to ensure the absence of PIO.

The concept of an optimum design aim for handling qualities designated Level 1* (Level 1 star) was introduced for the EAP control law design in Gibson [1985a]. This specified a minimum PIO attitude frequency of 0.65 Hz and maximum local phase rate of 70 deg/Hz. Maximum PIO attitude gain limits were set at -16 dB deg/lb for phase rates better than 70 deg/Hz and -20 dB for phase rates of 100 deg/Hz. These gains equate to a stick force of ±6.3 lbs and ±10 lbs respectively to generate a ±1 degree attitude oscillation at the PIO frequency. Assuming that the response remained linear, this ensured that the maximum possible attitude response at the PIO frequency could not exceed some ±2 degrees or so as a result of the pilot’s inceptor travel limits.

A secondary criterion limit of 0.18 seconds was placed on the initial lag in the attitude acceleration time response to a step stick input. The criteria were applied to the EAP with the certainty that PIO would be prevented, an optimism that proved to be fully justified. This success has continued with the Eurofighter 2000.

With the updated criteria of Figure 15, the author recommends the following Level 1* limits:

- Maximum average phase rate of 50 deg/Hz, equal to a phase delay of 0.07 seconds.
- Minimum attitude PIO frequency of 1.0 Hz.
- Maximum attitude to stick force gain of -20 dB or 0.1 deg/lb at the PIO frequency.
- Maximum attitude acceleration lag of 0.18 seconds in the time response.

Finally, it should be remembered that the criteria here are based on linear responses, and that they cannot detect or prevent PIO of the low order type (§8.2) that are prevented by exercising control over the response shapes (Chapter 10). Unlike the formal Level concept that permits reduced qualities depending on the probability of system failures that lead to the reduction, the Level 1* criteria in Figure 15 should be the aim for all failure states of the flight control system.

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17In all three designs, an equally important factor in the absence of PIO was an exacting specification of actuation performance, rate and acceleration limits. See Chapter 9.
8.5.2 Testing the author’s criteria

The Tornado solution

Since the Tornado PIO problem initiated the development of the author's PIO criteria, but was solved without them, it is of interest to see how the original solution fits the current successful forms. The four landing approach responses from Figure 8-8 are plotted in Figure 8-16 against the PIO gain, average phase rate and PIO frequency criteria boundaries [Gibson 1995b].

The criteria identify the first and second versions control laws as having similarly poor PIO dynamics. As the author had surmised at the time, the most significant difference between them was the PIO gain, effectively the gain margin. This is indicated here as unacceptably small in the first version and an unsatisfactory Level 2 in the second. The third version is identified as having satisfactory PIO dynamics with Level 1 gain margin, while the unaugmented system has negligible PIO dynamics and Level 1 PIO gain, with a slightly low PIO frequency. All cases are identified correctly according to the actual flight experience.

This illustrates an important principle. Many criteria have fallen by the wayside because they were insufficiently tested against data other than those from which they were developed. The criteria here were derived from sources such as LAHOS and other material without direct reference to the Tornado data. The good agreement in Figure 8-16 between the retrospective analysis and the flight experience typifies a necessary condition for the acceptance of such criteria.

An in-flight experiment in the Calspan Learjet

Another unique opportunity to test the new criteria was given by a short flight experiment carried out by the author at British Aerospace, in the variable stability Calspan Learjet 25 [Gibson 1991b]. A pitch tracking exercise was set up with the basic Learjet pitch dynamics, which had significant attitude dropback. A second order stick filter with three damping values was designed to investigate the effect of PIO gain. Figure 8-17 shows the frequency responses and a table of other features. Although the aircraft was well damped, the filter gave a range of apparent damping from high to rather low with a high oscillatory frequency. The essentially Level 1 features of Case 1, with the dropout practically eliminated by the filter, were flattened progressively with reduction in filter damping to increase the PIO gain and pitch rate sensitivity measures. All cases had a high PIO frequency and all had a similar quite small transient acceleration lag, suggesting relatively small high order effects, which was better matched by the small variations in average phase rate than by the wider spread of the localised phase rate.

The author, in the aircraft during the experiment, was able to observe the handling directly and to relate his own observations to the pilot’s comments given at the time. Case 3 generated a tracking PIO in the form of a small, rapid oscillation, but though rated Level 3 it gave no sense whatever of an overall loss of control and occurred only when tracking. It was associated with poor effective damping, high PIO gain and excessive pitch rate sensitivity rather than with the high order lag. This is evidenced by the absence of tracking PIO when the stick force gradient per g was doubled. Case 1 was an excellent Level 1, in accordance with its essential satisfaction of all the criteria indicated. Case 2 lay in between and was rated accordingly as Level 2.

Although this was a very limited experiment, it was a good illustration of the handling assessment made possible by the criteria, since the pilot rating followed closely to the author’s expectations. It emphasised the importance of excessive sensitivity and PIO gain which were the primary cause of the PIO that did occur, and showed the importance of small acceleration lag and high PIO frequency in maintaining the pilot’s sense of direct control.
It also illustrated how the pitch sensitivity in Case 3 could be excessive even when the stick force gradient was a quite heavy 6 lb/g, and only became satisfactory when the latter was increased to a very heavy 12 lb/g. To put this in context, typical combat aircraft stick forces per g might be around 2 to 3 lb/g, though they would still require pitch rate sensitivity within the criterion range. To achieve this requires proper shaping of the frequency response as discussed in Chapter 7 and later in Chapter 10.

8.6 Other "equivalent" criteria

Formal requirements, e.g. Anon [1980, 1987 etc.] contain little in the way of specific criteria to prevent high order PIO, apart from stating that they should not exist. Their principal method for dealing with high order effects is the concept of "time delays". These are in the form of a single equivalent or effective delay which is taken to represent all the high order effects.

A critical assessment of time delay criteria

The principal time delay method is the "low order equivalent system" or LOES. This was discussed in Chapter 7 with respect to the analysis of general handling qualities, where the author concluded that LOES is an inadequate design and analysis tool for fly by wire control systems with excellent but non-classical response characteristics. LOES has been used successfully for aircraft with a reasonably "classical" type response. A small equivalent time delay is a clear indication that PIO due to high order dynamics will not occur. Large equivalent delays are an equally clear indication of poor dynamics likely to lead to PIO.

The term "time delay" is convenient general purpose label when discussing high order effects. It is used by the author in the case of flight path time delay (Chapter 7). However, great care must be exercised in understanding the implications. Difficulty can arise with the use of time delays to represent the significant high order effects. Flight experiments have shown that, with larger delays in particular, pilots can distinguish the time-shifted nature of a delayed acceleration onset in a time response from the sluggish effects of a lag, even if the frequency responses match reasonably well over the limited range required for this method. Studies of large aircraft have led to calls for greater allowable equivalent time delays in their specification, since they seem to be more tolerant to time delays than smaller aircraft.

The Calspan fighter and LAHOS cases are exactly described by their basic short period mode characteristics plus the added command pre-filter lags, and substitution of the lags by artificial equivalent delays and replacement of the short period parameters by equivalent ones cannot improve the modelling. This is likely to be even more so when the high order dynamics are rather more complex. Referring back to Figure 8-12, casual observation does suggest that the increasingly high order changes from case 2-2 to 2-10 might be represented approximately by an additional time delay. This applies not only to the frequency response but to the transient response - provided that this is applied only to the attitude and the pitch rate response. This seems to have been the origin of the LOES concept.

The so-called "effective time delay" was another time domain concept illustrated in Figure 8-18, approximating a time delay in the rate response. Limits are set at 0-1 seconds for level 1, 0-2 seconds for Level 2, and 0-25 seconds for Level 3 [Anon, 1980]. Its proponents did not appear to consider accounting for the acceleration characteristics directly, which to the author is a physically meaningful parameter at the heart of high order control difficulties.
While the effective time delay of 0.3 seconds in LAHOS Case 2-10 is certainly large, is greater than the nominal Level 3, and indicates a serious handling problem, other examples are less successful. In the Figure 8-19 example, the effective delay is only 0.1 seconds, a Level 1 value that entirely fails to identify the actual Level 3 handling. In the Figure 8-20(a) example, the effective time delay is 0.18 seconds, a Level 2 value despite the uncontrollable PIO and pilot rating of 10. The first and second versions of the Tornado control laws gave effective time delays of about 0.13 seconds, a good Level 2 value completely inconsistent with the PIO event. The final version delay was 0.1 seconds, which is Level 1, and is a relatively small decrease in view of the profound improvement.

**Lag effects - interactive not incremental**
The value of an additional pre-filter lag time constant cannot define its effect on handling characteristics or the likelihood of high order PIO. There are many examples from these cases where a small range of time constants produces effects ranging from excellent, to no change in the rating but with altered characteristics, to catastrophic. There is no obvious relationship to the size of the lag because the lag dynamics in each case interact with the basic dynamics to which they are added in entirely different ways. Examples of two responses which lie at opposite ends of the spectrum are sketched in Figure 8-20.

In Figure 8-20(a) the resonance due to the basic low damping in Case 3-1 is not as much of a problem for closed loop control as might be thought. Pilots are able to ignore oscillations of the aircraft which are obviously due to its poorly damped open loop dynamics, and can follow the average response between the attitude peaks. Although the handling was not optimum, it was acceptable, and with its negligible high order features it was actually rated slightly better for the touchdown than for the approach. For Case 3-3, adding a filter with a 0.25 seconds lag time constant moved the resonance into the PIO region with high PIO gain and phase rate. The transient acceleration peak time increased from 0.16 to 0.44 seconds (though the pitch rate effective time delay was only 0.1 seconds) The handling was Level 2 in the landing circuit, that is acceptable but unsatisfactory, but uncontrollable PIO resulted at every touchdown attempt.

In Figure 8-20(b), the attitude dynamics of Case 8A (shown in Figure 8-4 also) caused a severe bobble when tracking despite the well damped short period mode, because of sharp dropback with large pitch rate overshoot (§8.2). Despite excellent open loop manoeuvring qualities, these oscillations could not be ignored by the pilot. Adding a filter with a 0.3 seconds lag time constant, greater than the previous example, created negligible high order effects in Case 8D. The transient acceleration peak time increased only from 0.05 to 0.1 seconds. The attitude response was converted into excellent K/S-like dynamics with very good tracking and no PIO tendencies. This is a rare example of a simple lag pre-filter that was entirely beneficial.

Although LOES derives a pseudo-time delay added to artificially derived short period dynamics that are not the original dynamics, the idea gained currency that the equivalent and effective time delays are much the same and similar in effect to phase lags. Referring back to Figure 8-4, the effect of adding a time delay directly to Case 8A is shown (representing a pilot delay). Its value of 0.22 seconds is smaller than the lag time constant added to create Case 8D, but its effect on the frequency response is profoundly different. In the time domain, it would not alter the abrupt dynamics in any way except to add a confusing hiatus in the response that would make the handling even worse rather than excellent as with the use of the lag filter.

In the author's experience LOES is very much second best to a proper analysis of the actual high
order dynamics. True time delays due to digital FCS effects are usually almost negligible in handling quality terms if they are small enough for satisfactory system stability. The author has found that the use of effective or equivalent rate response delays cannot be relied upon to give an accurate and reliable quantitative identification of PIO problems. At the time of writing, this view has been endorsed by a formal decision to delete reference to LOES methods in the handling qualities specification for the Eurofighter 2000, and to depend solely on the author's methodology by which its control laws have been shaped for excellent and PIO-free handling.

8.7 Unanswered questions

The description of high order handling effects by the methodology discussed here is quite comprehensive but highly multi-faceted, and it is quite clear that there is no unique answer to the question "How exactly will any given response measure on its own affect the pilot?"

During his research on PIO problems, the author has had occasion to consider to possible effects of parameters not incorporated in his criteria. Some are briefly discussed below.

The PIO acceleration gain

If the prediction is for poor tracking performance with high order PIO tendencies, how seriously would the oscillatory behaviour affect the pilot? Figure 8-19 shows the pitch attitude frequency response of Calspan Case 4D from Boothe et al (1974) and a record of a tracking attempt. This began with severe oscillations that became divergent as the target range reduced to a few hundred metres. The response dynamics and gain level are not much different from those of the Learjet experiment Case 3 in Figure 8-17 [Gibson 1991b] and unlike the latter its pitch rate sensitivity only just exceeded the upper limit of Figure 7-27, yet in the Learjet Case 3 the tracking oscillations observed by the author in flight (though not recorded) remained less than ±0.5 degrees as estimated by the pilot.

The author has considered, though not exhaustively, the possibility that the pitch acceleration gain at the PIO frequency may influence the pilot in the need to track with large enough inputs to excite more than the detection threshold of pitch acceleration. Increasing the PIO frequency will produce larger acceleration response and reduce the likelihood of a large amplitude PIO even if the PIO gain is greater than desired. Mostly due to its higher PIO frequency, the acceleration gain of Learjet Case 3 is about 15 deg/sec²/lb, 2½ times that of Case 4D. Hence, somewhat like the stick pumping in landing, the pilot of Case 4D may have felt a little disconnected whereas all the cases in Figure 8-17 may have had a high enough acceleration gain at the PIO frequency to avoid the problem.

An increase in the PIO attitude gain margin to satisfy the desired Figure 15 limit also reduces the acceleration gain at a given PIO frequency. However, the trade-off appears to be that even if the pilot gain is influenced by the need to generate enough pitching acceleration, this will be insufficient to create closed loop attitude instability. It was noted earlier from Gibson (1982) that PIO did not seem to occur given a low enough attitude gain at the PIO frequency. When the stick force in Case 4D was increased to 11 lb/g for flight refuelling, a notoriously difficult task, there was no oscillation tendency and the handling was rated Level 2. This parallels the Case 3 example discussed above (Figure 8-17) when its stick force per g was doubled from 6 to 12 lb/g, and the early Tornado examples in Figure 8-16 where the PIO gain was progressively reduced with an accompanying improvement in pilot ratings from Level 3 to Level 1.
The author surmises that the average phase rate upper boundary limits in Figure 8-15 could be relaxed to an upward slope permitting more phase rate as the PIO frequency increases. This may be academic in practice, because it is in the nature of the attitude frequency response that the higher the PIO frequency is pushed, the lower the resulting phase rate is likely to be.

Such considerations lead to the conclusion that the PIO attitude gain can never be too small and the PIO frequency can never be too high. The achievement in control law design of the Level 1 criteria in Figure 8-15 satisfies this philosophy sufficiently in practice.

Inceptor displacements
It is usual to use the stick force as the primary measure of pilot inputs, but the possibility cannot be ruled out that the movements of the pilot's inceptor may be significant in PIO problems in their own right. The overall handling qualities are certainly affected by the inceptor displacements as was noted in §6.3.2.

It was noted in Gibson [1986] that, with its rather stiff stick feel gradient of 22 lb/inch, the attitude response gain in Case 4D above was 7 deg/inch of stick movement at the PIO frequency, an intuitively excessive amount. In the author's Learjet Case 3 result, with similar dynamics but with a feel gradient of 8 lb/inch, the attitude gain was less than 3 deg/inch. The author surmises that this difference may also have been another factor in reducing the tendency to excite large oscillations exhibited in the latter case. By contrast, the PIO-free FBW Jaguar and EAP had stick feel gradients of 5 and 8 lb/inch respectively, and with PIO gains designed to more optimum levels the attitude responses were less than 1 deg/inch at their PIO frequencies.

The author made tentative proposals to incorporate secondary design criteria for the Eurofighter based on the attitude gain relative to stick displacement in Gibson [1987c]. Such criteria cannot be as generally applicable as those based on stick force. The wide range of stick travels found in different aircraft would have a major impact on the specified PIO dynamics, which may be unreasonably restrictive. It does seem likely that any PIO tendencies that exist will be exacerbated by the use of unduly small stick travels, which as shown in Chapter 9 readily induce the instantaneous application of full travel inputs at the commencement of a PIO.

In-flight simulation investigations would be necessary to isolate any effects related purely to the influence of the stick travels. Since control laws can be designed with confidence of achieving excellent PIO resistance, it is a question that may never need to be answered, but it remains a gap in knowledge.

Generality of the criteria
The criteria have been formulated using material from combat type aircraft and manoeuvres with relatively conventional stick force and gradient characteristics. To determine whether any of the numerical limits prescribed might have to be altered to accommodate other types of stick, particularly the PIO gain, or other classes of aircraft such as commercial airliners, it would be necessary to conduct experiments to that end. Nevertheless, in the increasingly available number of PIO records from all classes, the author has found no example of attitude PIO which did not conform accurately to the principles of behaviour discussed above, or which could not have been prevented by the application of the design principles presented in this thesis.

The author has been unable to develop high order roll PIO criteria to the same extent due to lack of data, but has found that the phenomenon is identical in principle and has addressed
the problem with similar design criteria. The PIO gain seems better quantified by the roll angle generated by full stick inputs, limiting this to say ±5° in up-and-away flight and substantially less in the landing. The PIO frequency should ideally be kept above 1 Hz. These are tentative design aims that have worked well. The average phase rate and the transient roll acceleration lag limits have been taken as identical to the pitch values. Duda and Duus [1997] examined a number of linear frequency domain PIO criteria, and concluded that of these the phase rate criterion of Figure 8-14 was well suited to the roll axis.

Figure 8-21 illustrates data from another short experiment in the Calspan Learjet where a number of roll command lag pre-filters were added to a nominally good roll response for landing. The author’s pilot rating predictions were very closely fulfilled, including a 10 for the largest lag pre-filter. Establishing reliable roll PIO gain limits for different Levels and for Level 1* would require much more experimentation, however.

8.8 Summary

This chapter has discussed the nature and extreme importance of the pilot induced oscillation phenomenon, or PIO. Only linear PIO and the associated aircraft dynamics are considered here. There are two main classes:

Low order PIO
These are typical of conventional aircraft and essentially result in poorly damped coupling with the pilot. They can be subdivided further:

- The "PIO syndrome", a result of poorly shaped aircraft open loop dynamics that lead to problems of closed loop droop and resonance. These force the pilot to adopt considerable compensation dynamics to avoid oscillatory coupling when tight attitude tracking is attempted.
- The "bobble PIO", a result of an excessively abrupt angle of attack response that creates a sharp attitude disturbance every time the pilot initiates or ends a pitch control input. It seems to be more a problem of precognitive pursuit tracking than of compensatory closed loop tracking, but is often described as PIO by pilots.
- The "overdriving PIO", a result of low bandwidth aircraft open loop dynamics. The pilot applies excessive control to speed up the initial response, but the response overshoots because it is so slow that the end result cannot be predicted, setting up a continuous cycle of slow overcontrol when tight tracking is attempted.

While these were sometimes unavoidable with response dynamics dominated by aerodynamic and inertial properties, they are quite straightforward and could be alleviated by quite modest control augmentation systems.

- The "linear low order PIO" was the result of early variable stability simulations of a region of short period frequency and damping properties that could not occur by natural aerodynamics but were generated by the use of control feedback. Of little practical importance at the time, they contained much of the problem dynamics of the later high order PIO. They might have contributed to an understanding if they had not been forgotten by then.

High order PIO
These are typical of fly by wire aircraft and often resulted in unstable coupling with loss of control. With response dynamics dominated by the flight control system, designers inadvertently
introduced non-classical handling qualities often typified by excessive phase lags. The author has found that the offending high order dynamics are characterised thus:

a) PIO always occurs at or slightly higher than the frequency where the open loop phase lag reaches 180°, the "PIO frequency".
b) The open loop phase lag reaches 180° at low frequency, typically well under 1-0 Hz.
c) The phase lag increases rapidly with frequency in this region.
d) The open loop response gain at the PIO frequency is large, allowing large amplitude oscillations to be forced within the limits of the control stick travel.
e) In the time domain, there is a significant lag in the onset of the attitude acceleration

The author's development of criteria to identify and eliminate high order PIO by design is presented. They have been successfully applied to a series of advanced fly by wire aircraft, the Tornado, the FBW Jaguar, and the EAP and the Eurofighter 2000 unstable canard deltas.
Figure 8-1  Tornado tracking assessment in [Gibson, 1978], based on criterion of Neal and Smith [1970]. Pilot assumption: gain and time delay only. Pilot gain selected to achieve zero droop at 0.25 Hz.

Figure 8-2  Elimination of Tornado pitch tracking PIO by command pre-filter improvements
In plots a and b with no pre-filter, an excessive extent of near constant gain creates the possibility of pilot-attitude closed loop resonance, the "PIO syndrome". In plots c and d, the use of a command pre-filter reshapes the response to optimise the pitch tracking characteristics.
Figure 8-3 Example of moderate "bobble PIO" in a tracking task, from [Boothe et al., 1974].

Figure 8-4 Example of frequency response of a case with extreme bobble effects. (Case 8A, [Boothe et al., 1974]: further details of this case can be found in Figure 7-30)
Figure 8-5 Example of "linear low order PIO" characteristic. Pilot-aircraft closed loop instability is possible due to the attitude phase lag exceeding 180°, according to [Ashkenas et al, 1964].

Figure 8-6 Effect of "Linear PIO" on pilot activity from [Hall, 1958, 1963].
Figure 8-7 Tornado landing PIO, 1976, from [Gibson, 1978].

Figure 8-8 Tornado pitch attitude responses at landing: solution to PIO by development of the command pre-filter.

The unaugmented and third version pre-filtered dynamics are PIO-free.
Figure 8-9 Tornado incipient landing PIO with second pre-filter version
Figure 8-10 The stick pumping hypothesis [Bihrlle, 1966]. The pilot pumps the stick to obtain a pitching acceleration nearly in phase with the stick.

Figure 8-11 Stick pumping amplitudes at constant peak pitch acceleration
Figure 8-12 Response trends to PIO in LAHOS Case 2 data from [Smith, 1978].

Increasing added high order lag causes:
1. Increasing gain at -180° phase angle
2. Increasing phase lag increment per unit increase in frequency
3. Decreasing frequency at -180° phase angle
4. Increasing lag in transient pitch acceleration
Figure 8-13 General PIO tendency trends of attitude response shapes

Figure 8-14 Attitude response data for PIO criterion [Gibson, 1986]
Additional amplitude requirement:
Attitude gain at 180° phase angle frequency preferably less than 0.1 deg/lb (-20 dB)
Figure 8-15 Final development of PIO criteria [Gibson, 1995a]
1. Level 1, 2 and 3 boundaries represent historical data.
2. Undesirable residual high order characteristics exist within the Level 1 region near the low frequency boundary limit.
3. Best design practice for freedom from linear high order PIO requires the more stringent Level 1* gain, phase rate and frequency limits.
Figure 8-16 Tornado in retrospect against author’s later criteria

Figure 8-17 Calspan Learjet PIO phase and gain experiment [Gibson 1991b]
Figure 8-18 Concept of "effective time delay"

Pitch acceleration peak time 0.6 sec.

Figure 8-19 Example of severe tracking PIO with low acceleration gain, Case 4D from [Boothe et al., 1974]. Pilot ratings 8 and 9 at 6-9 lbs/g stick force. Compare with Figure 8-17 Case 3.
(a) Landing approach case from [Smith, 1978]

Addition of a pre-filter with a small time constant (0.25 sec) transforms the poor handling of Case 3-1 to catastrophic in Case 3-3.

(b) Fighter tracking case from [Boothe et al., 1974].

Addition of a pre-filter with a larger time constant (0.3 sec) than in (a) to the basic Case 8A transforms the handling from poor to excellent in case 8D.

Figure 8-20 Examples of non-uniform effects of added pre-filter lag.

See also Figures 7-30, 8-4

Linear pilot induced attitude oscillations
Figure 8-21 Learjet roll PIO experiment data [Gibson, 1991b]
Roll control PIO tendencies can be identified by measures similar to pitch control criteria. Simple lag pre-filters produce the same adverse PIO effects.
Chapter 9
Non-linear PIO, and other miscellaneous PIO types. A pilot model for PIO.

9.1 Introduction

Chapter 8 dealt with the problem of attitude PIO in linear systems, usually associated with linear control laws containing non-classical and higher order mode characteristics, deliberately or inadvertently. However, non-linearities have also played a major role in many PIO events, principally the result of software or control actuator hardware rate limits. Sometimes these cause a major PIO divergence after beginning in the linear regime, and sometimes an immediate full scale PIO with no initial linear phase occurs. Although examples have occurred for over four decades, it is only recently that the problem has been addressed as a major issue. It is now more widely understood and means have become available to alleviate the effects.

Most PIOs have occurred in the pitch axis of control, with a small number of examples also in the control of roll angle. However, any loop closed by a pilot is potentially liable to PIO, with dynamics that are either naturally liable to unstable piloted control or that are made so by improper design. Such loops include height or flight path control, heading angle control, nosewheel steering, and flight director following, all of which have been discussed in Gibson [1995a, 1995b].

A PIO is normally a consequence of voluntary pilot control activity, but another form of oscillation phenomenon can occur that depends on the presence of an entirely involuntary contribution from the pilot. This is biomechanical PIO, in which the coupled pilot inputs are forced mechanically by the motions of the aircraft. This is truly more properly identified as an APC or aircraft-pilot coupling than as PIO, but the cause can lie within improper control law design and needs to be addressed along with other PIO issues.

Despite the progress over the years in pilot modelling, even the advanced optimal pilot models have never actually predicted a PIO event before it happens. In Chapter 8 the concept of the synchronous gain-only pilot was shown to provide a powerful tool in PIO analysis, permitting attention to be directed to the aircraft dynamics alone to find and eliminate PIO tendencies. The same concept is applicable to the PIO types discussed in this Chapter. However, the author has found that the pilot behaviour ranges in practice from sinusoidal inputs tied to the attitude response to completely non-linear switched or relay-like behaviour. Nevertheless it is always synchronised directly to features of the attitude response.

9.2 Non-linear PIO problems

The non-linear effects of actuator and system saturation have been the subject of major studies recently, as noted in §8.1. Many other examples of PIO influenced by such non-linearity are to be found in Klyde et al [1995] and Anon [1997]. Such PIO was experienced long before the advent of aircraft with high order control laws. There is almost no limit to the range of possibilities for adverse saturation effects - "a confounding variety of input-amplitude-sensitive effective vehicle dynamics" [Klyde, ibid].

Prior to both SAAB Gripen PIO accidents, it was actually known from piloted simulation that large and rapid control activity could lead to loss of control due to actuator rate limiting. It was
assumed that "pilots would not fly like that". Pilot knowledge of this and the provision of a cockpit voice warning of "excessive activity" could not prevent it even though the pilot was the same in both events. In very early Tornado test flying with the control law that led to the accident discussed above, a pilot in another Panavia partner company experienced two major PIOs involving full amplitude stick inputs for some 11 cycles or more in each case. One of these is shown in Gibson [1995b]. These Gripen and Tornado PIOs were noteworthy examples of the "explosive" type due to actuation saturation effects. The oscillations did not diverge from a limited linear beginning but were fully saturated immediately with maximum stick inputs.

It is essential to grasp that pilots have absolutely no training to cope with severe PIO, and they have never experienced anything remotely similar. It is extremely dangerous to assume that they would not, or even should not, employ very large control inputs when plunged into such a situation, or to expect them to be able to limit their inputs to some arbitrary extent. Because such events are traumatic and bizarre and because pilots invariably believe that there has been a system failure, they will apply whatever control seems to be necessary at the time, even though the best input is none at all. In the Figure 8-7 PIO, which began in the linear dynamic regime but was subsequently deeply locked in by actuator rate saturation, the pilot was not even aware that he was in a PIO but thought the aircraft had suffered a large uncommanded pitch-up.

Figure 9-1, from Klyde et al [1995], shows a take off PIO in the NASA digital control F-8 in an experiment with selectable added time delays [Berry et al 1980], conducted after a landing PIO of the early Space Shuttle. After a successful landing, a go-around was begun. Because of the ground speed limit of the nosewheel, it had to be kept clear of the ground. This was a difficult task because of very limited tail ground clearance, and a small divergent PIO began, initially with linear actuator dynamics. The tail struck the ground, which automatically de-activated the stability augmentation system (SAS) but left the selected time delay active. The PIO then rapidly diverged to maximum amplitude, becoming sustained by the rate limit saturation. Recovery was made only after the pilot switched off the time delay and switched the SAS back on. Letting go of the stick would have stopped the oscillation at once, but with a major loss of control very close to the ground and the belief that the system had failed, the pilot was unable to contemplate this, and probably could not have done even if he had understood the cause.

A roll PIO in the landing flare of an early Tornado prototype version is shown in Figure 9-2. This was caused by the high stick to control surface gain. The pilot applied inputs of up to 80% of full stick, demanding a maximum rate of about 240 deg/sec from spoiler actuators with an actual maximum rate of 100 deg/sec. The solution was to modify the forward path command structure and filter arrangements to reduce the higher frequency control amplitudes, which effectively eliminated the likelihood of inducing rate saturation. An additional non-linear feature of electrically signalled spoilers used for roll control is that, in an oscillatory control situation with rapid stick reversals, the spoiler being retracted is still in motion when the other begins to be extended. During the overlap period the roll control power is doubled from the nominal case, in which only one spoiler or the other should be extended. However, the extent to which this may have contributed was not studied.

**Open loop or series rate limiting**

By series rate limiting, the author means a rate limit that forms one of a series of elements in a flight control system which is not included within a closed loop of the system. The conventional aircraft of the past with power actuated controls, but with little or no augmentation, have always been subject to series rate limits of the control surfaces. The pilot's stick provided a close indication of the actual surface position. When an actuator rate limit was reached, the flow control
valve would be bottomed, preventing the stick from being moved at a higher rate and clearly signalling the presence of saturation. Hence it was physically impossible for a pilot to apply surface demands grossly in excess of what was possible. With stable dynamics, the aircraft response could not diverge unexpectedly and always remained clearly under the pilot's control.

Electrically signalled controls removed this connection. It became possible to apply large stick movements unrestricted by the actuator limitations, causing the effect sketched in Figure 9-3(a). The output saturation reduces the response amplitude and increases its phase lag relative to the limiting element input. Since the output rate has a maximum value, the effect changes with increasing input amplitude from unnoticeable as the rate limit is just invoked to gross at a very large relative input demand. A complete treatment of rate limiting is given in Duda [1997a].

The sketch in Figure 9-3(b) is typical of a stick command rate limit applied to prevent extremely rapid inputs to a manoeuvre demand system, but which might still permit a maximum neutral-to-full stick command in one quarter of a second. Since it is physically difficult to exceed such rates, the saturation regime will in practice have little influence on the response characteristics. As sketched, there is little change to the "PIO gain" though the "PIO frequency" is reduced. If the linear dynamics have been well designed, this will be an insignificant effect. However, the first Gripen PIO accident followed from saturation of just such rate limiters. These were placed only in the proportional signal paths, which when saturated left another heavily lagged path as the primary control input. A divergent PIO ensued when the pilot applied large input reversals.

At the other extreme, a PIO of ±4 degrees pitch attitude lasting for 16 seconds to the touch down point occurred in the first flight landing of the unaugmented NASA X-15 [Matranga, 1961]. This was enabled by the very low tailplane actuator rate limit of 15 deg/sec coupled with use of the small motion side stick. With no restriction on the stick rate from physical feedback of the actuator position, allowing readily applied maximum stick inputs, the response deeply saturated the actuator rate limit resulting in uncontrollable PIO. The solution was to increase the actuator maximum rate to 25 deg/sec, to use only the less sensitive centre stick for landing, to increase the stick feel gradient by 30%, all of which reduced the physical possibility of operation in the saturation regime, and to fly only if the pitch SAS damper was operative.

The X-15 PIO was analysed in Ashkenas et al [1964] by the describing function method, and was revisited in Klyde et al [1995] which gives the PIO flight record. A computed estimate of any such non-linear frequency response can be made easily by running a batch of time response calculations for a range of stick input amplitudes over a desired spread of frequency, extracting the amplitude ratio and phase of the output/input relationship automatically.

The actuator acceleration limit effect shown in Figure 9-3(a) is little discussed in the literature but is of profound importance. It is created by the dynamics of the servo control valve or device that converts the electrical demand signal into the output hydraulic flow. Unlike the simple assumptions used in most rate limit analyses, the actuator cannot reverse direction instantly. The servo device has to first decelerate the actuator and then accelerate it by a physical movement of the flow control valve from one end of its stroke to the other. Since the valve displacement controls the actuator rate, its own velocity controls the acceleration of the actuator. Unless this is high enough, the output lag in a saturated oscillation is significantly increased and the amplitude ratio is also increased. As an example, a seemingly rapid servo valve movement from centre to its maximum in one fourth of a second represents an acceleration of 4000 deg/sec² in an actuator with a maximum rate of 100 deg/sec., but introduces an additional 5 degrees of control surface position error in a maximum rate reversal through a total rate change of 200 deg/sec.

Non-linear and miscellaneous PIO. A pilot model for PIO

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Closed loop rate limiting

A closed loop rate limiter, unlike the series or open loop type discussed above, lies within the closed loop of the augmentation system, and therefore its saturation alters the aircraft response dynamics directly. It may be an inherent actuator limit, a limiter placed in a feedback signal path, or a limiter placed in front of an actuator to prevent its saturation.

An extreme example of gross amplitude-dependent changes in the response dynamics was reported by Fielding [1984]. Non-linear analysis by the describing function method and by a full non-linear simulation was performed by Fielding [1982] from which Figure 9-4 is derived. Saturation of both rate and acceleration limits of the tailplane actuators caused large response gain and phase changes in the actuator performance. These resulted in reduced damping of the augmented aircraft closed loop system, causing a large attitude resonance which peaked at phase angles around -180 degrees, the PIO frequency. A relatively small increase of the pilot's input activity caused a jump from a satisfactory linear behaviour to a considerably larger PIO amplitude ratio and much lower PIO frequency. In this case the response amplitude increased by a factor of eight for only a doubling of the pilot's input commands. This was seriously destabilising to the pilot-aircraft closed loop and a major PIO occurred, though only after after a considerable period of test flying because of several pre-conditions that had to be simultaneously satisfied before the PIO could be sustained.

As it was not feasible to consider a major re-design of the actuator, the solution was found in modifying the control laws. A major element in this was a feedback of actuator rate to back off control commands that were large and rapid enough to saturate the rate limit. The response dynamics were effectively linearised and the problem was completely resolved.

The "cliff edge" nature of such a response is typical of many PIO events in highly augmented fly by wire aircraft. If the closed loop system is very PIO-prone, a stick command rate limiter can prevent its saturation, but if sufficient to achieve this aim it too will degrade the handling. At the worst, a closed loop rate limit could cause instability of the aircraft system alone. Once excited by a large enough pilot input, this instability would then continue with the stick held fixed. The only satisfactory solution is to eliminate significant closed loop effects or to reduce them as far as possible, and to incorporate the minimum necessary forward path command rate limits to avoid invoking them.

Design criteria for non-linear responses

To the author's knowledge, all criteria aimed at preventing adverse high order effects up to quite recently have been based on linear response characteristics. Non-linear problems have had to be solved on an ad hoc basis after the event, e.g. Fielding [1984] or Ashkenas et al [1964].

The author at one time considered applying his linear phase rate measures to non-linear responses. This proved to be difficult, requiring much non-linear time response simulation etc., and it was eventually realised that the idea was actually inappropriate. Phase rate or phase delay is a valid measure as a symptom of adverse dynamics in normal linear control. Severely rate saturated dynamics do not affect the handling in normal circumstances and are not involved in control tasks. These dynamics are themselves the root cause of severe PIO and appear "out of the blue".

After developing the linear system gain limits of Figure 8-15 in 1993 [Gibson 1995a], the author concluded that they would also provide some protection from rate saturation PIO if the sat-
urated attitude frequency response with the largest practical oscillatory stick input satisfied the same gain boundaries. In practice it would probably be acceptable for this response to stray into the "Level 2" range, since the associated dynamics at smaller input amplitudes would certainly satisfy the Level 1 boundary and the pilot would not be precipitated into a maximum amplitude PIO. An alternative is to apply an absolute amplitude limit of some ±2° attitude response at the PIO frequency at the same maximum stick oscillation input, as discussed in §8.4.2. This result is independent of the stick force applied by the pilot.

The "Open Loop Onset Point" or OLOP criterion [Duda, 1995, 1997b] is currently under study at the DLR, Braunschweig, as part of their comprehensive investigations into the effects of non-linear control saturation effects. Shown in Figure 9-5, the OLOP is the location on the linear frequency response Nichols plot of the point at which a rate limit onset is first reached with the maximum possible pilot input amplitude (implying stop-to-stop control). The criterion estimates the likelihood of a saturation PIO from the position of the OLOP, for a range of pilot gains defined as low to high according to the resulting crossover phase angle (see Figures 5-3 to 5-5). The synchronous gain-only pilot model for a fully developed PIO is assumed. The pilot gain is that which causes the pilot plus linear aircraft closed loop frequency response $Y_P$ to cross the 0dB line at the chosen crossover phase angle. This is -110 degrees for a low gain pilot and -160 degrees for a high gain pilot.

Originally evolved for pitch axis dynamics, the criterion has been extended to the roll axis also [Duda and Duus, 1997]. It has been validated by extensive ground based flight simulation in the absence of data from in-flight saturation PIO. The criterion predicts that saturation PIO is very likely if the OLOP reaches the boundary with a low gain pilot, while PIO is very unlikely in the case of a high gain pilot. A high success rate has been found, although the writers acknowledge that it has yet to be proven over a wide spectrum of different aircraft types to develop exactly defined application rules. They state that experience and engineering judgement is necessary for the application of the OLOP criterion. This is of course true for most practical criteria.

It has been found [ibid] that the PIO potential due to rate saturation in the feedback loop, that is closed loop rate limiting, is even higher than the writers had expected. On the other hand, they found that series rate saturation in the forward path, or open loop rate limiting, is less critical. They speculate that this may be because the pilot is able to understand the change in response dynamics that occurs more easily in this case. Series rate limiters create lesser changes, unless extreme as in the NASA X-15 example above, and the conclusion is certainly in accord with the experience of the author and his colleagues. This has led the writers to suggest that the OLOP boundary should be considered more as a transition area with a ±3 dB width than as a simple line. An extra 3 dB margin should be allowed for closed loop rate limiting, and a 3 dB relaxation could be allowed for series rate limiting (Figure 9-5).

The OLOP criterion has all the hallmarks of the present author's methodology for practical design guidance. It is based on a real and physically obvious connection with the pilot's experience, and is not couched in obscure equivalences or indeed in mathematical abstractions. It is extremely simple to apply, requiring in practice only a boundary template to be moved vertically on the Nichols plot until it crosses the linear response at the desired crossover phase angle. Applied to Figure 9-4, it would certainly have predicted PIO for only moderately large control input amplitudes, but the dramatic changes in dynamics visible in the figure would have predicted this on their own had they been available before the event. The suggested upper margin indicates that an open loop rate limiter, e.g. at the stick output, is unlikely to pose a problem to the sketched response, but the author counsels caution in using this extra margin.

Non-linear and miscellaneous PIO. A pilot model for PIO 181
Both the OLOP and the author's criteria methods apply to the aircraft dynamics alone, implicitly assume a pure gain pilot acting synchronously but do not actually require a pilot model to be present in the calculations, and set an upper limit at a single point in the frequency response. If only the high gain pilot assumption is considered in the OLOP criterion, a necessity for proper design assumptions and flight clearance, then both criteria amount to requiring that the saturation effects for very large stick input amplitudes are postponed to relatively high frequency parts of the response closely approaching the PIO region of -180° phase angle, or even better to a higher frequency.

For all practical purposes, both criteria show that the response must ideally remain virtually linear, or quasi-linear with only slow or slight changes in dynamics, in the normal range of frequencies used by the pilot at all possible amplitudes. This frequency range extends essentially up to the -180 degree phase angle point.

Hardware design considerations

It is in principle possible to satisfy the above non-linear system criteria if the actuator design has sufficient rate and acceleration capability. Input signal or series rate limiters are only necessary if the actuation system cannot provide sufficiently high rates to meet all manoeuvring needs without significant saturation. Low rates may be due to a low hydraulic pump flow delivery capability, as in the Gripen where low weight was an important consideration. In any case, it is possible to use actuators with much higher rate capability than can be sustained by the pumps and to use hydraulic accumulators to supply extra flow in the very short term. This permits the rapid application of a large control input without exhausting the supply, and the accumulator can recharge before another large control input is required.

The choice of desirable maximum rates can be confused by misunderstanding the implication of the units of rate. The only important parameter is how long it takes for a control to be applied. If a minimum time of 0.2 seconds is desired, the corresponding rate for roll control by a differential tailplane system of ±5 degrees authority is 25 deg/sec (although this would be inadequate for the tailplane's symmetrical pitch control function with perhaps a total travel of 30 degrees). For a spoiler system with 50 degrees deflection, the equivalent rate is 250 deg/sec. Allowing for the differing control surface sizes and hinge moments, the hydraulic power requirements would be roughly similar despite the 10 to 1 range of angular rates.

It is possible to specify the desired actuator maximum rates by application of the principles for effective linearity in the preceding section. Knowing the desired OLOP frequency, the control surface peak rate at that frequency can be determined for maximum amplitude stick inputs. This is then the minimum value of the desired maximum rate. The difficulty with this is that the actuator design will most probably have to be determined before the control law design is sufficiently developed to be sure of the OLOP frequency at critical flight conditions. However, the minimum time of 0.2 seconds to reach full deflection discussed above permits a full cycle of maximum amplitude oscillatory control travel in about 1.25 seconds if the maximum rate is just reached at the peak, or in 0.8 seconds at maximum rate provided there is no serious acceleration limiting. It is difficult to imagine that this would not be sufficient to avoid non-linear PIO.

The OLOP criterion suggests a fall-back position if the optimum actuator rates are impossible to obtain. The use of a series type stick command rate limiter allows a much lower OLOP to satisfy the upper limit of the criterion boundary. This can be set at a rate just sufficiently lower than the actuator rate limit to maintain linear actuator behaviour, which is the most desirable factor.
Even sufficient actuator rates will not prevent problems if the control laws are poorly designed. Walker et al [1982] relate how a North American F/A-18 prototype in its initial sea trials experienced lateral PIO with rate saturation despite an aileron maximum rate of 100 deg/sec. This was in the general context of an aircraft that was "plagued with bobble, overcontrol tendencies and PIOs, especially if the pilot used a high gain in a tightly controlled task", and that ultimately required dozens of control law changes before achieving satisfactory handling.

**Recent developments towards a software solution**

While stable aircraft have demonstrated severe rate-limited PIO, in the case of highly unstable combat aircraft it is even more essential to prevent rate limit degradation of the closed loop, as all control of the aircraft may be lost. Although it is highly desirable to achieve this aim by provision of adequate control actuator rate capability, it can happen that this is not entirely practical. Recently, rate limiter software algorithms that make the output motion reverse direction at the same time as the input, even when fully rate saturated, have been tested. This eliminates the additional phase lag that is so damaging, though the output gain attenuation remains.

Discussion of analysis and flight experiments is given in e.g. Martin and Buchholz [1995], Chalk [1995], Rundqvist and Hillgren [1996], and Hanke [1988]. Jensen et al [1996] describe the successful development and flight testing of the SAAB rate limiting algorithm in the JAS 39 Gripen, the first to be applied operationally. Two prototypes of this highly unstable design had suffered catastrophic PIO events due to the severe rate limiting necessitated by low actuation rates. Although there is a detectable influence on unsaturated handling qualities, typically reduced responsiveness or some non-linear effects, the handling of the Gripen was actually enhanced by the smoother response with the new algorithm installed.

The current algorithms essentially remove the phase shift normally caused by a rate limiter, but they do not eliminate the reduction in gain. For this reason they are especially suitable for stick command rate limiting, preventing rate limiting in the more vulnerable closed loop parts of the system, although they can be used here if necessary. The forerunner of the algorithms discussed above [Fielding, 1984] achieved its aim by preventing rate saturation in the first place. The design relief afforded by these algorithms should not, in the author's view, be taken as justification for adopting low actuation rates at the beginning of a project design, after which an increase might be very difficult to introduce. Only at much later stages of the control system design, after a thorough investigation of the non-linear response characteristics, could it be determined that the original decision for low can be sustained without some degradation of the handling.

**9.3 Height- and flight path-related PIO**

Most PIO events have been in pitch and roll attitude. Other modes are sometimes involved. Klyde et al [1995] shows the record of the Boeing Stratocruiser accident in 1954 in a ground radar controlled approach (GCA) when the pilot entered a divergent PIO at phugoid frequency. The loss of flight path stability led to a fatal impact from a final descent at 1500 feet per minute from 500 foot altitude. It is not clear whether the phugoid mode was itself unstable, but the problem was certainly exacerbated by inability to relate the available cues to the actual flight path. It is possible that pitot-static system lags caused vertical speed indication errors. Such an event is now probably impossible with improved instrumentation and guidance systems.

Stable height control through pitch inputs usually requires a dominant inner attitude loop, with much weaker height-related outer loops, as was discussed in §7.1 and shown in Figure 7-1.
Novice pilots sometimes break into a height PIO in landing flare attempts if they fail to look ahead to obtain a strong attitude cue. Even experienced military pilots occasionally suffer PIO in close formation flight or when flight refuelling because the very small errors that can be tolerated may lead to the use of high closed loop path gains without sufficient attitude stabilisation.

Reference to Figure 7-24 shows that the flight path angle frequency response of an aircraft is much more highly disposed to enabling a PIO than the corresponding pitch attitude response, to which it is related by the $T_m$ lag in Eq. 3.7. Its phase angle passes through -180° at the short period natural frequency, allowing a pilot-aircraft closed loop instability if the pilot gain is sufficient. At any given phase angle the flight path angle amplitude is much higher than that of pitch attitude, and its frequency is much lower. These features promote the potential for very large amplitude PIO motions to occur. In effect, it has the dynamics of pitch attitude with a considerable high order lag, and is subject to the same PIO problems when controlled as a single loop.

The problem intensifies if the pilot attempts to control flight path displacement. Since the path angle represents a vertical velocity, and path displacement is its integral, another -90° phase angle is added. The entire path displacement frequency response is consequently located at phase angles more negative than -180°, closely resembling the $K/s^2$ acceleration response that is so hard for pilots to cope with in a closed loop manner.

Closed single loop control of any form of height error by pitch control alone is very difficult. Unlike control by a machine, the pilot distributes attention somewhat discontinuously between the attitude and path loops. Pilots develop a strategy that certainly contains significant open loop and precognitive control methods as well as the highly developed attitude inner loop. When these are not employed, a PIO will result.

Such problems occur only in control of flight path by pitch axis demands. Control of path angle or height level by the use of power adjustment is more stable. The cues provided by visual approach glide slope indicators or by an altimeter or vertical speed indicator change slowly, and very precise path control is not required except in carrier landing. In this task, most aircraft are operated well on the back of the drag curve where thrust control must be used.

In the absence of flight recorded data when a PIO has been reported, the possibility of a PIO related to the control of path angle or height should always be considered. There is a substantial difference in the likely frequency of attitude and path PIO. If the frequency can be identified, oscillations at or below the short period natural frequency are likely to be path-related. If the frequency is substantially higher than this then the PIO will probably be in attitude, positively identified by the stick phasing relative to the rate zero crossings.

**Shuttle orbiter PIO**

A PIO occurred in a Shuttle Orbiter in gliding flight at an altitude of 6 km. The pilot attempted to zero a flight path error using the flight path angle symbol in the head up display director [Twisdale et al 1984]. The symbol was effectively a substitute for attitude around which the pilot closed a loop. The author found that the generic pitch attitude PIO features in Figure 8-13 are equally applicable to this closed loop path control problem [Gibson, 1995b]. The PIO frequency of 2 radians/second was the same as the calculated -180 degree phase angle frequency of the path angle response, shown in Figure 9-6. There was also an extremely high average phase rate value, larger than usual because the normal attitude to path angle lag was augmented by the extra lag created by the uncompensated pitch rate demand system (Chapter 4). These fea-
tures led to the closed loop instability, and the criteria would in fact have predicted the PIO successfully if they had been in existence at the time.

Indication of the velocity vector, the path along which the aircraft is travelling, is common in military head up displays and is now becoming more common in transport aircraft. It is most useful for low altitude high speed flight and for glide slope control. If used as the only cue for landing flare, the attitude to path lag effect can cause overcontrol or ballooning, which could initiate a PIO. It is usual to "quicken" the symbol by phase advancing it with a pitch rate term to remove the apparent flight path lag. The result is to convert the displayed dynamics to a \( K/s \) like response, allowing flight path changes to be made with great precision.

Low altitude high speed simulation experiment

Figure 9-7 illustrates results from a ground-based simulation experiment by A’Harrah [1964]. A “g seat” (3-65 metre vertical travel, ±6g), thoroughly validated by cross comparison with flight experience, was used to simulate low altitude high speed flight. In the simulation, the short period frequency and damping were varied while the parameter \( T_{\alpha} \) was kept constant. (The results have been re-plotted here by the author in the format of Figure 7-12.) Boundaries at which PIO was found were established for different values of stick force per g. Critics argued that these results were invalid, because some of the relevant combinations of short period frequency and damping could not exist at the fixed value of \( T_{\alpha} \), and so the "linear attitude PIO" boundary discussed in §8.3 had been improperly crossed.

The author, taking a different view based on the information given in the reference, was able to explain the apparent contradictions by showing that the PIOs were in flight path. Attitude was displayed only on a standard head-down attitude indicator "ball" which could not have been of much use for precision attitude control, and no attitude task was defined. The lack of pilot access to attitude cues was further supported by the fact that good handling was reported at short period frequencies up to 2 Hz, at which in-flight simulation experience shows poor attitude precision but excellent flight path control. The prescribed task was to maintain a 60 metre ground clearance over rolling desert terrain, represented by a cathode ray tube line display showing height error. Vertical velocity (effectively flight path angle) was also indicated to provide some necessary phase advance cue. Possibly because the terrain following task was of relatively low bandwidth, no PIO was encountered in that task. However, PIO occurred when very tight height control was assessed, representing for example close formation flying, etc.

The frequency responses for attitude and flight path of Case "A" on the 5 lb/g PIO boundary in Figure 9-7 show that in fact the closed loop margins for attitude tracking were very large, even at 1 lb/g though the point is far beyond the suggested PIO boundary for the lower stick force. Unlike other examples of linear PIO boundary violation, where the short period frequency and damping were both usually low, they were not very low in this experiment. The -180° phase angle crossing took place at a large attenuation and satisfactory frequency that satisfy the author's PIO criteria. However, the flight path angle response shows typical features leading to PIO, while the path height response (the primary task requirement) has unstable margins. Hence there is no doubt that the reported PIOs were in flight path. Unfortunately, no records are available for study, but in conversation with the author, A’Harrah agreed that this was the most likely explanation.

Influence of pilot location

The phase advance provided to a pilot situated a long distance ahead of the centre of gravity can
substantially counter the effects of the sluggish response typical of many large transport types as noted in Figure 7-25 (§7.5.3). In the Shuttle Orbiter, the cockpit is about three metres aft of the centre of instantaneous rotation, causing a significant delay before the pilot receives a physical indication of a change in flight path, either visually or by way of a normal acceleration, after applying a pitch control input. Its PRD control law was designed to produce little pitch rate overshoot, as had been specified. In the landing flare its flight path time delay was about three seconds with one second attitude overshoot.

These sluggish dynamics (§4.3) make it extremely difficult to land with precision, and indeed they caused a PIO on Flight 5 of the Approach and Landing Test vehicle prototype [Powers, 1985]. The aircraft requires extremely intensive pilot training to ensure safe landings. Automatic landings have almost never been attempted in case a failure forced the pilot to take over control at a late stage. Research was conducted to improve the control laws [Gilbert, 1983]. These introduced significant pitch rate overshoot to quicken the angle of attack response. Although the proposed laws were found to be satisfactory by conventionally trained pilots, the intensive investment in training the Shuttle pilots led to the decision to retain the original laws.

9.4 Lateral-directional PIO

The possibility of pilot/aircraft closed loop roll angle PIO in a conventional aircraft with low Dutch roll damping and proverse yaw roll control was identified very many years ago, associated in principle with \( (\omega_p / \omega_d)^2 \geq 1 \) [e.g. Ashkenas et al., 1962]. This is the ratio of the natural frequencies in the roll numerator and Dutch roll denominator of the roll angle to roll control input transfer function (§3.3.1, Eq. 3.10). Proverse yaw means that when aileron is applied to roll right, the nose yaws to the right also, a feature likely to be found with roll control by differential tail whose asymmetric pressure fields applies a sideforce to the fin. Adverse yaw, where \( (\omega_p / \omega_d)^2 < 1 \), is usually found with conventional aileron controls and does not cause PIO. An aileron to rudder crossfeed was commonly arranged to obtain a unity ratio, or sometimes more completely to achieve the same frequency and damping in both. This ensured that aileron inputs were well co-ordinated and did not cause yawing and sideslip excursions, since the numerator and Dutch roll denominator in the transfer functions cancelled out, leaving only the spiral and roll modes in the response.

PIO of this type, generally found only at very low levels of dutch roll damping, was more of a nuisance than a threat since it did not usually diverge to dangerous amplitudes. Normally such PIO does not occur in fly by wire aircraft which are invariably well damped, but can become significant if they possess a reversionary control mode with no augmentation. Flight testing of the unaugmented handling of the Tornado, with low natural Dutch roll damping and with differential tail roll control giving strong proverse yaw, yielded many records of such small lateral PIOs at high airspeeds. Typically these were induced at the point of switching from augmented to unaugmented mode and were not a flight safety issue. In studying these flight records, however, the author could find no evidence whatever that the pilot was closing a roll angle loop. In every case, the PIO was in yaw angle.

Figure 9-8 shows such a PIO recorded in the "spin recovery" mode of the FBW Jaguar. The spin recovery mode used only direct electrical signalling from the stick to the control surfaces to enable the pilot to apply the specific control deflections required for spin recovery. Simulation showed that lateral overcontrol was easily excited at high speeds, but was manageable. When the mode was tested in flight, a persistent lateral-directional PIO resulted. This could be kept
small by a delicate touch on the stick, but the potential roll amplitudes were such that this was considered unacceptable. From the coincidence of the stick oscillation peaks with the yaw rate zero crossings, it was clear that a yaw angle loop was involved.

The flight records showed no correlation with roll angle closure, and that the pilot was applying lateral control inputs in anti-phase to the yaw angle peaks. This is identified by the coincidence of the stick peaks and the zero crossings of yaw rate. Frequency response analysis also showed no instability in roll angle loop closure, but there was excellent agreement in both the calculated PIO frequency and pilot gain required in the yaw loop, Figure 9-8 [Gibson, 1995a]. A lateral stick to rudder interconnect was added to minimise the excitation of the Dutch roll, with a 9 dB increase in gain margin and a 90% reduction in the open loop response amplitudes at the Dutch roll frequency. Flight tests showed very much improved and acceptable handling, and as intended there was little Dutch roll in response to lateral control.

The more recent combination of comprehensive digital flight data records and two aircraft types with highly proverse yaw characteristics (the Tornado and Jaguar) provided the opportunity for the author's discovery that the theoretical pilot/roll angle PIO due to low Dutch roll damping and roll controls with proverse yaw did not appear to exist in practice. It is not too surprising that it was not made earlier, since most aircraft exhibit adverse yaw which is stabilising. It is perhaps an object lesson in the need to avoid preconceptions in the struggle to understand pilot induced oscillations of all types.

9.5 Biomechanical coupling PIO

The accelerations of an aircraft in motion, both linear and rotational, exert inertial forces on the pilot's arm and the stick. Biomechanical coupling causes additional control stick inputs as a result of these forces. Deliberate use was made of normal acceleration forces in the early spring and bobweight feel systems, either to increase the stick force per g or in an attempt to maintain more nearly constant stick force per g over a wide speed range. The stick free dynamics in the short period frequency region were altered by such systems, sometimes unfavourably [Neal, 1971], but it was not the intention that the pilot's actual control activity should be modified.

Biomechanical PIO occurs when inertial forces on the pilot and stick cause unwanted and inadvertent control inputs that reinforce and sustain the motions. One form is known as roll ratchettified by a rapid neutrally damped roll oscillation. Another form is aircraft structural mode coupling where airframe vibrations at typically up to 4 Hzr have a similar effect.

The pilot as a mannikin\(^{18}\)

The simplest hypothesis to explain biomechanical PIO is that it is a passive form in which the pilot is not consciously engaged in the closed loop. The pilot's arm mass can be considered as a bobweight mass attached to the stick, responding to local accelerations of the airframe by forcing additional motions of the stick. Because of the high frequency the pilot is unable to prevent the stick motion, which couples with the high frequency dynamics of the aircraft response in an neutrally damped oscillation.

Examples of roll ratchet are given in Hoh et al [1982]. Their frequencies range between about 2 to 3-3 Hz, with roll attitude oscillations of less than ±0.5 degrees despite roll accelerations of up to 100 deg/sec\(^2\). It is suggested in Johnston et al [1987] that roll ratchet with a rigid sidestick is

\(^{18}\)Mannikin - a tailor's dummy
caused by resonance of the neuromuscular system, or neuromuscular peaking, which has a natural frequency typical of ratchet oscillations. However, in a study of the neuromuscular system by van Paassen [1990], it was found that the simple assumption of a lateral acceleration bobweight loop could produce roll ratchet with both rigid and moving sticks. The aircraft high frequency dynamics were found to be of much greater significance than the neuromuscular effects.

In Smith et al [1996], the pilot who experienced roll ratchet in a General Dynamics F-16XL was not aware of any body or arm movement and felt the forces at the arm were low or unchanged. However, the signal records of the quasi-rigid stick showed a ±1.25 lb force oscillation equivalent to a displacement of ±0.015 inches. Analysis using the biomechanical arm, wrist and controller models of Allen et al [1973] led to the conclusion that almost the whole of the arm mass could be attributed as an additional mass at the stick to form the lateral bobweight.

The author has discussed ratchet events with several pilots, none of whom was in any doubt that they were passive elements with no voluntary inputs to the oscillation. In the FBW Jaguar, mild roll ratchet was experienced by one pilot who held the stick lightly, but not by another who held the stick firmly with the arm braced on the knee. When the pilots exchanged their techniques of holding the stick, the first then experienced no ratchet and the second did. In this example the ratchet occurred with the stick near centre.

In the total absence of any known evidence of voluntary pilot control of roll acceleration as a factor in roll ratchet obtained from actual flight events, the author has no difficulty in accepting the passive bobweight hypothesis as the most likely and practical explanation. This hypothesis was used to eliminate the ratchet in the FBW Jaguar, by adding a stick damper and modifying the high frequency dynamics. In the EAP fly by wire aircraft the hypothesis was used in the design process to prevent the problem. In flight the stick proved to be completely insensitive to ratchet despite extremely fierce roll accelerations [Gibson, 1995a]. The hypothesis has also been used successfully in analysis of the Eurofighter 2000 design.

9.5.1 Roll ratchet examples
In most aircraft, the stick and arm mass combination are usually mounted well above the rolling axis, where the rotational acceleration is greatly augmented by the linear acceleration. Roll ratchet cannot occur in conventional aircraft with negligible added dynamics, and although an example of roll ratchet in a McDonnell F-101B from 1958 is given in Klyde et al [1995], it does not seem to have been widely experienced until the more recent fly by wire aircraft. Many of these have experienced the problem due to excessive command gain and high frequency phase lags. The inadvertent stick inputs couple through the lateral feel system dynamics and the higher frequency aircraft roll response dynamics.

Roll ratchet motion can be very brief or it may last for many seconds. If the stick command gain is non-linear, as commonly found in fly by wire systems, the ratchet will usually occur at large roll stick inputs where the gain is high, but it can occur at any stick input if the gain is linear. It never occurs with the stick on its stop because the pilot applies sufficient force to hold it there. As it is induced by roll acceleration, it is most likely to appear at high speeds.

An extreme example of roll ratchet from early test flights of the McDonnell-Douglas F/A-18 is shown in Figure 9-9, from Walker et al, [1982], also in Klyde et al [1995]. The large amplitude of the stick motions is noteworthy considering that the pilot would be trying to hold the stick stationary. Although the roll accelerations reach 300 or 400 deg/sec², the roll angle oscillation
is too small to appear on the record as it is only about a degree. When a similar oscillation was excited and the stick was released, the oscillation ceased, showing clearly that the cause was biomechanical coupling. The installation of an eddy current damper (viscous damping effect) to the roll control circuit removed the problem.

Figure 9-10 shows a different ratchet effect at low speed in early Tornado flight testing. It was driven by an aerodynamic non-linearity of the spoilers giving higher roll control power at small opening angles. This resulted in a roll damping augmentation instability localized within a small range of spoiler angles near their closed position, exacerbated in turn by the inadvertent control inputs generated by the lateral acceleration loop at the pilot's arm. The cause was successfully identified by including a lateral bobweight in the analysis. The solution was achieved by linearizing the spoiler control moments by aerodynamic modifications, by a non-linearSpoiler command, and by a reduction in higher frequency control law lags.

In nearly every case the control laws leading to roll ratchet have had several common features. They had excessive high order lag at higher frequencies, and highly augmented roll damping with high roll rate feedback gains. Since this reduces the maximum roll rate achieved from a given control input, the performance had to be restored by a high forward path command gain. The result was high roll damping with very small roll mode time constants and correspondingly high roll acceleration. Further, the roll command gain was commonly made non-linear with reduced gain near the stick centre to provide more acceptable sensitivity, with increased gain towards full stick travel where precise and sensitive adjustments were not required. This resulted in extremely high roll acceleration feedback with the stick held at larger deflections. The roll ratchet has sometimes been alleviated by the addition of a stick command lag pre-filter, but this introduces high order problems instead as noted in Hoh et al [1982].

In fact, very high roll acceleration is counter-productive to effective agility because it does not allow a pilot to control roll angle with precision in combat manoeuvres, e.g. Drajeske et al [1990]. This is particularly so with the typical cockpit location above the rolling axis, subjecting the pilot to severe lateral acceleration particularly at head level. On the other hand, high roll damping augmentation is very desirable for pilot comfort in heavy turbulence. The means to provide the required levels of roll acceleration for pilot comfort and roll damping for turbulence suppression without the introduction of roll ratchet will be discussed in Chapter 10.

9.5.2 Airframe structural mode coupling
Another type of passive coupling occurs when an airframe aeroelastic mode vibrates the cockpit sufficiently to cause inadvertent pilot arm and stick inputs, resulting in control surface deflections which then amplify the airframe motions. This problem is akin to the common servoelastic instability where the FCS sensors detect the airframe vibrations and feed them back into the loop by the surface motions. The usual solution of notch filters in the sensor paths would apply equally to the stick command path in a passive pilot coupling. Although pre-flight ground resonance tests usually include the stick outputs, the pilot's presence is not accounted for and such problems do not appear to have been predicted. Much early work is referenced in Jex [1971], including studies of manual control of the Saturn booster rocket with structural mode excitation by inertial forces from the deflection of the main engine mass.

Norton [1995] lists examples of aeroelastic coupling in the Lockheed YF-12A, the General Dynamics F-111, the Rutan around-the-world Voyager, the Bell-Boeing tilt wing V-22 Osprey, and the McDonnell Douglas C-17A heavy military transport. Many modes were involved in-
cluding engine nacelle pitching, fin fore-and-aft bending, horizontal stabiliser yaw mode, symmetric and anti-symmetric wing chordwise bending modes, landing gear oscillatory mode, coupled wing and fuselage vertical bending, underwing heavy store oscillation, and fuselage vertical bending. Oscillation frequencies were found to up to 4 Hz. Limb-bobweight coupling resulting from these modes caused inadvertent inputs to roll, pitch and thrust controllers. Yaw coupling due to fuel sloshing occurred in the Cessna T-37A, Boeing KC-135A and McDonnell Douglas KC-10 when attempts were made to damp Dutch roll oscillations with rudder inputs.

It is all too easy in fly-by-wire aircraft types to incorporate high forward path command gains which exacerbate the tendency. These gains should be minimised at higher frequencies to avoid high order rigid-body PIO in any case, but large flexible airframes add another dimension to be accounted for. The design of the pilot's controls also becomes an issue, such as the inertial coupling with the centre stick used in the C-17A [Iloputaife, 1996]. The conventional transport wheel control by contrast is insensitive to lateral accelerations. The typical combat aircraft stick, can usually be adequately balanced against vertical and longitudinal accelerations, but not to any extent against the lateral acceleration associated with roll initiation or structure vibrations.

Of the above examples the YF-12A case was not truly passive. The pilot interacted with the vibration mode of the long fuselage nose at about 2½ Hz when trying to exert tight attitude control for in-flight refuelling, producing a limited amplitude PIO [Klyde et al 1995].

9.6 Modelling pilot behaviour in PIO events

In current literature, e.g. Perhinschi [1998], Innocenti et al [1998], Hess [1998], there is a renewal of interest in pilot modelling on the basis that it is essential for the prediction of PIO, or at the least to understand the problem better. Another subject of growing concern in some quarters is to discover suitable flight test techniques to uncover PIO tendencies and to establish standard "trigger events" that might initiate a PIO [Anon, 1998].

Some of the more esoteric modern pilot models, e.g. the Structural Isomorphic Model of the Man-Machine System; the Algorithmic Linear Optimal Control Model of Man-Machine System [McRuer, 1988]; Hess's Structural Model of the Human Pilot [Hess, 1998, Gibson and Hess, 1997] seem better suited to the post-event matching by human response specialists than as an aid for control law designers. The author's view is simple, has not changed in 20 years and has been reinforced by the recent flood of published PIO flight records. In essentially rigid-body PIO resulting from control system design (control laws, actuation, etc.), regardless of how it has been triggered, once in the PIO the pilot acts in a highly synchronised way with the visual perception of the oscillation. Usually this is linked directly to the attitude response but can also be linked to the motion of a symbol in a pilot's display.

The synchronised pilot

The idea of the simplest possible pilot model acting in anti-phase with the attitude oscillation, first proposed long ago by Ashkenas and McRuer [1964], has been reconfirmed after thorough research in Klyde et al [1995]. In the author's first experience with the problem the pilot did act essentially in that way (Figures 8-7 and 8-9) allowing the author's effort to be concentrated on the dynamics of the aircraft alone and leading to a successful understanding of the basic nature of high order PIO. Nevertheless it has become obvious from studying many PIO records that the pilot does not always behave in the purely attitude-synchronous manner with sinusoidal inputs. In many PIOs the pilot inputs are quite non-linear, but they remain linked to the attitude re-
sponse though synchronised in a different fashion as discussed below [see also Gibson, 1995b].

Figure 9-1 shows a range of pilot inputs from attitude-synchronous distorted by high frequency noise, divergence through decreasing frequency to a rate-limited relay switching, finally decreasing in frequency to a brief attitude-synchronous noise-distorted input in recovery. Although this behaviour appears to be quite different from that in Figure 9-9, the pilot's actions are again clearly controlled by the pitch rate zero crossings. The pilot tracks these very closely as the oscillation period changes while rate saturated, governed by the amplitude of the response. There is a corresponding increase in frequency when the amplitude decreases.

In Figure 9-2 the pilot's inputs are somewhat in the form of flattened sinusoids. Although the fundamental response of the sinusoids is not attitude-synchronous, the end of each flat is clearly signalled by the roll rate zero crossing. It may be noted that the form of the pilot action differs between left stick (positive) and right stick (negative) inputs. There is a more positive control of the former, in which the pilot can apply the full pressure from the arm muscles via the palm of the hand. In right stick inputs, the force can only be applied through the pull of the fingers, and the input peaks are more rounded.

Figure 9-11 shows a roll PIO in the NASA M2-F2 lifting body research aircraft [Smith, 1981]. The record saturates at ±60°/second. The skilled test pilot, attempting to fly with wings level, could not restrain the roll angle to within ±50 degrees. The pilot's input appears to be a square wave with a fundamental response 180° out of phase with the roll rate. In fact the action is that of a pure relay, an action seen in §8.3 and Figure 8-6 which was associated by McRuer [1960] with extreme pilot lead. Such lead is usually considered very difficult to generate, but here there is no doubt that it is easy and instinctive. The pilot simply switches each input reversal at the readily foreseen instant of zero roll rate, i.e. at each roll angle peak.

The variety shown here is quite typical of the inputs to be seen in PIO records. It is highly improbable that an accurate forecast of the exact behaviour in a predicted PIO configuration could ever be made, at least to the extent of being useful in the design process. The nature of the stick force and displacement characteristics will influence these variations [Gibson 1995a, 1995b]. The author has found that a conventional stick with relatively large displacements and forces will tend to be used in a sinusoidal manner with its peaks matched to the attitude response peaks. A stick with shorter travel or lighter forces, which can be moved rapidly to its maximum deflections, is likely to result in a more relay-like action, while probably retaining elements of the nominal sinusoid. A stick with very short travel will almost certainly produce a pure relay action, switched almost instantly between its maximum limits at the rate zero crossing trigger point. This can also be seen in the pitch PIO record of the Lockheed YF-22 [Gibson, 1995b], its stick also having a very small travel.

Simple relay-based pilot models for analytical purposes, triggered by the rate zero crossings, are discussed by Smith [1995] and Chalk [1995]. These models are entirely in accord with the author's views on the need for the simplest approach to the problem. The author recommends that they should be developed to encompass approximations to the pilot inputs seen in Figures 9-1 and 9-2. However, it should not really be necessary in the author's opinion to perform closed loop calculations with pilot models to find out if PIO is likely with non-linear configurations, not least because it is uncertain how much pilot gain should be assumed. The effects of non-linearities do not change the essence of the aircraft-alone response qualities that should be looked for to prevent PIO in just the same manner as for linear systems, and which are actually based on the implicit presumption of the attitude-synchronous pilot.

Non-linear and miscellaneous PIO. A pilot model for PIO
PIO triggers

PIO is usually initiated following what has become widely known as a trigger event. This can take many forms, all creating the need for tightened pilot control. The increased pilot gain employed just before a landing touch down is a classic trigger, which is why so many PIOs have occurred at this point in a flight. The events shown in Figures 8-7, 8-9 and 9-2 were triggered in this way. In Figure 9-1 the trigger for the take off PIO was the combined effects of the tight pilot control needed to avoid a tail strike, and the changed response dynamics after the tail actually did strike. Parrag [1998] discussed the "vast set of circumstances in which PIOs can occur".

Much attention is being given to devising flight tests intended to seek out all possible triggers to give confidence that PIO tendencies will be uncovered, e.g. in Anon [1998]. These tests are intended to force the pilot to use high closed loop gains. A well known test is the offset landing approach that has been employed by many years at Calspan. The approach is conducted parallel to the runway but offset from it laterally, and a last minute correction is started at a point which ensures that it can only be completed virtually at the touchdown point. Many roll and pitch PIOs have been initiated by this test in experimental research flights, e.g. Chalk [1995].

The author considers that reliance on flight test procedures simulating mission tasks to uncover PIO may well give false comfort to the testing authorities. Apparent success may mean only that the critical trigger has been overlooked or not identified. The tests are intended to force high pilot gain, but it cannot be known if even this is high enough. A simple example is described in Klyde et al [1997]. In-flight refuelling was evaluated on a Grumman F-14 using simulated emergency hydraulic power, with very low actuator rates. In preliminary formation flight tests, maintaining height within ±1 foot was very easy, and as a result of this precision performance no difficulty was expected with the actual drogue engagement. The task ended in a violent PIO.

Although this PIO was a complete surprise, it should not have been. A simple ground simulation exercise using coarser inputs would have shown a potential for severe PIO with the easily saturated dynamics, and this should have been enough to prepare for one. So should the two PIOs shown in Klyde [1995] that were experienced in an F-14 dual hydraulic failure evaluation, in which the pilot applied stick forces of ± 50 lbs with a maximum actuator rate of 10 deg/sec. Possibly nobody remembered that, long ago, the first F-14 prototype was destroyed on its third flight following crew ejection in a severe PIO. This developed after pipe fractures in both of the primary hydraulic systems caused the emergency hydraulic power system to be switched on.

The author's view has been expressed before, that PIO occurs because the aircraft dynamics enable it. It behoves control law designers to ensure that all trim changes are faded smoothly, no abrupt control gain changes occur when switching to a different control law configuration, that there are no sharp discontinuities or non-linearities in the response, and that no major change in dynamics occurs with increasing stick input amplitudes. Linearity is all-important, or at least the system should be quasi-linear to the pilot. The repeated experience of the author and his colleagues has been that it is entirely possible to design control laws that can be confidently predicted before flight to prevent both linear and non-linear PIO.

It is of course entirely appropriate to test a combat aircraft, for example, in a wide spectrum of task related manoeuvres. This will determine whether the handling qualities are as good as they need to be, since this is still a partly an art rather than an exact science. With today's technology, there should be no other surprises such as PIO, especially a non-linear type. In past years it was not possible to model the hardware, software and aerodynamics with high accuracy. The only
place where the dynamic performance of the complete aircraft system could be evaluated might well have been in flight testing. Nowadays, the only significant uncertainty lies in the aerodynamic model, and even this can be derived before flight to an astonishingly accurate degree.

The only testing known to the author that can truly guarantee to find a major non-linear response problem is that associated with carefree handling in a combat aircraft, as demonstrated in the EAP research aircraft and currently in the Eurofighter. The intention is that no possible control input or series of inputs can either break the aircraft or cause loss of control, the latter clearly being vital if the airframe is highly unstable. Needless to say, this must be thoroughly worked up in ground simulation before being attempted in flight. The control inputs comprise rapid, full deflections of the stick in pitch, roll, pitch and roll together, rapid stick reversals, large half-cycle motions, and anything the test pilots can think of to catch the system out. Such inputs are unlikely to be employed in any test procedure supposed to simulate a mission task.

In a fly by wire aircraft the hardware and software dominates the behaviour in any case. In the author's opinion, the primary reason why a serious non-linear problem could now be found only after flight trials commence would be that nobody looked for it beforehand.

9.7 Summary

Chapter 9 has further discussed the mechanism of PIO, including the consequences of actuation saturation which has played a major part in most high order PIO. The author's linear PIO gain criterion can be used to assist in the design of control laws that prevent PIO tendencies due to a saturated response, and the OLOP criterion is a further more comprehensively researched approach to this problem. Both criteria assume the implicit presence of the pure gain attitude-synchronised pilot model, with the assumption that the pilot gain is sufficient to excite full amplitude control inputs.

The early concept of the "synchronous pilot" in a PIO, adopted by the author as the fundamental basis for and understanding of high order PIO, is extended to embrace the attitude-synchronised pilot where the pilot's actions are triggered by the response zero rate crossings. The pilot's actions range from linear to non-linear. The conditions for PIO prevention are clearly that it is necessary to allow for maximum pilot gain and control amplitude and sufficient to consider only the dynamics of the aircraft response. They effectively require that linear or sufficiently quasi-linear response characteristics must be provided over the frequency range of direct interest to the pilot, that is up to the point where the linear response phase angle reaches -180 degrees. A major consideration in achieving this goal is the provision of adequate actuator rate capability and especially in the assurance of a high actuator acceleration limit.

Many additional types of loop closures with potential for PIO occur other than the principal attitude loops, and of these height and heading angle control PIO are explained. In each case, similar response dynamics are found and similar solutions can be applied as have been discussed in Chapter 8 for linear attitude PIO.

While the pilot's actions are voluntary in all of these, a further involuntary type of PIO is discussed, the biomechanical PIO. In these the pilot's contribution is confined to the passive provision of an inert arm mass attached to the stick, amplifying the inertial forces applied by motions of the aircraft to the stick. Roll ratchet has been prominent in many fly by wire aircraft, caused by excessive command gain and high order dynamics.
Figure 9-1 PIO during take off of NASA F-8 DFBW (from Klyde et al 1995)

Figure 9-2 Roll PIO due to high order lags and spoiler actuation rate limit (taken from a flight test record)
(a) Non-linear gain and phase effects due to rate and acceleration limiting.

(b) Possible effect of open loop rate limiting on attitude response. Gain loss and phase lag due to rate limit move frequency point 1 to position 2.

Figure 9-3 Gain and phase effects of simple rate limits

Figure 9-4 Example of effects of rate and acceleration limited actuator as a function of the control surface amplitudes, from [Fielding, 1982/1984]. Note the "explosive" growth of the PIO gain with increasing control amplitude.

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Figure 9-5 Application of Open Loop Onset Point (OLOP) criterion with extension to variable margins and use of a moving template. Adapted from [Duda and Duus, 1997].

Figure 6 Analytical identification of Shuttle Orbiter flight path director PIO taken from [Twisdale et al., 1984].

The PIO criterion frequency correctly matches the in-flight PIO frequency of 2 rad/sec. The phase rate criterion predicts strong PIO tendencies.
Original results were claimed to be invalid because of linear PIO boundary violations. Analysis shows that the PIOs were in height, not in attitude, and were valid.

**Gain margin dB**

<table>
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<tr>
<th>SF/g</th>
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<td>5</td>
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<tr>
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<tr>
<td>1</td>
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Attitude and flight path gain margins for various stick force per g values

Flight path angle $\gamma$ [deg]

$\omega_p = 3.2 \text{ rad/sec}$

$\zeta_{sp} = 0.4$

$T_{\theta_2} = 0.31$

$2\zeta_{sp}\omega_p T_{\theta_2} = 0.8$

Response of case A

Figure 9-7 Analysis of a probable height-related PIO from experimental data in [A'Harrah, 1964].

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Figure 9-8 PIO in yaw angle due to roll control inputs in the Jaguar FBW (unaugmented spin recovery mode, from flight records).

Figure 9-9 Example of severe roll ratchet in a prototype McDonnell Douglas F/A-18, taken from [Klyde et al, 1995].
Roll acceleration caused by small amplitude spoiler instability
oscillation caused coupled stick inputs augmenting the instability

Figure 9-10 Small amplitude instability augmented by roll ratchet
(taken from flight records)

Figure 9-11 Roll PIO in M2-F2 lifting body research aircraft
showing relay-type pilot activity
(from Smith, 1981)
10.1 Introduction

Good handling is not just a means for giving enjoyment to the pilot. It performs an essential part in task performance. The combat pilot must devote all possible attention to the management and operation of a variety of complex weapons and defensive systems against an increasingly capable opposition. The ability actually to fly the aircraft with minimum attention to safety and structural limits releases considerable mental capacity to the combat task. The commercial airliner pilot is in a different situation, for most of the time relinquishing direct flight control to an automatic system while attending to the peripheral flight management systems. When direct pilot control must be assumed, it will usually be as the result of system failures that may occur at any point in a flight and in any weather conditions. The lack of continuous hands-on piloting experience can then be a safety hazard in a critical situation, where the provision of precise and instinctive handling greatly reduces the risk.

Pilots (and any passengers) are not the only beneficiaries of good handling. Governments, the owners of combat aircraft, enhance the efficiency of their forces when the less skilled and experienced pilots can fly their aircraft as aggressively as their more skilled colleagues. Manufacturers, properly reluctant to bear higher costs than absolutely necessary, can misjudge the financial penalties that will occur as a result of late modifications that should have been implemented very early. Abzug and Larrabee [1997] give examples where the correction process took several years. Given the increasing incidence of PIO problems in commercial fly by wire aircraft types, their manufacturers are becoming more sensitive to the possibility of litigation if passengers are injured in accidents due to faulty handling that should have been foreseen and eliminated.

Up to the 1940s, aircraft could be tuned to give desirable handling, though sometimes only by expending considerable effort. Ashkenas [1984] recalls the hundreds of hours of flight testing spent developing the control system balance, tab and spring gearing and control circuit of the Vultee Vengeance dive bomber. In the post-1940s era, the handling qualities were dominated by the increasingly poor aerodynamic and inertial properties of aircraft. These could be ameliorated at best by stability augmentation that was sometimes mechanically quite complex, but they could seldom be made perfect even assuming that such a state could be adequately defined. In the rapidly growing sphere of fly by wire technology, in both military and civil applications, the means for good handling resides largely (but certainly not only) in software code. It costs little more in software overheads to write "good" code rather than "bad" or merely careless code, assuming again that the desirable end result is known. The overall effort required to write the software is very large and should not be wasted to any degree on inferior handling code.

The best criteria available may not be fully effective, however, unless the whole management structure of the design process is involved properly, as was noted in Chapter 1. This Chapter describes some methods of applying the author's research results to the design of flight control laws to optimise the handling qualities of fly by wire aircraft. However, no general guidance to suitable control law structures can be given here, because they are closely linked to the individual properties of each aircraft type and because they must be related to a wide variety of different tasks.
10.2 Laying the foundation

Before good handling can be designed, it is necessary to start with a complete definition of the control laws structure and layout that can be shaped to perform all the functions that will be required. All the potential inter-axis cross feed paths must be provided, at relatively small or negligible cost at the beginning even if any turn out to be unnecessary. Incorporating a missing piece of the structure in later flight testing to overcome an inadequacy is expensive, time-consuming, and subject to management resistance even if it is "only software".

Many modern optimal design methods are very powerful, but they are based on linear systems. If their output is simply assumed to be optimal, painful surprises can almost be guaranteed to follow in the real world that contains performance limited actuators and sometimes highly non-linear aerodynamics. Attention is currently focussed on robust computer design methods able to deal with the inevitable discrepancies between aircraft and system realities. A major European study of this was recently completed [Magni et al., 1997], which provides a good overview of the methods available and how they fit into the industrial process. It was concluded that retention of a solid engineering grasp of all connections with the actual aerodynamic and inertial properties of the aircraft is essential in any computerised design method, so that any pathway through the control laws can be followed and related to an individual property.

All aircraft operate in a number of specific flight phases. They must be able to operate unattended by the pilot, even if only briefly, so that the pilot may pay attention to other tasks. It must be possible to trim them easily, but to which parameter needs careful thought. Conventionally, trim is to angle of attack from which follows a looser but adequate trim to airspeed. With most manoeuvre demand systems, angle of attack is not explicitly controlled, and it must be decided where this is acceptable and where it (or a speed substitute) must be introduced. Some demand systems can result in excessive attitude disturbance when flying in turbulence.

Precision flying involves the essentially linear regime, where the pilot is in closed loop control. The characteristics should enable precise control of attitude or flight path to the degree required for any given task without undue mental effort, which should not be greatly increased by the presence of turbulence. This involves the fundamental analytical linear design, to which most handling criteria are directly applicable.

Gross or large amplitude manoeuvring inevitably introduces the difficulties associated with non-linearities of the system or the aerodynamics. In this regime the inter-axis crossfeed paths assume particular importance, since the basic linear design may be inadequate to cope with inertial cross coupling and the non-linear system effects. Design analysis is difficult and success depends heavily upon large scale computer response calculations with all possible maximum control input variations. This is unlikely to be sufficient to uncover the critical combinations of inputs that a pilot may make which can lead to loss of control, and resort to piloted simulation is vital. It is essential that extreme pilot inputs are not ruled out as inadmissible, since it is impossible to predict all the circumstances a pilot may experience in flight.

10.3 The prevention of PIO

As noted earlier, PIO is not a new problem. More than 30 years ago the following check list of basic causes of PIO was given by Ashkenas et al [1964], applicable to almost every eventuality known at the time.
• Incomplete pilot equalisation
• Incomplete training
• Inappropriate transfer of adaptation (i.e. carryover of improper techniques from another aircraft)
• Excessive demands on pilot adaptation
• Required gain, lead or lag lie outside the range of normal capabilities
• Rate of adaptation is too slow to preclude oscillation
• Inadequate capability to cope with system non-linearities
• Limb-manipulator coupling
• Impedance of neuromuscular system (including limb) on control stick or pedals changes feel system dynamics
• Motion-induced limb force feedback (e.g. arm becomes a bobweight)

Though written before the advent of the modern high order PIO, and with no associated design criteria, the list is still useful in directing attention to some elements of control systems that could give rise to similar difficulties.

Following the author's principle that PIO occurs because the response dynamics enable it (§9.6), there must be a specific intention of preventing PIO by design before consideration is given to the general and detailed handling qualities. Many handling qualities criteria in the past failed to address the PIO problem at all, or else they did so only somewhat indirectly. The author's PIO criteria, as discussed in §8.4 and given in Figure 8-15, are directed solely at this high order problem and are separate from the general question of handling qualities.

10.3.1 The control law structure

One question that may exercise the minds of programme managers is that, given prediction of poor or unsatisfactory handling qualities, what is the impact on safety? A subject of current inquiry is how to uncover trigger conditions that might initiate PIO, as discussed in Anon [1997]. The author's view of this (§9.6) is that no guarantee can ever be given that a PIO-prone configuration would not exhibit serious PIO simply because no trigger can be predicted.

It must be assumed that at some time a pilot may use enough gain to enter a PIO in response to some stimulus not yet thought of. If the possibility of serious landing PIO is predicted, it would be exceedingly imprudent to permit flight before a safe control law update can be provided. Such laws have been proven possible in a series of aircraft with handling qualities designed to the author's principles, that is the FBW Jaguar, the EAP, and the Eurofighter 2000.

The author has found that the two common features of high order PIO dynamics are excessive response gain and excessive phase lag at higher frequencies. These are not necessarily found in equal measure. High gain without the introduction of excessive phase lag may cause rate limit saturation of the actuation system, which then creates the dynamic lag and further gain changes that initiate PIO as shown in §9.2. Excessive phase lag, although associated with an actual decrease in the high frequency gain, reduces the critical PIO frequency and simultaneously increases the corresponding response gain associated with PIO as shown in §8.4 and Figure 8-13. Hence the intended control law structure must first be examined for unacceptable elements that can be associated with high order PIO, before beginning the process of achieving the desired handling and response characteristics.
Excess gain

High response gain coupled with large and rapid pilot control inputs can saturate the actuation rate limits, or some other rate limits within the control law, creating potentially disastrous changes in the response dynamics at the worst or oversensitive response at best. If substantially more control angle than is physically available can be easily commanded by the pilot, it is very likely that control problems will result.

Excessive gain characteristics can be identified in the command forward path, defined here as the sum of all paths carrying signals from the pilot's inceptor to the control surface actuation. This can include paths common to the feedback system, of course. At higher frequencies representative of the rapid and large control input reversals that may trigger a non-linear PIO, there is little aircraft response feedback to oppose the control demands, and most of the stick command is sent relatively undiluted to the actuation system. Hence the feedback signals do not need to be considered in this simplified assessment, but the positioning of feedback filtering is highly significant. Feedback stabilisation system phase advance filters may function correctly when placed in the forward path, but they will magnify the command signals unacceptably.

Figure 10-1 shows the typical basis for proportional plus integral control laws (neglecting any feedback path filtering that may be necessary). The integrator provides infinite forward path gain at zero frequency, the means by which the system offers exact following of the steady state manoeuvre demand. Its gain is zero at infinite frequency, but in practical terms it is very small at much lower frequencies, leaving the high frequency gain largely set by the proportional path \( K_p K_q T_q \). Since \( K_p \) represents the demanded response and not a control surface deflection, it cannot be arbitrarily increased to excessive values. \( K_q \) is limited by considerations of closed loop system stability. \( T_q \) does not need to be large because the static gain is provided by the integrator path, and it is is restrained by the same closed loop system stability considerations as \( K_p \). In this example it is unlikely that excessive high frequency forward path gain would be likely, though that should not be taken for granted since \( K_p T_q \) could still be large relative to the available control surface authority.

The simple stability augmentation system shown in Figure 10-2 has no integrator path. In older aircraft, the feedback was implemented through small authority servo actuators, and the conventional mechanical control links between the stick and the control surface actuation had an authority equal to the available surface deflections. Implemented in fly by wire systems, the feedback will usually be given unrestricted authority. The stick command authority must then be increased to supply both the actual surface deflection required to balance the aerodynamic forces and the additional signal to balance the feedback during manoeuvres.

This layout was used for some earlier pitch systems (it is the basis of the Tornado fly by wire controls), and it is still the layout commonly used for roll demand systems. The stick control authority may equal three or more times the conventional full surface deflection available, greatly excessive at high frequency as discussed above. Even if this did not provoke rate saturated PIO, it would be a likely source of the roll ratchet discussed in §9.5.1. Typically a pre-filter would be incorporated to attenuate the high frequency gain.

Another source of high forward path gain is the typical stick demand non-linearity sketched in Figure 10-2, frequently used in roll systems and sometimes in pitch systems. It can greatly increase the possibility of roll ratchet when the stick is held at a large deflection, the oscillation stopping when the stick is returned to centre where the command gain is least.
Excess phase lag

As discussed in Chapter 8, linear high order lag effects are usually extremely obvious in the step input transient acceleration response, appearing as a lag in reaching the peak amplitude which theoretically should be instantaneous. While an acceleration response peak time of up to 0.25 seconds may be acceptable, there is really no reason at all why good design should not be able to achieve a more highly desirable value substantially less than the 0.18 seconds recommended as a maximum in §8.5.1.

The major source must again lie in the forward path since no feedback response is possible in the first instant. The structure of this path should be such that a step input applied at the stick results in a step or near-step demand to the actuator, sufficient to produce a desired initial response acceleration. This does not preclude the use of filters in the path, but it essentially implies that the number of poles and zeros must be equal. In this context, an apparently negligible command pre-filter lag with a 0.1 second time constant adds 45° phase lag at 10 radians/second, well within a pilot’s range of interest, while achieving only the small attenuation of -3 dB. Unfortunately, much larger pre-filter lags have been used in some past designs, with a substantial degradation of the handling at best and PIO tendencies at worst. A practical design can permit a few very small lag terms in this path, but they should be strictly limited in their total effect.

The subsequent aircraft acceleration response will be influenced by the total closed loop response, as the latter must become significant before the acceleration can decay to zero, or even reach its peak if there is a large high order lag effect. In the structure of Figure 10-1, over-dependence on the integrator to control the system, especially an unstable one, can lead to poor or PIO-prone handling. The proportional forward path produces an initial instant acceleration response, but if its gain is low it results in a low acceleration peak and cannot stop the integrator from creating a sluggish response, adding considerable high order lag at low frequencies.

Hoh et al [1982, page 160] illustrate this problem by the response of the independent backup pitch control system of the unstable USAF AFIT/F-16 in the power approach phase, with the control law block diagram shown in Figure 10-3. The integrator gain is 4.6 times greater than the proportional gain, which itself is very low. The pre-filter on the stick signal was necessitated by the severe signal noise problems associated with the quasi-rigid force sensor stick, but even though the time constants seem small it adds significant high order phase lag at relatively low frequencies. With a PIO frequency is 0.26 Hz, average phase rate of 173°/Hz and PIO gain of -8.1 dB °/lb, serious PIO tendencies are predicted by the author’s PIO criteria of Figure 8-15. When tested in the Calspan NT-33 in-flight simulator, the aircraft was considered extremely sluggish with very heavy stick forces and the pilot ratings were 8 or 9. (It is not stated whether the design was later improved.)

Incorrect implementation of integrators has been the cause of another non-linear high order delay effect contributing to overcontrol and PIO. If the integrator path saturates at an amplitude limit and is allowed to continue to "wind up" beyond it, then on reversal of the demand below the limit the signal remains there until the integrator has "unwound" again. It has to be ensured that the integrator cannot not do this so that it also reverses immediately. An example of this is given in Anon [1997, page 60] from a PIO very early in Boeing 777 flight testing, but the solution also required the reduction of the integrator gain to 38% of the original to minimise further high order effects. (In this case, proper corrections were implemented during an intensive analytical and flight test investigation to eliminate all possible causes of PIO before service entry.)
Pre-filter response shaping

As discussed further in §10.4, response shaping by command path pre-filtering is often used to optimise the handling. It must be ensured that such filters do not become contributors to PIO.

Figure 10-4 shows notional pre-filtering and command gain augmentation added around the outside of the closed loop part of a proportional plus integral PRD system. They have no effect on the steady state response, which is always determined by the integrator. The initial acceleration response can be increased to any desired level by the gain factor $K_{PF}$, adding to the existing proportional forward path term if the latter is not sufficient on its own to produce the rapid angle of attack response required for good flight path control. As noted in §4.3.1, this technique of a direct stick to actuation signal was used successfully in the FBW Jaguar project [Daley 1984, Smith 1997]. No undesirable high order effects are likely, though the gain increase should be no more than is absolutely necessary in order to avoid actuation saturation.

Pre-filters may also be incorporated in forward paths to further shape the general handling responses. These should generally abide by the equal "pole and zero numbers" rule above. It will be shown in §10.4 that pre-filter enhancement of the pitch response in accordance with the author's handling methodology results in filters that satisfy this rule. The basic form of pre-filter that should always be considered as the first option is a lag-lead or a lead-lag. Figure 10-5 shows gain and phase characteristics of a typical lag-lead filter, which provides desirable attenuation characteristics. The gain at high frequencies is equal to the ratio of the pre-filter lead and lag time constants. The phase angle dips to a moderate lag before returning to zero lag at higher frequencies, preventing the introduction of PIO-prone behaviour.

In systems of the Figure 10-2 structural type, the pole-zero rule has been commonly violated by the use of a first order lag pre-filter, intended to attenuate an excessively abrupt response caused by the high forward path gain basically inherent in this layout. Figure 10-5 shows how a plain lag filter departs far from the desired gain and phase characteristics. It has the undesirable high frequency feature of completely suppressing the demand signal in the initial instant of a step input response, whereas what is wanted is simply some high frequency gain reduction. The lag pre-filter provides attenuation similar to the lag-lead type only at low frequencies, but adds substantial phase lag while doing so and even more lag as frequency increases further. The potential for PIO should be clear from §8.4.

10.4 Design for good pitch handling

The poor and sluggish pitch handling that can be exhibited by an aircraft with a proportional plus integral pitch rate demand system was exemplified above by the AFTI/F-16, though the problem is not confined to this demand mode type. That aircraft was only mildly unstable. Achieving a stable closed loop system is quite difficult when the aircraft is extremely unstable, even with phase advance filtering to compensate for actuation lags, notch filtering, digital transport delay and so on. This may prevent the achievement of optimum handling qualities directly, because it may not be possible to utilise the desired gains due to system stability limitations. However, such aircraft can have excellent attitude stability and resistance to disturbance.

An example of the effect of added feedforward elements on the superaugmented highly unstable EAP is discussed in Gibson [1986] (in §6.1 and Figure 14 of the reference). The basic stabilised response was well damped but had low bandwidth, with CAP of only 6 deg/sec/g, some attitude overshoot, a long flight path delay and marginally satisfactory high order PIO dynamics.

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The CAP was increased to a nominal 30 deg/sec²/g by adding a feedforward gain path to generate a fast response. Final tuning for precision attitude control was then provided by lag/lead pre-filters, achieving an excellent $K'/s$-like response with CAP of 18 deg/sec²/g, zero dropback, high bandwidth, reduced flight path delay and excellent PIO-resistant dynamics. The result was a combination of light and quick handling response with heavily damped attitude stability and resistance to turbulence disturbance, features not possible with conventional aircraft.

In the special case of optimised precision attitude control with a $K'/s$-like response, little attitude dropback and moderate pitch rate overshoot (§5.2.5 and Chapter 7), the flight path time delay closely matches the attitude response numerator coefficient $T_{\theta_2}$. For aircraft of the 1939-1945 era, this was commonly the natural result of their aerodynamics and resulted in good attitude control. Much of the data base which determined the standard specification of short period frequency limits [Anon, 1969] came from these aircraft, for which attitude criteria were unnecessary. Such pre-filtering will have an impact on the flight path control and it may be necessary to compromise between the two responses. Occasionally the needs of path control will be predominant, as in the landing aircraft.

10.4.1 Basis of pre-filter design

Figure 10-6 revisits Figure 7-5 to illustrate a simple graphical process for determining the time constants for a lag lead filter. If this is intended to produce a $K'/s$-like attitude response, a step control input should result in a ramp-like profile commencing at time zero, apart from the inevitable initial transient dynamics. The desired pre-filter time constants are taken as the flight path delay in the unfiltered time response and the nominal $T_{\theta_2}$. A property of a step transient response is that the addition of a lag or lead filter to the input moves the response's steady state path intercept with the time axis by an amount equal to the added time constant, to the right for a lag and to the left for a lead.

More generally, the effect of adding a pre-filter of this type, where the pre-filter numerator time constant $T_{\theta_2\text{new}}$ is chosen to suit the desired purpose, e.g. equal to the unfiltered $t_y$ for zero dropback, has different consequences for the pitch attitude and normal acceleration responses as follows:

\[
F_1(s) = \frac{1 + T_{\theta_2\text{new}}s}{1 + T_{\theta_2}s} \tag{10.1}
\]

(eq. 3.2)

\[
\frac{\theta(s)}{\delta_e(s)} = \frac{K_q(1 + T_{\theta_2}s)}{s[1 + (2\xi_s/\omega_p)s + (s^2/\omega_p^2)]} \tag{10.2}
\]

\[
\frac{\theta(s)}{\theta_{com}(s)} = \frac{\theta(s)}{\delta_e(s)} F_1(s) = \frac{K_q(1 + T_{\theta_2\text{new}}s)}{s[1 + (2\xi_s/\omega_p)s + (s^2/\omega_p^2)]} \tag{10.3}
\]

where \(\frac{\theta(s)}{\theta_{com}(s)}\) is the commanded attitude.

The resulting attitude transfer function (10.3) remains classical in form, but behaves in response to pilot commands as though the wing loading and/or lift slope have been altered.
\[
\frac{n(s)}{\delta_e(s)} = \frac{K_q V}{g(1 + \frac{2\xi_s}{\omega_p} s + \frac{s^2}{\omega_p^2})} (1 + T_{02new}(s))
\]

\[
\frac{n(s)}{n_{com}(s)} = \frac{n(s)}{\delta_e(s)} F_1(s) = \frac{K_q V}{g(1 + \frac{2\xi_s}{\omega_p} s + \frac{s^2}{\omega_p^2}) (1 + T_{02}(s))}
\]

where \( \frac{n(s)}{n_{com}(s)} \) is the commanded normal acceleration.

The normal acceleration response to pilot commands is no longer in the classical form, having acquired the extra lag-lead filter dynamics in full. The vertical velocity response (effectively the flight path angle) and the path displacement response also contain the filter dynamics. However, the responses to the external disturbance caused by turbulence are not altered in any way.

To obtain the filter dynamics graphically in Figure 10-6, a choice of extrapolation data points at 2 and 4 seconds represents a time scale within a pilot’s prediction capability, within which a settled response should be achieved. A secondary "goodness" check can be made to compare the response slopes at 4 and 7 seconds for example. A significant difference could indicate an excessive amount of low frequency mode response with a too long settling time. Clearly this process could not be followed if the full three degree of freedom response characteristics are calculated, since the effects caused by speed variation would be confused with the filter effects. Hence only the two degree of freedom fixed speed approximation to the short period response should be used in this process.

The need for a lag-lead or a lead-lag filter is determined by whether the basic unfiltered response has attitude dropback or overshoot characteristics respectively. In Figure 4-2, a marked attitude overshoot characteristic can be deduced from the lack of pitch rate overshoot, as well as a sluggish flight path response from the slow increase of the angle of attack. The use of a lead-lag pre-filter \((1 + T_{02s})/(1 + T_{02})\) would restore the "short period" frequency and transient responses to the "conventional" shape seen also in the Bode plot. It would have no effect on the low frequency long term responses. These must be considered separately and modified for example in association with the provision of some form of speed stability, as noted in §4.4. It will be recalled that this speed stability may have no connection with the classical angle of attack stability through which conventional aircraft maintain a stable trimmed speed.

10.4.2 Bandwidth effects - attitude and flight path

The influence of the pre-filter parameters can also be inferred from the Bode gain asymptote sketches in Figure 10-7. In the classical \( K/s \) like response, the flight path time delay \( \tau_p \) is equal both to \( 2\xi_s/\omega_p \) and to the attitude numerator coefficient \( T_{02} \), so that \( \omega_p T_{02} = 2\xi_s \). As the width of the attitude "shelf" (see the Bode plot in Figure 7-15) is also \( \omega_p T_{02} \), it must be equal to \( 2\xi \) in the \( K/s \) case. The example sketched in (a) represents a typical high bandwidth case, with a very wide shelf created by a high short period frequency. The result is a fast flight path response, but it also creates large attitude dropback and high pitch rate overshoot ratios due to a rapid angle of attack change as in the example of Figure 7-17.
To achieve the desired shelf dimension for a $K/s$-like response, the short period frequency could easily be reduced by control system means. This reduces the bandwidth, and the phase lag associated with the steeper gain slope of -40 dB/decade shifts to lower frequencies, reducing the potential piloted crossover frequency (Chapter 5). The nominal $K/s$ would be achieved but the initial response transient would be more sluggish. $T_{p2}$ is a property of the aerodynamics and wing loading of the aircraft and cannot be changed, but its effective value could be reduced by direct lift control. This would require additional actuation and control surfaces, unless lift spoilers are available already, and the effect is limited to rather small increments of lift.

The most practical method is to change the apparent value of $T_{p2}$ by use of the Figure 10-6 lag-lead command pre-filter as sketched in Figure 10-7(b). The hatched area shows the added effects of the pre-filter. The attitude response gain asymptotes retain the conventional form with the associated phase characteristics, and the $K/s$ slope is maintained to high frequencies. The bandwidth is reduced slightly but retains a great proportion of the value in Figure 10-7(a), maintaining low phase lag in the region of interest to the pilot and enabling the achievement of a high open loop crossover frequency. Responses related to the flight path, e.g. normal acceleration, angle of attack and flight path angle, are altered to a non-conventional form represented by the added hatched area, as noted in §10.4.1.

Examples of these effects are shown in Figure 10-8, the transient responses corresponding to configurations A and C in Figure 7-18 (see §7.5.1 for a further discussion). Case A with large dropback and pitch rate overshoot was modified to Case C by the use of a lag-lead pre-filter. Case C has zero dropback but a much increased flight path time delay relative to A. The "two-slope" normal acceleration response in Case C is typical, showing the separate influence of the pre-filter lead and lag components. It can be seen that, completely unlike a conventional response, the pre-filtered attitude and rate responses settle quickly to a steady state, whereas the normal acceleration response (with the corresponding angle of attack not shown) continues to settle for a much longer period. The somewhat delayed arrival of the response at the steady state value was christened "g creep" in Bland et al [1987], though it was said to have no adverse effect on pilot opinion. The author believes that a pilot has less interest in the subtle variations in the transient shape of the normal acceleration than in the overall control of the flight path, borne out by the excellent low level flying qualities of the Tornado with a similar creep pattern.

With the use of such pre-filters, it is possible to construct responses differing remarkably from the conventional. Whereas the upper attitude bandwidth limit derived by Hoh et al [1982], indicated in Figure 7-16, was determined by excessive dropback effects at short high period frequencies (see §7.5.1), there is no evidence of any upper bandwidth limit when dropback is eliminated by a pre-filter. In this case, the transient flight path delay always equals $T_{p2}$. As $\omega_p$ or its equivalent in a high order system increases, the flight path bandwidth converges to $1/T_{p2}$ radians per second instead of increasing with $\omega_p$. This is because the attitude phase lag remains close to 90° up to higher frequencies. The flight path phase lag is derived by adding to this the phase lag of the of path angle to attitude transfer function, $1/(1 + T_{p2}s)$ from Eq. 3.7. Since this is 45° at $1/T_{p2}$ radians per second, the path angle phase lag is 135° at this frequency, and so by definition this is the flight path angle bandwidth.

### 10.4.3 Sensitivity

Figure 10-9 sketches a nominal attitude response with several unsatisfactory features. The right-hand "knee" lies well to the right of the early "Gibson criterion" boundaries shown in Figure 7-7 (see §7.3.2), indicating unsatisfactory precision attitude control with excessive dropback and pitch rate overshoot. The PIO gain (at -180 degrees phase angle) just fails to satisfy the desired
PIO gain limit from Figure 8-15. The pitch rate response gain at the bandwidth frequency, obtained by adding the bandwidth frequency value in dBS to the attitude gain, is well above the author's sensitivity criterion limit from Figure 7-27 (see §7.6). The attitude oversensitivity could be corrected by reducing the response gain, either with an increased feel system stiffness or a reduced command gain. This would result in an increased stick force per g, which might be undesirable from a manoeuvrability point of view, and it does not address the dropout problem.

A suitable pre-filter can satisfy all these criteria simultaneously. The typical lag-lead filter characteristic sketched in Figure 10-5 predominantly alters the phase lag at low frequencies with little effect on the gain, and alters the gain at higher frequencies with little effect on the phase lag. After applying such a pre-filter, the modified response now lies close to the 90° phase lag line at low frequencies, indicating a $K/s$-like behaviour. The pitch rate sensitivity has been reduced to lie with the criterion limits. The PIO gain lies below the optimum design aim limit of -20 dB deg/lb (Chapter 8). The resulting open loop aircraft plus control system response is now suitably configured for simple pilot gain adjustment to achieve satisfactory closed loop attitude control. All this is achieved without altering the stick force per g.

**Flight path considerations**

When flight path control is the dominant task, a $K/s$-like response in flight path is more important than in attitude, requiring low values of the parameter $t_r$. Most probably this will be in landing approach conditions, where many aircraft with high wing loading and high lift flap systems may possess a $T_{a2}$ value of about 2 seconds, indicating that they require a 2 degree angle of attack increment per degree per second of pitch rate. Zero dropout then implies a pitch delay $t_r$ of about 2 seconds also, which may be marginal for tight control of the flight path in the landing flare. The author suggested in §7.5.2 that values of 1.5 seconds and 1 second should be provided for normal and precision flight path control respectively. In turn, this implies dropout ratio values between 0.5 second and 1 second for such aircraft, with a substantial pitch rate overshoot in order to acquire the necessary angle of attack reasonably quickly. The author's landing approach criterion in Figure 7-7 permitted a substantial "knee" in the attitude frequency response to allow for this.

In principle, similar flight path responsiveness is as valid for large commercial aircraft as for small combat aircraft. While these aircraft commonly have a $T_{a2}$ of around 2 seconds also, it is unnecessary to provide small $t_r$ values which even if achievable could seem too abrupt due to the phase advance effect of a cockpit located far ahead of the centre of gravity (§7.5.3). These aircraft are set up for the final approach a long time in advance and normally have large runway length margins, and so are less dependent upon extreme precision in the very short term.

The necessity for high flight path bandwidth is alleviated by the use of head up displays that present a quickened flight path vector symbol. As mentioned in §9.3, this adds a pitch rate element to the path vector, causing it to move more quickly in response to a pitching motion of the aircraft. By the visual elimination of the path lag with $K/s$-like symbol dynamics, the pilot is enabled to control the flight path with great precision.

Large values of $T_{a2}$ are also a consequence of flight at high altitudes and Mach number, as can be inferred from the expression given in Figure 4-4:

$$T_{a2} = \frac{m}{\sqrt{\rho V SC_{Na}}}$$
The pitch rate per g in a manoeuvre is inversely proportional to true airspeed while the angle of attack per g is inversely proportional to dynamic pressure. Hence $T_{92}$, the angle of attack per unit pitch rate, increases when altitude is increased at constant true airspeed, or approximately at constant Mach number, and can easily reach values of 2 seconds or even 4 seconds in a very high wing loading aircraft. $T_{92}$ decreases less rapidly in supersonic flight than subsonically because the wing lift slope reduces with Mach number. The combined effect of conventional short period dynamics and path response with the larger $T_{92}$ leads to large dropback and pitch rate overshoot ratios. This is the natural result of the very low steady pitch rate - actually the flight path angle rate that ensues in the steady response because of the high true speeds - relative to the initial transient pitch rate while acquiring the necessary angle of attack. It does not mean that the angle of attack response is excessively abrupt.

Cox et al [1997] give the extreme example of the Lockheed SR-71 and its predecessor YF-12. Their $T_{92}$ values reach 5 to 8 seconds in Mach 3 cruise at altitudes of 70,000 feet or more. The primary precision control task is control of altitude, normally performed by the autopilot. In a NASA investigation of the handling, a piloted height change task was flown requiring the pilot to acquire a 1000 feet per minute climb rate and to capture an altitude increment of 2000 feet. It can be estimated from the limited data given by Cox that the path delay was approximately 0.5 seconds (quite small due to the short period natural frequency of 2.7 radians/second), the attitude dropback ratio was 5 seconds and the pitch rate overshoot ratio was 6. Only 0.3° attitude change was need for the climb, due to the high speed, and this was acquired at a pitch rate of about 0.07 °/second and with a normal acceleration increment of 0.11 g. The actual dropback was about 0.35°. Clearly, the extreme attitude time response parameters were irrelevant since they could scarcely be observed by the pilot. It was concluded that the pilot did not use attitude as a primary reference cue as would normally be the case. Instead, the climb rate display derived from the inertial platform, mounted just aft of the cockpit and hence quickened or phase advanced by its position a long distance ahead of the centre of gravity, was the dominant cue.

For most other aircraft, such precision in piloted control of of flight path is seldom a strong requirement at high altitudes, but the task is certainly likely to take priority over precision attitude control. Zero dropback should therefore not be provided unnecessarily, where suppressing it would lead to a slow angle of attack response and sluggish control of flight path. This would arise from the need for very large time constants in the pre-filter denominator, equal nominally to $T_{92}$ seconds, In such conditions, the attitude sensitivity plays an important role, because the pilot must normally maintain the stabilising attitude inner loop when controlling the flight path. In §7.3.1 the author quotes from early studies that suggest a pitch rate overshoot ratio of 3 as an upper limit. It seems more likely that the significant factor is the pitch rate sensitivity. As noted in §7.6, the author has found that this can be identified in the time domain by the ratio of the peak transient pitch rate to the applied stick force. The criterion suggests that a peak maximum value of 1.1 °/second/lb could be taken to mark the upper limit of quick attitude response regardless of the pitch rate overshoot ratio, unless there is an overriding need for a faster flight path response.

Much of the conflict between attitude and flight path control considerations can be reduced by use of an non-linear variable lag-lead filter algorithm given in Winter [1976]. This alters the filter numerator time constant as a function of the amplitude and rapidity of the stick motions. It enables precise attitude control with near zero dropback to be achieved in tracking situations where small stick inputs are used, but provides rapid flight path response for large stick inputs by increasing the numerator and so reducing the pre-filter attenuation. This algorithm was used extremely successfully in the EAP [McCuish and Caldwell, 1996] and in the Eurofighter.
10.5 Some roll response design considerations

Roll control fly by wire demand systems typically do not exhibit the wide range of unconventional, non-classical response dynamics found in many pitch systems. In essence, they are intended to maintain the basic roll mode response exemplified by the one degree of freedom roll damping simplifications given in §3.3.2. In practice, the actual response is far from being such a simple first order type, as it will contain higher order elements such as the actuation system and contaminants arising from the Dutch roll mode, inertial cross-coupling, etc. Suppression of these requires the introduction of a number of lateral-directional-pitch crossfeed and yaw/sideslip control and and damping enhancement paths in the lateral control law structures. These are invariably designed to embrace roll and yaw into a single entity.

A substantial discussion of roll performance and handling is presented in Hoh et al [1982, §3.5 pp. 347-518], with guidelines for acceptable limits of roll response contamination by the other much more complex modes. However, provided that these are well suppressed, the roll response can be represented very adequately by the first order model equations 3.11 to 3.13 for the purposes of generic handling qualities study. Only these models will be discussed below. The reference considers high order effects found in a number of past fly by wire systems, including roll ratchet and high order PIO tendencies, but fails to identify the optimum means to avoid both of these problems.

In the 1960s the author proposed a quasi-fly by wire structure for improved roll control of variable wing sweep designs then being studied, shown schematically in Figure 10-10\textsuperscript{19}. It comprised a conventional mechanical control circuit and a parallel large authority electrical signalling system with a stick pick-off signalling a scheduled roll rate demand. The lag pre-filter in the latter path approximately matched the roll rate feedback dynamics, the difference between these signals being sent to an electro-hydraulic actuator that was summed with the pilot's mechanical signals at the main power control actuator. This system was capable of providing a much more consistent rate of roll than the simple mechanical control could do, and also provided excellent roll damping.

While lacking their total flexibility, the Figure 10-10 structure contained the necessary elements of the later full fly by wire systems, being identical functionally to the widely used Figure 10-2 structure. The combination of a direct mechanical forward path with a lagged electrical forward path is equivalent to a single forward path with a lag-lead pre-filter, as can be seen from the following assuming typical system values:

\[
1 + \frac{2}{(1 + 0.45s)} = \frac{3(1 + 0.15s)}{(1 + 0.45s)}
\]

With the high frequency or step signals passed primarily through the mechanical path, there was no excessive high frequency gain. At low frequency the full manoeuvre demand process prevailed, with a total forward path gain three times the mechanical gain.

The system was used in the Sepecat Jaguar fighter-bomber aircraft for its roll autostabiliser with an authority of some 20% of the roll control. It became the basis of all subsequent fly by wire roll control designs at BAe Warton, with the mechanical path replaced by an electrical one.

\textsuperscript{19} The system evolved naturally from a small authority pitch manoeuvre enhancement system for the British Aircraft Corporation TSR-2, but it was not tested before this project was cancelled in 1965.
10.5.1 Optimising the roll demand system

The stability margins in a roll damping loop are typically large and it is quite easy to reduce the augmented roll mode time constant towards a value of 0.1 seconds, at least at high speeds. This may be equivalent to three, four or even more times the natural aerodynamic damping in a combat aircraft. Since the additional damping is inertial, i.e. derived from a roll rate gyro, the aircraft becomes heavily attitude stabilised and unresponsive to turbulence, a very desirable feature in the turbulent lower atmosphere levels. The forward path demand gain must be considerably increased relative to the actual control authority available, in order to counter the inertial damping and also provide the nett roll control surface angles required to match the aerodynamic roll damping. This potentially gives rise to actuation saturation or roll ratchet problems.

With a 0.1 second roll mode, the initial roll acceleration in a step response initiated by the pilot is numerically ten times the steady roll rate. At the 200 deg/sec typical of combat aircraft, the resulting acceleration of 2000 deg/sec$^2$ is grossly excessive, though unlikely to be reached in practice due to attenuation by the control actuators. There is little documented evidence of a general upper limit to acceptable roll acceleration from flight experience. A piloted fixed base simulation reported by Drajeske et al [1990] found that the time to capture accurately a change in roll angle of 90 degrees was a minimum when the initial acceleration was 800 deg/sec$^2$, and this time increased for either less or more acceleration. Simulated without motion, this result neglected the effects of the corresponding lateral acceleration at the pilot's head.

In the British Aerospace EAP research aircraft, the maximum acceleration of 500 deg/sec$^2$ was considered to be exceptional by its pilots [McCuish, 1996], but they also felt it detracted from precision roll angle capture. This was equivalent to 1g lateral acceleration at the head level. Lower accelerations of 400 deg/sec$^2$ in the control laws for initial flying had been satisfactory. In this aircraft the pilot was seated substantially above the instantaneous rolling axis, and it is possible that higher accelerations could be acceptable for lower cockpit positions.

Figure 10-11 shows the time to roll through different roll angles at these accelerations. Typical maximum times permitted in performance specifications [Anon, 1980 and 1987] are easily satisfied. From this point of view, very high roll rates are not required and roll mode time constants less than about 0.4 seconds seem unnecessary for maximum roll rates of 200 deg/sec, since that would give an initial roll acceleration of 500 deg/sec$^2$. However, precision roll control is enhanced by smaller roll mode time constants since the higher damping reduces the roll angle overshoot after the stick is centred. This can be resolved by using smaller time constants in the linear design, but limiting the maximum roll acceleration for gross and rapid stick inputs by a rate limiting function. There seems to be no reason why the non-linear amplitude-dependent pre-filter noted above should not be used for this purpose, though it has not been so used to the author's knowledge.

Control law strategy

As always, the essential first consideration in the design of a roll control law structure is the elimination of any potential for high order PIO. This was discussed briefly in §8.7 with suggested criteria adapted from Figure 8-15. The same generic design requirements apply as for pitch control, namely that there must be a sufficient measure of direct unlagged signal path from the pilot's inceptor to the control surface to provide the connectivity needed if the pilot is to feel in control. The magnitude of this direct path must be restricted at high frequency to avoid control saturation, but the special problem of roll ratchet is an additional reason to limit it.
The generalised control law structure of Figure 10-2 can be refined for roll control as shown in Figure 10-12. The direct link path gain $K_{DL}$ is scaled to provide the actual control surface angle needed to drive the roll rate, providing also an instantaneous roll acceleration response. The lagged manoeuvre demand path $K_M F_M K_p$ provides the additional control angle needed to counter the roll rate feedback and to generate corrective steady state and dynamic signals towards achievement of the required roll performance and piloted roll mode time constant.

The dynamics of this twin forward path structure in Figure 10-12 exactly resemble that of a simple lag-lead filter applied to a single forward command path equivalent to $(K_{DL} + K_MK_p)$. The time constant $T_3$ is selected to represent the nominally desired roll response model. $T_1$ follows from the ratio of the direct link gain to the total forward path gain. Using the earlier example, with a manoeuvre demand path gain twice the direct link path gain, $T_1$ is one third of $T_2$. Because there is in effect always a direct unlagged path to the surface actuator, no significant high order effects are introduced. The command gain at high frequency is 33% of the low frequency gain, and is similar to the control authority of conventional aircraft where ratchet is virtually unknown. Hence both PIO and roll ratchet can be prevented.

Attempts to alleviate ratchet or excessive acceleration by simple lag pre-filters have caused PIO, as discussed in §9.5.1. The total dynamics then include the normal roll mode and the added lag, requiring the pilot to control roll angle through an integration (the roll rate) and two first order lags (the pre-filter and the roll mode), with a potential maximum of 270 degrees instead of the classical 180 degrees of phase lag. Further problems have been caused by placing structural notch filters and other dynamic compensation (necessary elements not shown in Figure 10-12) in the forward path. These should be placed in the feedback path where they cause no damage to the directness of the path between the pilot and the controls. An object lesson in the above is given in a paper on the McDonnell Douglas C-17 control law development [Iloputaife, 1997].

Further roll acceleration considerations

In conventional aircraft, the steady roll rate for a given control input is proportional to true speed but the initial roll acceleration is proportional to dynamic pressure, if the effects of aeroelasticity and Mach number are neglected [Gerlach, 1983]. Consequently the roll acceleration for a given control deflection reduces while the roll rate remains the same as altitude is increased at constant true speed, or approximately at constant Mach number. Since the roll mode time constant is equal to the initial acceleration divided by the steady rate, its natural variation is to decrease as speed increases and to increase as altitude increases.

In a fly by wire aircraft, it is possible to maintain constant maximum roll rate by scheduling the control surface demand, within the available control authority. To maintain a constant roll acceleration and hence also a fixed piloted roll mode time constant requires a different control demand schedule that cannot simultaneously provide a constant roll rate. However, it does become possible if the twin forward path structure of Figure 10-12 is used in a slightly different way. The direct link gain $K_{DL}$ is now scaled to provide the required initial acceleration. The structure is unchanged but the value of $K_M$ is modified to maintain the actual control angle required for the steady roll rate as well as to counter the roll rate feedback, so that $(K_{DL} + K_MK_p)$ is maintained unchanged.

In practice, it is unlikely that such a constant response could be achieved with practical control authority limits, particularly at low speeds at high altitude where large control inputs will be required to achieve it. At high speeds, where roll ratchet or excessive lateral g at the pilot's head is
more likely to be a problem, the principle can be useful in reducing the roll acceleration due to the high frequency control demand. In the landing condition, where PIO prevention is most important, it is certainly advisable to use the available control authority to the full in the direct link path, to provide a very positive and direct response.

Such variations are unconventional but they do not in principle depart from the classical roll response form. They further illustrate the power of a modern fly by wire system to provide response characteristics desirable to the pilot rather than just mimicking what has gone before.

10.6 Testing the design before flight

No aircraft of any significant complexity will enter the flight testing stage before its handling has been closely examined, both by intensive computer simulation and by some form of ground or airborne flight simulation. The latter tools were discussed in §5.3. Where detailed handling qualities specifications have been imposed, the computed handling will be checked against them. These are however largely small perturbation requirements, and much of the testing must involve pilots exercising the design to the limits of its manoeuvrability. At this stage the control law designer's confidence may be severely tested in the face of unexpected criticisms by pilots.

At the present time there is no universal set of handling criteria widely accepted as offering a guarantee of good handling in all circumstances. As noted before, pilots may disagree among themselves (and with the engineers) over the apparent qualities revealed in a simulator, to an extent depending on their previous experience both of the type of control laws used and of simulation itself. This situation should improve as fly by wire technology becomes more widespread and the population of pilots becomes more familiar with it.

In the author's experience, the most important conclusion to be established before first flight is that it will be safe, and that it will also be easily controlled. Whether or not the handling is optimum can be left until examined in actual flight, the only domain in which proof is ultimately possible. It should always be expected that some development of the handling may be required during the development program, and if this is not allowed for with a series of planned updates extreme frustration and future delays are probable.

To establish safety, it is essential that all the non-linear and dynamic effects of the flight control system hardware are exactly simulated. Unexpected loss of control or PIO cannot be predicted with any certainty unless this is done. Severe but realistic tasks are essential, designed to provoke high gain pilot behaviour in critical situations. For combat aircraft, pilots should attempt to upset the system stability by control inputs limited only by overstressing considerations. If this is supposed to be prevented by the system, then full control inputs should be applied in any and in random order. In principle, even commercial aircraft ought to be subjected to similar testing, the extraordinarily thorough PIO testing carried out by Boeing on the 777 prototype pointing the way admirably [Anon 1997, Dornheim 1995].

In flight, PIO may not be encountered for a long period until initiated by an unforeseen trigger. The most widely used simulation test for the landing case is the offset approach, with a correction to line up initiated at the last possible moment. This can provoke either pitch or roll PIO during the coarse control activity required immediately prior to the touchdown. If tested by in-flight simulation, it is essential that the task is completed to the touchdown, requiring a safety system by which the safety pilot can take over control from the middle of a PIO with the wheels
just about to touch. Any in-flight simulator which cannot provide this level of performance will not discover landing PIO tendencies, where PIO commences only within a few seconds of touchdown. While the cues in a ground simulator are much less strong, it does have the advantage that no safety is involved and a landing can always be taken to the point of a crash.

However, as discussed in §9.6, attempts to uncover PIO by looking only for triggers may not be entirely advisable. It should be taken as axiomatic that if the aircraft dynamics are such that a large oscillation can be excited deliberately at the appropriate frequency and relative phasing described previously, then an inadvertent PIO will eventually occur. PIO susceptibility can be assessed quite well in fixed base simulation by exciting the oscillation at all stick amplitudes including the largest possible. As far as is known, this technique has not been used for in-flight simulation or indeed in tests of actual aircraft, but there seems to be no reason why it should not. Simulation cannot be relied on to uncover every possible trigger situation, however.

What is absolutely essential is to avoid complacency just because the basic handling appears to be good. To dismiss a simulated PIO resulting from extreme inputs by saying that a pilot would never fly like that is a sure path to disaster.

10.7 Summary

In this chapter, the author's discoveries of previously unidentified facets of handling qualities, unique methods to describe them, and a practical approach to the understanding of the widely experienced problems of high order PIO, have been put together into the basis of a simple design process by which control laws can be shaped to provide good handling.

This methodology is not associated with any particular design method for aircraft stabilisation and feedback control systems, which range from the classical to modern control theories of many types. It is a final stage in the process of turning a stable and well controlled aircraft in the closed loop system sense into one which the pilot can operate with the minimum of workload and the maximum of good opinion in all its required tasks.

The most important starting objective in the control laws design is to set up a structure in which every consideration is taken account of in the elimination of high order PIO potential. The most significant rule to be followed is that there must be an absolutely proportional connection between the pilot's inceptors and the control surface actuation. To this extent the methodology may influence the technique actually used in the overall system design, because the engineer must maintain a detailed knowledge of and exercise proper control over the signal pathways and interconnections. The control laws must be tested to confirm a negligible possibility for PIO as they are developed, not after they have been nominally completed when a costly redesign might prove to be necessary.

The basic tools for the final optimisation are the direct stick-to-surface feedforward and the command pre-filter, almost invariably of the lag-lead characteristic and practically never a simple first order lag. A number of rules are given to determine the filter time constants, based on principles of shaping the pitch attitude and flight path time and frequency response plots to provide aircraft dynamics that are optimised for satisfactory pilot control.
Figure 10-1 Nominal proportional plus integral PRD system

Figure 10-2 Simple proportional feedback fly by wire system

Figure 10-3 Example proportional plus integral PRD system with unacceptable features (AFTI/F-16 independent back-up mode in power approach, Hoh et al [1982])
Figure 10-4 PRD system with command pre-filtering
Feedforward elements outside the closed loop have no effect on the steady state.

Figure 10-5 Typical prefilter gain and phase characteristics
Lag-lead pre-filters provide response shaping and attenuation with negligible adverse high order effects.
Lag pre-filters usually provide unsatisfactory response shaping and have substantial adverse high order effects.
1) Estimate basic response dropback and path time delay by intercepts from 2 and 4 second data points.
2) Make pre-filter denominator time constant = $T_{\theta 2}$ secs.
3) To obtain $K/s$-like attitude response, make pre-filter numerator = $t_{\gamma}$ secs.
4) Command pre-filter $= \frac{1 + t_{\gamma}s}{1 + T_{\theta 2}s}$ for zero dropback

Figure 10-6 Selection of prefiltet time constants to achieve a "$K/s$"-like transient response

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Figure 10-7 Short period Bode gain asymptotes required for precision attitude control
Figure 10-8 Effects of pre-filter on pitch response transients showing "g creep" in normal acceleration.

Figure 10-9 Attitude response optimisation by pre-filtering to K/s-like shape, with optimum pitch rate sensitivity and low PIO gain.
Figure 10-10 Schematic layout of an early partial fly by wire roll control system

Figure 10-11 Time to roll vs roll rate for given peak roll acceleration

Forward path structure: \[ \frac{K_{DL} + K_M K_P (1 + T_1 s)}{(1 + T_2 s)} \]

- \( K_{DL} \): Direct link gain, control angle per unit stick
- \( K_M \): Roll rate demand gain, deg/sec per unit stick
- \( K_P \): Feedback gain, control angle per deg/sec roll rate
- \( F_M \): First order lag filter \[ \frac{1}{(1 + T_2 s)} \]
- \( T_2 \): Nominal desired roll mode time constant for pilot input
- \( T_1 = \frac{K_{DL}}{K_{DL} + K_M K_P} T_2 \)

Figure 10-12 Basic roll demand control law structure for response optimisation
Chapter 11
Concluding remarks and recommendations

This thesis has been written after 51 years in which the author has been engaged in aeronautical engineering, and 60 years after his fascination with the world of aviation began. This period has been filled with exceptionally interesting developments and advances, though possibly not relatively more so than in the preceding half century. While there has been little or no further increase in the sheer speed performance of combat aircraft, the great advances in the technological content of airframes, engines and systems would seem revolutionary to earlier practitioners. The impact of some of these developments on flight control has been outlined in this thesis. The result has been ultimately favourable, though sometimes the initial application revealed a lack of understanding that led to the serious handling difficulties discussed in the above pages.

It has been the author's good fortune to have been involved in the design of five production series aircraft types, one of them with a pre-cursor fly by wire control system and two with full fly by wire controls, and additionally of three fly by wire research types. Today's young control designers will be fortunate to have detailed experience of more than a couple of types or so in their working life, due to the now usually long life cycle of design, development, flight testing and production. This is frequently followed by recurring mid-life system upgrades that have come to replace the frequent new airframe designs that were once the life blood of aircraft companies. The author has therefore been placed in a unique position to have experienced a multitude of handling problems, to have observed those of many others, to have played a significant part in developing solutions, and to have seen these followed through to satisfactory flight testing.

Chapter 2 outlined how an understanding of handling qualities, or at least of their definition, was acquired very slowly indeed, the ability of pilots to cope with an extraordinary range of qualities being a primary reason for this. This ability was also fortunate in that despite the nature of aircraft dynamics that can be made inherently both controllable and stable without much difficulty, all too often they were either insufficiently stable or were unstable, and though if they flew at all they were obviously controllable, this was sometimes only with considerable difficulty. It took 40 years from the Wrights' first powered flights to arrive at a comprehensive definition of satisfactory handling qualities. It took another 25 years to expand the definition to the realm of the high performance high altitude jet powered era. In each case the results were available essentially at the end of the major period of development of each class of aircraft, and proved inadequate for the challenges of new technologies in later classes.

Chapters 3 to 5 discussed the basic elements of handling qualities. These elements are the dynamics of conventional aircraft, dominated by their aerodynamic and inertial properties, the dynamics of "superaugmented" aircraft, dominated by the flight control system properties, and the dynamics of the human pilot who requires compatible and well matched aircraft dynamics for satisfactory handling. While the first two element sets can nowadays be computed to an extremely high degree of accuracy, the pilot remains something of an enigma, with little certainty that any two pilots will agree exactly about handling qualities. Decades of research have led to very advanced models of the human pilot functioning that can offer a quite close match to the pilot behaviour in observed events. Despite this, the author knows of no evidence that they have contributed effectively to the design of flight control laws that ensure excellent handling and prevent pilot induced oscillations. By contrast, the earliest and very simple pilot models have retained their power to illuminate the response dynamics that satisfy the handling design goals.
Chapter 6 discussed the influence of control system hardware on handling qualities. While at first sight the hardware might appear to be nothing more than a mechanical interface between the pilot and the control surfaces by which the aircraft is manipulated, and therefore should easily be made "transparent" to the pilot, it has often greatly degraded the handling. Again this was due to misunderstanding and neglect of its importance until the lessons were learned. In aircraft with powered controls, further control hardware became essential for the provision of good handling. As the pilot was now disconnected from the aerodynamic forces on the controls, artificial feel substitutes had to be supplied, which varied widely in complexity and effectiveness. The greatly expanded flight envelopes of advanced aircraft often required further complexities in the form of variable or non-linear gearings and inter-axis couplings. In the modern fly by wire era, the pilot-aircraft interface hardware reduces essentially to the inceptors and simplified artificial feel devices, but handling qualities are as dependent on their qualities as ever. With the commonplace aerodynamic pitch instability of many combat aircraft, the actuation dynamic performance is even more critical to the stability of the aircraft and control system closed loop.

Chapter 7 presents the core discoveries of the author from which his new methodology for the handling qualities design of control laws was developed. In the circumstances of his industrial employment, it was not possible to conduct experiments except in a very limited way. The material on which the research was based was found in the literature, mainly conference papers, technical journals and research reports in the public domain, dating back to the 1950s. This material, based on parametric formats and criteria, had showed little agreement in its conclusions about response limits for satisfactory handling. From the evidence of poor handling qualities in many early fly by wire aircraft, there was little clear guidance that could be applied to the design of control laws that would ensure acceptable, let alone excellent, handling in such aircraft. The author developed a non-parametric graphical description method that directly illuminated the response qualities desired by pilots. When the earlier criteria were transformed into this graphical descriptive form, the author found that there was rather good agreement in most of the sources about the essence of satisfactory handling. As his method of describing the handling was independent of response mode parameters, it could be applied without difficulty to any category of manoeuvre demand control system regardless of its non-classical dynamic form. Since these systems can be made to provide any desired response dynamics to a reasonably close degree, it became a simple task to design the control laws to provide excellent handling from the start rather than after a trial and error process of flight development.

Chapters 8 and 9 discuss a variety of pilot induced oscillation problems, that is oscillations that are actually sustained by the pilot's efforts to control them. Although PIO in some forms existed previously in conventional aircraft, it has come to prominence in fly by wire aircraft in the so-called high order PIO. These have caused many catastrophic accidents and serious incidents, mainly in the landing and sometimes in the take off phases of flight. In the aftermath of the author's first encounter with a PIO, he found that there were no criteria or any relevant material in the literature that could have predicted it, but he was able to develop both an explanation and a solution. As experimental PIO material became available, it provided further data to confirm the author's initial hypothesis. The author found a universal consistency in the response dynamics associated with high order PIO and in the pilot's reactions to them that enabled him to develop simple design criteria that have eliminated this particular PIO problem in every subsequent project with which he has been associated.

Chapter 10 describes the application of the author's methodology to the shaping of the time and frequency response dynamics, providing desired combinations of flight path and pitch or roll attitude, rate and acceleration characteristics. An essential tool for this task is a direct feedforward
path from the pilot's inceptor to the control actuation, containing no lag dynamics above the irreducible minimum. By this means the pilot is given the direct connectivity with the aircraft rotational acceleration response to maintain the feeling of positive control, which has been present in all conventional aircraft since the beginning of aviation. Other feedforward is provided as necessary in association with the elements of fly by wire control laws, such as a proportional plus integral controller or other highly augmented feedback. The second tool is a command path pre-filter. Its purpose is to provide aircraft attitude or flight path open loop responses that resemble as closely as is desired the idealised K/s integrator-like dynamics, with which pilots can easily interact with a low workload, where this cannot be provided directly by the basic feedback augmentation. Constraints are also placed on the initial time response transients to ensure satisfactory sensitivity.

An essential companion to the author's methodology is the provision of a linear or quasi-linear actuation system. This requires the close consideration of the actuation rate and acceleration limits to enable large and rapid control inputs to be applied without the introduction of sudden and gross changes in the actuation dynamics. This subject is outside the scope of this thesis, but the best linear handling design methods cannot prevent catastrophe if such effects are neglected.

By such means it is possible to enjoy the manifest benefits of modern high order fly by wire technology, coupling heavily stabilised hands-off flight characteristics with idealised handling qualities responses, a combination that is impossible in the conventional low order aircraft with classical controls. At the same time the high order handling problems that in many quarters have been wrongly seen as an inevitable consequence of fly by wire are completely prevented. The low order dynamics that are beneficial are replicated while their minor irritations and imperfections that have had to be accepted in the past are avoided.

The graphical and entirely non-mathematical descriptions and measures of "what the pilot sees" that are the basis of this thesis have enabled flight control engineers and pilots to speak the same language of handling qualities. The proof of their success lies not in formal expressions and equations but in the pilots' flight test reports of such aircraft as the FBW Jaguar and EAP experimental research aircraft, and the Eurofighter 2000 now entering the production phase. Some pilot comments reported in §7.6.2 that illustrate this are repeated here:

*Immediate, accurate and easily controllable response; transitioning from hard or maximum rate manoeuvres to fine tracking was exceptionally easy with excellent control response; settling times into a fine tracking situation were negligible. (EAP)*

*Took no getting used to; much better than any other jet I've flown; approach path deadbeat; my gentlest landing ever; landing remarkably easy; a joy from start to finish. (Eurofighter 2000 in initial reversionary control law mode)*

The extraordinary agility and and power of Eurofighter demonstrated at the 1998 airshows is coupled with precision control succinctly described as "easy to operate and fly accurately". It has exhibited excellent handling in the difficult flight refuelling task, and is considered to be of an order of magnitude better than conventional aircraft. In this it repeats qualities first demonstrated by the FBW Jaguar.

While this has resulted from the efforts of a great many engineers of the flight control and other disciplines, these words are the ultimate vindication of the author's work.
Recommendations

Of necessity, the author's investigations have centred almost wholly on the handling qualities of fly by wire combat aircraft. The same methodology is certainly applicable to large commercial and transport aircraft with fly by wire, but the appropriate limiting values of the metrics for these categories should be investigated since they are likely to be different.

An intensive study has been recently under way by many researchers of methods to correctly attribute by analysis the exact pilot ratings given to past PIO events. However, criteria that absolutely prevent the onset of high order PIO in future designs are of much greater importance. The aims of these researchers should be redirected to the pursuit of such design criteria, following the lead given by the author for the past 20 years.

Because all handling qualities design methodologies and criteria can ultimately only be tested in flight, there is a strong case to be made for the resurrection of in-flight simulation experiments to evaluate the great advances that have been made in recent years. The USAF variable stability F-16 Vista aircraft is the ideal tool for this. However, instead of the past practice of identifying the range of acceptable to unacceptable qualities, it would be more effective to search out the much smaller range of excellent qualities, since the power of modern fly by wire flight controls enables this goal to be satisfied to a considerable extent.

Such experiments should also investigate the differences in qualities desirable for different pilot inceptors, principally centre sticks for combat aircraft and wheel controllers for large aircraft, and side sticks for both types.
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De ontwikkeling van een ontwerpmethode voor besturingseigenschappen van Fly-by-Wire vliegtuigen

Samenvatting

De afgelopen periode geeft bijzonder interessante ontwikkelingen en een grote vooruitgang te zien op luchtvaartgebied. Het gevolg van een aantal van deze ontwikkelingen voor de handbesturing van vliegtuigen wordt in dit proefschrift behandeld. Uit vliegproeven bleek te vaak een fundamenteel gebrek aan inzicht hetgeen resulteerde in besturingsproblemen die in het onderhavige werk worden besproken.

Hoofdstuk 2 zet uiteen hoe het inzicht in besturingseigenschappen wel zeer moeizaam tot stand kwam. De belangrijkste reden daarvoor was wel dat vliegers adequaat bleken om te kunnen gaan met een bijzonder grote verscheidenheid aan besturingseigenschappen. Dit was te meer een gelukkige omstandigheid omdat, hoewel vliegtuigen zonder veel moeite goed bestuurbaar en stabiel kunnen worden ontworpen, ze meer dan eens of onvoldoende stabiel of onstabiel bleken te zijn.

Er was meer dan 40 jaar voor nodig om vanaf de eerste gemotoriseerde vlucht te geraken tot een beknopte definitie van goede besturingseigenschappen. Het duurde nog eens 25 jaar om deze definitie uit te breiden tot het gebied van de hoog vliegende straalvliegtuigen met grote motorvermogens. In alle gevallen waren de vliegproefresultaten pas beschikbaar aan het einde van de ontwikkelingsperiode van zo'n klasse vliegtuigen, en ze bleken bovendien niet toereikend voor de uitdagingen van nog nieuwere technologieën.

De hoofdstukken 3 tot en met 5 behandelen de basiselementen van besturingseigenschappen. Deze elementen zijn de dynamika van conventionele vliegtuigen die wordt gedomineerd door hun aerodynamische en traagheidseigenschappen, de dynamika van zogeheten 'superaugmented' vliegtuigen, die wordt bepaald door de eigenschappen van het automatische besturingssysteem, en de dynamica van de menselijke vlieger. Voor het verkrijgen van goede besturingseigenschappen moet de vliegtuigdynamika goed aangepast zijn aan de dynamica van de menselijke vlieger. Terwijl de eerste twee elementen thans met een buitengewoon hoge graad van nauwkeurigheid berekend kunnen worden, blijft de dynamica van de vlieger toch ten dele enigszins onvoorspelbaar.

Tientallen jaren van onderzoek hebben geleid tot geavanceerde modellen van het gedrag van de menselijke vlieger. Deze vliegermodellen kunnen echter een zeer goede overeenkomst vertonen met het vliegergedrag zoals dat in de praktijk is waargenomen. Niettemin hebben de vroegste en zeer eenvoudige vliegermodellen hun geldigheid bewaard. Ze geven een goed inzicht in het soort vliegtuigdynamika dat nodig is om de gestelde ontwerp-doelen voor de besturing te bereiken.

Hoofdstuk 6 behandelt het effect van de mechanische deel van het besturingssysteem - de 'hardware' - die de besturing vaak ongunstig heeft beïnvloed. Wederom werd dit veroorzaakt door onbegrip en verwarling van de belangrijke rol van het besturingssysteem totdat lering werd getrokken uit de ervaring. Bij vliegtuigen met servobesturing had de vlieger geen voeling meer met de aerodynamische krachten op de stuurvlakken en moest er voor kunstmatige stuurkrachten worden gezorgd. Oplossingen hiervoor vertoonden voor wat betreft complexiteit en effectiviteit een grote verscheidenheid. De sterk uitgebreide 'flight envelope' van geavanceerde vliegtuigen maakten vaak verder mechanische complexiteit nodig in de vorm van variabele of niet-lineaire overbrengingen en van koppeling tussen de verschillende assen van de vliegtuigbesturing. In het moderne 'Fly-By-Wire', FBW tijdperk is de koppeling vlieger-vliegtuig gereduceerd tot de stuurorganen en tot de voorzieningen voor kunstmatige
Hoofdstuk 7 is een weergave van de belangrijkste bevindingen van de auteur waaruit zijn nieuwe methode werd ontwikkeld voor het ontwerpen van regelwetten voor FBW besturingsystemen ter verkrijging van goede besturingseigenschappen. Doordat hij werkzaam was in de industrie was het voor de auteur - behalve op zeer beperkte schaal - niet mogelijk om experimenten uit te voeren. Het materiaal waarop het onderzoek is gebaseerd is daarom volledig ontleend aan de literatuur, hoofdzakelijk conferentiebijdragen, technische tijdschriften en onderzoeksrapporten in het publieke domein verschenen vanaf 1950. Veel van dit materiaal was gebaseerd op parametrische analyses en criteria. Er waren geen eenduidige conclusies voor wat betreft de grenzen waarbinnen tijdresponsies voor goede besturingseigenschappen zouden moeten liggen. Het feitenmateriaal over slachte besturingseigenschappen in veel vroegere FBW vliegtuigen leverde weinig aanwijzingen voor het ontwerp van regelwetten die aanvaardbare, laat staan uitstekende besturingseigenschappen van zulke vliegtuigen zouden kunnen garanderen. De auteur ontwikkelde een niet-parametrische, grafische beschrijvingsmethode die de door vliegers gewenste responsiekarakteristieken direct zichtbaar maakt. Voor wat betreft de grenzen van goede besturingseigenschappen vond de auteur een goede overeenstemming van deze methode met het bronnenmateriaal. De methode kon moeiteloos worden toegepast op elke categorie van manoeuvre commando systeem.

De Hoofdstukken 8 en 9 bespreken verschillende vormen van "Pilot Induced Oscillations", PIO's, oscillaties die in feite worden onderhouden door de vlieger die ze juist tracht te beheersen. PIO's zijn in FBW vliegtuigen opgetreden als zogeheten hoge orde PIO's. Ze zijn de oorzaak geweest van veel noodlottige ongelukken en ernstige incidenten, vooral in de landing en soms in de startfase van de vlucht. Nadat de auteur voor het eerst geconfronteerd was met een PIO, ontdekte hij dat er geen criteria of enig relevant materiaal in de literatuur voorhanden was dat deze PIO had kunnen voorspellen, doch hij was in staat een verklaring te vinden en een oplossing aan te geven. Naar gelang er meer publicaties over PIO verschenen, kwamen er tevens meer gegevens beschikbaar die de eerste hypothese van de auteur bevestigden. De auteur vond een universele samenhang in de dynamica van de vliegtuigresponsie die gerelateerd was aan enerzijds de hoge orde PIO en anderzijds de reacties van de vlieger daarop. Dit opende de mogelijkheid om ontwerp-criteria te ontwikkelen waarmee dit soort PIO problemen geëlimineerd konden worden in alle volgende projecten waarbij hij betrokken was.

Hoofdstuk 10 beschrijft de toepassing van de methode van de auteur om de tijd- en frequentieresponsies te kiezen en zo in te stellen dat PIO wordt voorkomen. FBW systemen kunnen zo worden ontworpen dat elke gewenste responsiedynamika met een redelijke mate van nauwkeurigheid kan worden verkregen. Het is dan betrekkelijk eenvoudig geworden om regelwetten te ontwerpen volgens de criteria van de auteur. Op deze wijze worden dan vanaf het begin goede besturingseigenschappen verkregen in plaats van na een moizaam ontwikkelingsproces van gissen en missen.

De basisgeredeescappen voor dit ontwerp zijn een directe voorwaartskoppeling, vanaf het stuurorgaan naar de besturingsservo, voor de initiële versnellingen en een vormingsfilter in de besturingslus om de responsiedynamika te manipuleren. Het gebruikelijke doel is om een open lus vliegtuigresponsie te verkrijgen die zoveel mogelijk lijkt op de geïdealiseerde dynamika van de zuivere integrator waarmee de vlieger zijn taak makkelijk en met geringe werkelasting kan verrichten. Tenslotte moeten er beperkingen worden opgelegd aan de aanvankelijke tijdresponsie teneinde een goede besturingsgevoeligheid te garanderen.

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Samenvatting
Curriculum Vitae

John Campbell Gibson was born in 1929 at Swatow, China. After secondary school in Edinburgh, Scotland, he performed two years of national service in the Royal Air Force before joining the English Electric aircraft company at Warton in 1952. He began work on the Lightning prototype flight controls design. This initiated a life-long association with all aspects of flight control, encouraging collaboration between engineers and pilots to ensure high standards of control quality. While working he obtained the Higher National Certificate in Mechanical Engineering, and was then sponsored by the Company to take the post-graduate Aircraft Design course at the College of Aeronautics in 1956 to 1958 for the DCAe diploma, subsequently converted to the MSc. He returned to Warton in the Aerodynamics Department to work on the design and specification of controls, feel systems, actuation requirements, and flying qualities of the Lightning, TSR.2, Jaguar, Tornado, the unstable Fly-by-Wire Jaguar digital control research aircraft, the unstable EAP canard-delta technology demonstrator, the VAAC Harrier V/STOL research aircraft and the Eurofighter 2000.

In the 1970's he commenced research into the nature of conventional and highly augmented handling qualities in order to understand the unusual or unacceptable behaviour found in some fly by wire aircraft with complex control laws. This was shown to be caused by pilot-aircraft closed loop factors which had been safely neglected previously and by response modes which could not be satisfactorily defined by conventional methods. After re-visiting the handling qualities research literature dating back to the 1950s, he discovered many previously unnoticed relationships from which he developed a wide range of new handling analysis methods and criteria for the design of fly by wire flight control laws. These specifically address and overcome the previously common problem of pilot induced oscillations and provide excellent handling for specific flight tasks, including the ASTOVL regime. He has presented papers at AGARD, ICAS and AIAA conferences and served on the AGARD Working Group 17, reporting on "Handling Qualities of Unstable Highly Augmented Aircraft".

In 1974 he was elected as an Associate Fellow of the Royal Aeronautical Society. In 1991 he was awarded the Bronze Medal of the Royal Aeronautical Society for his work on handling qualities. In 1992 he retired from British Aerospace but has been retained as a part-time consultant for an indefinite period. He was a Visiting Professor at Delft University of Technology, and is a consultant for the Empire Test Pilots School at Boscombe Down.
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