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DEVELOPMENT OF A COMMAND VALIDATION ALGORITHM

By

David Kristopher Ellis, B.Eng

A thesis submitted to the Faculty of
Graduate Studies and Research in partial
fulfillment of the requirements for the degree of
Master of Engineering

Department of
Mechanical and Aerospace Engineering
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Graduate Studies and Research acceptance of the thesis

**Development of a Command Validation
Algorithm**

submitted by

David Kristopher Ellis, B.Eng.

in partial fulfilment of the requirements for
the degree of Master of Engineering

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Carleton University

January, 1998

ABSTRACT

It is desirable for fly-by-wire aircraft intended for variable stability use to be of a 'single string' nature. This implies that there is a single set of fly-by-wire actuators, one flight control computer, a single set of aircraft sensors and a single set of flight control software. The simplicity of the design facilitates the incorporation of software changes without the overhead of multiple coding sources, multiple languages or operating systems and in-depth code validation. However, since there exists only one set of flight control software, some form of protection against control command failures must be present. The safety systems in place on the National Research Council's Airborne Simulator will not be sufficient for their Bell 412 Advanced Systems Research Aircraft due to its increased control power and lower time delays. The purpose of this thesis is to develop a software algorithm to examine flight control computer commands to rapidly determine their validity.

This thesis describes the structure and validation of Bell 412 flight models used for the simulation of aircraft response to actuator hardovers as well as the results of the simulations. Next, a command validation algorithm is developed based upon an assumed form of safety pilot response to an actuator hardover. A simulation evaluation of the algorithm and the technical challenges involved with its implementation in the aircraft is then discussed.

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TABLE OF CONTENTS

DEVELOPMENT OF A COMMAND VALIDATION ALGORITHM	II
ABSTRACT	III
ACKNOWLEDGEMENTS	IV
TABLE OF CONTENTS	V
<i>List of Figures</i>	<i>viii</i>
<i>List of Tables</i>	<i>ix</i>
<i>Notation</i>	<i>x</i>
<i>Variables</i>	<i>x</i>
<i>Subscripts</i>	<i>xi</i>
<i>Designations, and Abbreviations</i>	<i>xi</i>
1.0 INTRODUCTION	1
1.1 PREVIOUS RESEARCH	5
1.2 THE ROLE OF CVA	13
1.3 OVERVIEW OF THESIS	17
1.4 SCOPE OF WORK	20
2.0 BELL 412 MODEL STRUCTURE AND VERIFICATION	21
2.1 INTRODUCTION	21
2.2 HELICOPTER FLIGHT MECHANICS	22
2.3 MODEL STRUCTURE	26
2.4 PARAMETER IDENTIFICATION	32
2.5 MODEL VERIFICATION	37
2.5.1 <i>Hover Model</i>	38
2.5.2 <i>60 Knots Model</i>	39
2.5.3 <i>120 Knots Model</i>	41
2.6 SUMMARY	42
3.0 BELL 412 MODEL STEP RESPONSE	44
3.1 INTRODUCTION	44
3.2 EVALUATING THE BELL 412HP'S STEP RESPONSE – METHOD	45
3.2 ANALYSIS OF STEP RESPONSE DATA:	49
3.3 IMPROVING PILOT RESPONSE	53
3.3.1 <i>Pilot Model Background</i>	54
3.3.2 <i>Pilot Model Structure</i>	56

3.3.3 PILOTED STEP RESPONSE ANALYSIS	57
3.3.3.1 <i>Pstep, Qstep, Rstep etc...</i>	59
3.3.3.2 <i>Actstep</i>	59
3.3.3.3 <i>Analyze</i>	59
3.3.3.4 <i>Maxcursion & Maxanalyze</i>	60
3.4 SUMMARY	60
4.0 REDUCED MODEL PREDICTION	62
4.1 PREDICTION (FINDING MAXIMA)	63
4.2 LEAST SQUARES METHOD	67
4.3 COMPARISON OF LINEAR AND LEAST SQUARES MODELS.....	69
4.4 INITIAL CONDITIONS	71
4.5 PREDICTION OF ALTITUDE LOSS	73
4.6 SUMMARY	74
5.0 SIMULATION EVALUATION OF CVA	75
5.1 CVA MODEL STRUCTURE	76
5.2 DESCRIPTION OF ADS-33C AGGRESSIVE MANEUVERS	77
5.2.1 <i>Acceleration and Deceleration</i>	78
5.2.2 <i>Rapid Sidestep</i>	79
5.2.3 <i>Rapid Slalom</i>	81
5.3 SUMMARY	82
6.0 ASRA CVA IMPLEMENTATION ISSUES	83
6.1 HARD HOVER ENVELOPE LIMITS	84
6.2 UP AND AWAY FLIGHT ENVELOPE.....	86
6.3 PILOT WARNING SYSTEM	86
6.4 CVA TEST AND VERIFICATION PROCEDURE	87
6.5 FEEDBACK SIGNAL FILTERING	89
6.6 CODE STRUCTURE AND FLOW	90
6.7 COMPUTATIONAL SAFETY	93
7.0 CONCLUSIONS AND RECOMMENDATIONS.....	94
7.1 CONCLUSIONS.....	95
7.2 BENEFITS OF DIGITAL CVA.....	97
7.2.1 <i>Elimination of Control System Rate Limits</i>	97
7.2.2 <i>Elimination of Rate Trip Limits</i>	98

7.2.3 <i>Simplicity</i>	99
7.2.4 <i>Description of Flight Envelope</i>	99
7.3 SHORTCOMINGS OF OPEN LOOP MODEL BASED CVA	100
7.4 POSSIBLE IMPROVEMENTS TO CVA	101
8.0 REFERENCES	101
APPENDIX A	102
OPEN LOOP MODEL DETAILS	102
PILOT MODEL DETAILS	104
APPENDIX B	112
APPENDIX C	124

LIST OF FIGURES

<u>FIGURE 1: ASRA BELL 412 HP</u>	3
<u>FIGURE 2: BASIC SAFETY SYSTEM STRUCTURE</u>	5
<u>FIGURE 3: FBW FAULT TREE</u>	15
<u>FIGURE 4: CVA FAULT TREE</u>	16
<u>FIGURE 5: EARTH FIXED AXES</u>	27
<u>FIGURE 6: AIRCRAFT FIXED AXES</u>	28
<u>FIGURE 7: 3-2-1-1 CONTROL INPUT</u>	33
<u>FIGURE 8: TYPICAL CONTROL INPUT SEQUENCE</u>	44
<u>FIGURE 9: LONGITUDINAL CYCLIC HARDOVER WITH VARIATION IN PITCH INITIAL CONDITION</u>	48
<u>FIGURE 10: YAW RATE RESPONSE DUE TO LONGITUDINAL CYCLIC INPUT</u>	51
<u>FIGURE 11: THE EFFECT OF VARYING HARDOVER DURATION (LONGITUDINAL CYCLIC -100%)</u>	52
<u>FIGURE 12: PILOT MODEL BLOCK DIAGRAM</u>	54
<u>FIGURE 13: MAXIMUM FINDING ALGORITHM</u>	57
<u>FIGURE 14: INPUT SEQUENCE</u>	63
<u>FIGURE 15: PITCH ANGLE PARAMETERS VS. FORWARD AIRSPEED</u>	69
<u>FIGURE 16: ROLL ANGLE PARAMETERS VS. FORWARD AIRSPEED</u>	69
<u>FIGURE 17: DISCRETE DIFFERENTIATOR</u>	71
<u>FIGURE 18: OVERALL CVA AND BELL 412 BLOCK DIAGRAM</u>	75
<u>FIGURE 19: BELL 412 CRITICAL DIMENSIONS</u>	83
<u>FIGURE 20: ABSOLUTE ROLL LIMIT VS. ROTOR HUB HEIGHT</u>	84
<u>FIGURE 21: ABSOLUTE PITCH LIMIT VS. ROTOR HUB HEIGHT</u>	84
<u>FIGURE 22: BELL 212 RADALT AND PRESSURE ALTITUDE TRACE</u>	88
<u>FIGURE 23: CVA FLOW CHART</u>	90
<u>FIGURE A1: OPEN LOOP BLOCK DIAGRAM</u>	108
<u>FIGURE A2: BELL 412 REV BLOCK DIAGRAM</u>	109
<u>FIGURE A3: EULER ANGLES BLOCK DIAGRAM</u>	110
<u>FIGURE A4: X EULER EQUATION BLOCK DIAGRAM</u>	111
<u>FIGURE A5: Y EULER EQUATION BLOCK DIAGRAM</u>	112
<u>FIGURE A6: Z EULER EQUATION BLOCK DIAGRAM</u>	113
<u>FIGURE A7: OVERALL PILOTED BELL 412 BLOCK DIAGRAM</u>	114
<u>FIGURE A8: PILOT MODEL BLOCK DIAGRAM</u>	115
<u>FIGURE B1: HOVER LONGITUDINAL CYCLIC</u>	117
<u>FIGURE B2: HOVER LATERAL CYCLIC</u>	118
<u>FIGURE B3: HOVER TAIL ROTOR</u>	119

<u>FIGURE B4: HOVER COLLECTIVE</u>	120
<u>FIGURE B5: 60 KNOTS LONGITUDINAL CYCLIC.....</u>	121
<u>FIGURE B6: 60 KNOTS LATERAL CYCLIC.....</u>	122
<u>FIGURE B7: 60 KNOTS TAIL ROTOR</u>	123
<u>FIGURE B8: 60 KNOTS COLLECTIVE</u>	124
<u>FIGURE B9: 120 KNOTS LONGITUDINAL CYCLIC.....</u>	125
<u>FIGURE B10: 120 KNOTS LATERAL CYCLIC</u>	126
<u>FIGURE B11: 120 KNOTS TAIL ROTOR</u>	127
<u>FIGURE C1: QUICK STOP CASE 1</u>	130
<u>FIGURE C2: QUICK STOP CASE 2</u>	131
<u>FIGURE C3: QUICK STOP CASE 2, 2 AXIS FAILURE</u>	132
<u>FIGURE C4: RAPID SIDESTEP CASE 1, CVA ENGAGED</u>	133
<u>FIGURE C5: RAPID SIDESTEP CASE 1, CVA DISENGAGED</u>	134
<u>FIGURE C6: RAPID SIDESTEP CASE 2</u>	135
<u>FIGURE C7: RAPID SLALOM CASE 1, CVA ENGAGED</u>	136
<u>FIGURE C8: RAPID SLALOM CASE 1, CVA DISENGAGED</u>	137
<u>FIGURE C9: RAPID SLALOM CASE 2, CVA ENGAGED</u>	131
<u>FIGURE C10: RAPID SLALOM CASE 2, CVA DISENGAGED.....</u>	132

LIST OF TABLES

<u>TABLE 1: 60 KNOT 8 DOF STABILITY AND CONTROL DERIVATIVES</u>	35
<u>TABLE 2: 120 KNOTS 8 DOF STABILITY AND CONTROL DERIVATIVES</u>	36
<u>TABLE 3: HOVER 8 DOF STABILITY AND CONTROL DERIVATIVES</u>	36
<u>TABLE 4: INITIAL CONDITIONS INVESTIGATED</u>	47
<u>TABLE 5: LONGITUDINAL ACTUATOR HARDOVER DURING SLALOM</u>	50
<u>TABLE 6: LATERAL ACTUATOR HARDOVER DURING SLALOM</u>	51
<u>TABLE 7: COMPARISON OF RESULTS</u>	52
<u>TABLE 8: LINEAR AND LEAST SQUARES REDUCED ORDER MODEL PARAMETERS</u>	69
<u>TABLE 9: QUICK STOP INITIAL CONDITIONS</u>	78
<u>TABLE 10: RAPID SIDESTEP INITIAL CONDITIONS.....</u>	80
<u>TABLE 11: RAPID SLALOM INITIAL CONDITIONS.....</u>	81

NOTATION

Variables

A	State Dynamics Matrix
B	State Control Matrix
C	State Observation Matrix
x	State Vector
h	Height Rate
δe	Longitudinal Cyclic Displacement
δcol	Collective Displacement
δa	Lateral Cyclic Displacement
δr	Pedals Displacement
u, v, w	Aircraft Velocities (In x, y, z directions respectively)
p	Roll Rate
q	Pitch Rate
r	Yaw Rate
a_l	Longitudinal Flap Angle
b_l	Lateral Flap Angle
X, Y, Z	Aerodynamic Forces (In x, y, z, directions respectively)
ϕ	Roll Angle
θ	Pitch Angle
γ	Yaw Angle, or Blade Lock Number (where indicated)
Ω	Rotor Angular Velocity
L, M, N	Aerodynamic Moments (In pitch, roll, and yaw axes)
τ_L	Lead Time Constant
τ_l	Lag Time Constant
τ_N	Neuromuscular Lag
τ_e	Effective Time Delay
s	Laplace Operator

H	Transfer Function Matrix
$\mathcal{L}$	Laplace Transform
$\% act$	Percentage of Actuator Displacement

Subscripts

0	Initial Value
u, v, w	Aircraft Body Perturbation Velocities
p, q, r	Aircraft Body Perturbation Angular Rates
al, bl	Rotor Flap States
i, j	Indices to a Matrix
max	Maximum Excursion Value
δa	Lateral Cyclic
δe	Longitudinal Cyclic
δcol	Collective
δr	Pedals
$curr$	Current Value (As measured)

Designations, and Abbreviations

ASRA	Advanced Systems Research Aircraft
CVA	Command Validation Algorithm
DAT	Digital Audio Tape
FBW	Fly-By-Wire
FCC	Flight Control Computer
FDI	Failure Detection and Identification
FRL	Flight Research Laboratory
GLR	Generalized Least Squares

HMU	Health Monitoring Unit
IAR	Institute for Aerospace Research
MMLE	Modified Maximum Likelihood Estimation
NASA	National Aeronautics and Space Administration
NRC	National Research Council
RADALT	Radar Altimeter
SAS	Stability Augmentation System
TPP	Tip Path Plane
VSRA	Vertical/Short Takeoff and Landing Research Aircraft
VTOL	Vertical Takeoff and Landing

INTRODUCTION

The Flight Research Laboratory (FRL) of the Institute for Aerospace Research and Development (IAR) is in the process of developing an Advanced Systems Research Aircraft (ASRA) based on a Bell 412HP helicopter. This thesis describes a method for ensuring that commands from ASRA's flight control computer are valid commands and not the result of potentially dangerous software errors. The command validation algorithm monitors the commands to the aircraft's actuators and the current state of the aircraft (position, altitude, and body rates) to determine if the current command is valid or, instead, could place the aircraft in severe peril.

The FRL has operated variable stability helicopters in the airborne simulation and systems development modes for nearly three decades. The current working aircraft, the Airborne Simulator, based on a Bell 205 A-1, is the third generation of such machine developed at the laboratory. A generalized approach to fly-by-wire has always been adopted by FRL, with the aircraft not married to a specific control system architecture. The Airborne Simulator consists of the host aircraft, a set of dual mode, full authority actuators, a general purpose computing system, a set of state and pilot input sensors and a variety of pilot displays. The lack of embedded software and dedicated fly-by-wire control system has left the laboratory free to adopt control system structures appropriate to the problem at hand. The Airborne Simulator is flown by two pilots; the evaluation pilot controls the aircraft in the fly-by-wire mode,

and the safety pilot has the ability to fly the aircraft via the standard mechanical connections.

The Bell 205 Airborne Simulator has always been driven by full authority high bandwidth actuators which, in a single channel system, have always left the aircraft potentially vulnerable to a full deflection hardover in the fly-by-wire mode. The FRL has relied on hardware health monitoring, and software safety related modules to assist the safety pilot in preventing catastrophic results from such an occurrence. It has been the experience of the laboratory that the majority of hardover failures (or unplanned step displacements) of the controls in the Airborne Simulator were commanded by the flight control computing system.

Three main factors contribute to FRL's confidence in the safety pilot's ability to ensure the aircraft's integrity under failure conditions.

1. All actuator commands generated by the fly-by-wire system are reflected back to the safety pilot's controls. This implies that all action must be in parallel with the safety pilot's controls.
2. The safety pilot remains always hands on and is provided with a distinctive fly-by-wire disable/disengage control in the form of a paddle switch mounted on the cyclic such that it can be activated quickly. The safety pilot quickly learns what is the norm for any given control system and tends to react very quickly when the perceived pattern changes for no clear reason.

3. The Bell 205 has a comprehensive health monitoring system that continually examines the states of electrical power supplies and hydraulic pressure at the actuators and disengages the fly-by-wire system at the moment a fault is detected.

Figure 1: ASRA Bell 412 HP

Despite the excellent record of safety experienced in the Bell 205, it is considered that an identical system would provide an inadequate safety margin were it to be fitted to the ASRA. The Bell *Soft In Plane* rotor system of the 412 displays appreciably higher control power and considerably less response delay than the teetering system present in the 205 (approximately 72 ms for the Bell 412 compared to 150 ms in the Bell 205). For this reason it is felt that a method must be found to limit the aircraft's response to hardover. Several options have been examined, including:

- shaped pilot valve porting to tailor the frequency/amplitude response of a single actuator,

- the use of multiple actuators per channel, and
- the use of software monitoring of the actuators to inhibit fast, large amplitude responses.

The first option, the more elegant and simpler to install, had to be abandoned because of funding limitations encountered by the NRC. Alternatively, the compound actuator proved to be too complex and space limitations in the Bell 412HP prevented their use. As for the remaining potential solution, first it was felt that the use of software monitoring would implicitly rate limit the actuators, thereby distorting the simulation responses of the aircraft. Through the use of an intelligent monitoring system it is possible to protect the flight envelope of the Bell 412 without limiting its performance. This system reviews the commands generated by the flight control computer, and determines, based on the magnitude and direction of the command and the current position of the aircraft, the validity of the command. Since the output of this 'command validation' is binary (i.e.: valid or invalid command) the software does not directly act upon the FCC commands, and is incapable of changing them. The structure of this system is shown in figure 2. The dotted lines represent the effective feedback connections when there is a 'pilot in the loop'. These connections represent the numerous visual, aural, and motion cues presented to the pilot.

Figure 2: Basic safety system structure

1.1 PREVIOUS RESEARCH

Aircraft that depend upon flight control computers to maintain either safe operation, or adequate handling qualities employ redundant computation with dissimilar software encoding. Computer hardware and software failures may be detected in these arrangements by comparing the outputs of a selected number of computers (usually three or four) and voting to determine the correct value. A typical arrangement for a system of this type is described by Ammons¹.

To achieve the integrity and reliability required the techniques adopted are based on replication of the basic computing task to form redundant computing lanes. Inter-lane redundancy management, based on output commands comparison, is then used to isolate the failed lane by a majority decision. Thus, in the general case, by adoption of

this philosophy and if the system degrades gracefully, $N - 2$ failures can be survived for an N lane system.

The use of dissimilarity in hardware and software in redundant systems, has been previously successfully employed to avoid generic failures. The benefits of this approach are based on the assumption that generic failures will occur at random and will be unrelated, thus the probability of two or more versions failing virtually simultaneously in a like manner will be extremely low. Examples of dissimilar hardware and software implementation can be found in the Airbus A310 and A320 secondary flight control systems^{2,3}.

The RTCA, an association of government and industry aeronautical organizations in the U. S., seeks technical solutions to problems involving the application of electronics and telecommunications to aeronautical operations. Safety monitoring is described in an RTCA report⁴ as a means of protecting against specific failure conditions by directly monitoring a function for failures which would contribute to the failure condition. The monitoring functions may be implemented in hardware, software, or a combination of the two.

Typically research aircraft (such as the Bell 205 Airborne Simulator) are modified production aircraft that have a single flight control computer owing to economic, size, and maintainability constraints. For these aircraft to be protected against computer

failures, safety is provided through a combination of a safety pilot, to monitor back-driven controls, and some basic electronic monitoring.

Through the use of monitoring techniques, the software level (RTCA defines software level based on the danger a software failure would pose to the aircraft) of the monitored function, in this case the FCC, may be reduced to the level associated with the loss of its related system function. To allow this reduction, there are three important attributes of the monitor that should be determined:

- Software level: Safety monitoring software is assigned the software level associated with the most severe failure condition for the monitored function.
- System fault coverage: Assessment of the system fault coverage ensures that the monitor's design and implementation are such that the faults it is intended to detect will be detected under all necessary conditions.
- Independence of Function and Monitor: The monitor and protective mechanism are not rendered inoperative by the same failure condition that causes the hazard.

A considerable amount of work has been done in the area of failure detection and identification in dynamic systems, and Willsky⁵ has provided a well-known survey of many of the available FDI techniques, including:

- Failure sensitive filters
- Voting systems

- Multiple hypothesis filter-detectors
- Jump process formulations
- Innovations based detection systems

Willsky identifies the need to take the degree of system redundancy into account, stating that, in a system containing several back-up subsystems, it may be possible to devise a simple detection algorithm that is easily implemented but yields moderate false alarm rates. The safety systems currently in place on the ASRA, including safety-pilot and health monitoring units, should allow the command validation algorithm to be simple. It will only be required to prevent sudden controller faults that pose potential safety risks. The argument for a simple algorithm is supported by the issue of computational complexity; the system should not impose excessive time delays due to algorithm complexity.

The detection of failures in sensors has been examined by a number of authors^{6,7,8,9}. Typically these failures are detected through the use of 'analytical redundancy'. In contrast to hardware redundancy (multiple copies of sensors and actuators) analytical redundancy exploits the relationship between different variables in a dynamic system to allow different sensors (or actuators) to serve as backups to each other. The generalized likelihood ratio has been investigated in several references^{10,11}. The GLR approach makes an attempt to isolate failures by using knowledge of the different effects such failures have on the system innovations. This technique has been

exercised in a simplified simulation of the F-8 aircraft dynamics¹², and in a linear simulation of the Boeing 737 aircraft longitudinal dynamics. Another FDI technique is the failure detection filter, developed by Beard¹³ for linear deterministic continuous systems.

Most techniques for failure detection make use of an observer based approach to perform state estimation. The optimal state estimator, if no failures occur, is given by the discrete Kalman filter equations¹⁴. It is possible for the filter estimate to diverge if there are substantial unmodeled phenomena. The problem occurs because the filter 'learns the state too well', i.e. the pre-computed error covariance P and filter gain K become small, and the filter relies on old measurements for its estimates and is oblivious to new measurements. Thus if an abrupt change occurs (for example a commanded hardover), the filter will respond quite sluggishly, yielding poor performance. Several techniques for addressing this problem have been developed, including exponentially age weighted filters¹⁵, and limited memory filters¹⁶. These methods, however, introduce a performance tradeoff. As the sensitivity to new data is increased the system becomes more sensitive to noise, and the performance of the filter in the 'no-failure' condition.

Typically FDI techniques are indirect. Several methods have been developed for the design of filters that are sensitive to specific failures. One method involves the inclusion of several 'failure states' in the dynamic model. Kerr¹⁷ has considered a

procedure in which failure modes, such as the onset of biases, are included as state variables. If the estimate of these variables vary markedly from their nominal values, a failure is declared.

Chow and Willsky¹⁸ have examined the problem of generating residuals from the system measurement data for use in decision-making processes to detect and identify failures. System fault coverage is typically ensured by modeling the monitored function, and comparing the actual output with the model. Examples of this include VSRA, NASA's Vertical/Short Takeoff and Landing Research Aircraft. Schroeder^{19,20} et. al. demonstrated a model following command validation algorithm for a YAV-8B Harrier jet.

In the VSRA control system, sensor and computer command failures are detected in a servo control unit whose prime function is to route commands from the primary flight computer to the appropriate servos. Using end-around and in-line techniques, sensor and servo failures can be detected and isolated in less than two computer cycles. As with the ASRA, a system had to be developed to detect command failures. The primary difference (from a command validation design standpoint) between the VSRA and ASRA is that the VSRA has only one pilot, therefore short term command failures and long term command failures must be detected. Schroeder et. al. examined several control command monitoring approaches including:

- Monitor duration of servo saturation,

- Flight envelope monitoring,
- Observer based, model following.

A simple check for hardover failures is to examine how long a given servo has saturated. However, saturation of many of the research system's series servos occurs in normal operation during transition and landing. The saturation time permitted before declaring a failure must be set small enough to catch the effects of a hardover failure in time for the pilot to make a safe recovery. This method was only successful for the pitch and roll axes. Other axes would occasionally have longer series-servo saturation times than could be allowed for in an adequate failure detection time. A problem with this type of monitoring scheme is that it is only applicable to hardovers. The scheme is totally ineffective for slow failures and in particular for failures that cause the servo to freeze at a given position.

NASA's flight envelope monitoring scheme involved the use of a linear functional of the states and the state rates to define a 'normal' operating flight envelope. If the functional exceeds a preset value, it is assumed that a failure has taken place, and control reverts back to the standard mechanical system. For example, for height control in hover it is reasonable to use the weighted sum:

$$K_1 \dot{h} + K_2 \ddot{h} \tag{1}$$

to monitor the vertical axis. The conjecture is that if the sum exceeds an envelope limit as a function of altitude, a failure has probably occurred. The problem with this

scheme is that failures can only be detected quickly enough if the functional is made heavily dependent on measured acceleration. Unfortunately such a functional tends to provide a very restrictive operational envelope and is sensitive to disturbances and sensor noise.

The monitoring system that was finally used is based on a comparison of the response of a dynamic model of the aircraft for a given pilot input to the response of the aircraft itself. If the difference exceeds a preset value, then, for whatever reason, the aircraft is not following the desired dynamics and a failure is assumed to have occurred. Since the implicit model-following system is self-trimming, no steady state bias exists between the command and the desired aircraft motion. Thus, the desired model outputs do not require adjustment to account for a steady state error buildup, and the model following error when tested against the preset values will not be contaminated with a steady state error. Both of these results minimize nuisance disconnects. The primary advantages inferred for this "model comparator" monitor are 1) independence of the measured aircraft acceleration, with less sensitivity to turbulence, hence tighter detection times; and 2) slow failures should be detected quickly, since the model comparators use the pilot's input, as well as the aircraft's state to determine the system's integrity. A potential disadvantage is that a change in the desired response will require a change to the monitor software, which violates the monitor's independence, and may compromise its integrity.

Isermann²¹ distinguishes the following functions of a supervisory system:

- *Monitoring:* measurable variables are checked with regard to tolerances, and alarms are generated for the operator.
- *Automatic protection:* in the case of a dangerous process state, the monitoring function automatically initiates an appropriate counteraction.
- *Supervision with fault diagnosis:* based on measured variables, features are calculated, symptoms are generated via change detection, a fault diagnosis is performed and decisions for counteractions are made.

Monitoring, and automatic protection are considered classical methods. Isermann states that in the case of closed loops, changes in the process are covered by control actions and cannot be detected from the output signals, as long as the manipulated inputs remain in the normal range. Therefore, feedback systems hinder the early detection of process faults. Two points are worth mentioning; if a failure occurs (we are concerned with controller failures), and does not significantly affect the output or the controller inputs, then it can hardly be classified as a true ‘failure’; secondly, Isermann makes no mention of directly monitoring the controller output to prevent potentially hazardous control motions from being input.

1.2 THE ROLE OF CVA

A broad range of systems are employed to secure the safety of research fly-by-wire aircraft. The command validation algorithm is a critical component of the ASRA’s

health monitoring unit (HMU). The HMU consists of all devices whose purpose is to prevent failures from endangering the aircraft. A fault tree for the HMU is shown in figure 3.

There are numerous safety components in place on the ASRA to protect the aircraft and its crew from failures. The fault tree shows the paths by which failures can affect the state of the aircraft, and the safety systems in place to prevent them.

From the tree it can be seen that a FCC commanded failure only threatens the aircraft if the following events all occur:

1. The CVA fails to disengage the fly-by-wire system given a hazardous FCC command.
2. The pilot recognizes the command failure, but the FBW trip switch fails, or the pilot does not recognize the command failure.
3. The mechanical override fails to operate.

Figure 3: FBW fault tree

Figure 4 shows the fault tree for the command validation algorithm. The tree shows all paths by which the CVA would not act upon a hazardous condition. Failure of the FBW trip circuitry is averted through the use of redundant reliable hardware. In the

event of instrumentation failure, the CVA may receive incorrect aircraft state information. To guard against such failures, the instruments make extensive use of built-in test equipment and health and usage monitoring.

Figure 4: CVA fault tree

The aim of this thesis is to demonstrate a simple algorithm that is capable of reliably determining the validity of commands from the FCC. A CVA failure would result in a command reaching the FBW actuators which could potentially endanger the aircraft.

Since the aircraft is piloted by both an evaluation pilot and a safety pilot the command validation algorithm need only be concerned with the effects of FCC commands over a time window of approximately 500 milliseconds. The safety pilot can detect command failures enduring greater than their neuromuscular delay (a motivated pilot may have an effective delay time in the range of 175-500 milliseconds²²). Since all commands are reflected back to their controls, the safety pilot is able to quickly determine the validity of the command, and apply corrective action if necessary. It is only necessary to monitor those commands that can place the aircraft in peril before the safety pilot can determine that there has been a failure and apply corrective action.

The bulk of FCC failures consist of undesired step commands (usually the result of software errors), thus it is logical to examine the step response of the Bell 412 in order to gain an understanding of the possible effects of actuator steps. To evaluate its step response it is necessary to have a good dynamic model of the Bell 412. The FRL has numerous mathematical models of the Bell 412, but their validity had to be investigated. Once the models are validated, the algorithm can be developed.

1.3 OVERVIEW OF THESIS

This thesis is organized into seven chapters, each outlining a fundamental step in the development of the command validation algorithm for the ASRA Bell 412. The ordering of the chapters follows the steps involved in the development and implementation of the CVA for ASRA.

Chapter 2 examines the process of verification of the Bell 412 dynamic models. Since the CVA is required to prevent catastrophic FCC commands from reaching the actuators, it is necessary to use some form of simulation to determine the ASRA's response. Dynamic models have been developed at the FRL, however they are limited to the FORTRAN environment. In order to take advantage of the power and portability of MATLAB it was required to convert the model from FORTRAN code into a SIMULINK block diagram. In order to verify the conversion process, the models were evaluated and time responses were compared to flight test data.

Chapter 3 details the results and procedure of the analysis of the Bell 412's step response. Since the 412 is a multivariable, non-linear, and highly coupled system, it does not have a step response in the classical sense. The basic controls of a helicopter affect rate values (e.g.: moving the stick sideways produces an increase in roll rate) which in turn affect attitudes. The non-linearities of the model are channeled through the attitudes via trigonometric terms. As a result of this, initial conditions figure heavily into the step response. Pure step response is not a particularly useful characteristic for the evaluation of the Bell 412's hardover response (or that of any aircraft for that matter) since there are two distinct components to a hardover response; the FCC commanded hardover, and the pilot's recognition and response to the hardover. In order to investigate this, an input signal was developed, based upon a doublet (or pulse), that mimicked pilot response to an FCC commanded hardover.

Through the use of this input sequence it was possible to quantify the step response. The Bell 412 model was also examined to determine the degree of non-linearity with respect to initial conditions and various magnitudes of input.

Chapter 4 presents the command validation algorithm proposed for ASRA. The derivation of a reduced order prediction scheme is presented and contrasted with least squares look up table values. The primary issues addressed include the prediction of attitude excursions and altitude loss.

Chapter 5 illustrates a simulation evaluation of CVA performance over a wide range of operating conditions and actuator hardover magnitudes. Most of the initial conditions were taken from flight test data of aggressive maneuvers within the ADS-33C²³ specification.

Chapter 6 introduces some of the implementation issues likely to be encountered before the algorithm is flown on the ASRA. The flight envelope limits are discussed, as well as the C code structure and flow. Another important issue is the concept of the complimentary filtering of feedback signals. This will be necessary, especially for the altitude channel.

Recommendations and conclusions are then presented in chapter 7.

1.4 SCOPE OF WORK

This thesis documents a method for FCC command validation for single string safety piloted fly-by-wire aircraft. The approach to command validation is limited to aircraft with mechanical reversion, and particularly suited to reconfigurable flight control systems. The thesis covers the theory behind the CVA, and documents a procedure for its testing. Unfortunately, time limitations prevent the thesis from describing the actual application of the CVA on ASRA.

Chapter 2

BELL 412 MODEL STRUCTURE AND VERIFICATION

2.1 INTRODUCTION

The Flight Research Laboratory has developed mathematical models of the Bell 412HP for various trim velocities based on project dedicated flight test data. The models²⁴ were developed using the NRC modified version of NASA's MMLE3 program, a time-domain parameter identification routine. Typically helicopters are modeled through the use of a six degree-of-freedom rigid body fuselage model, however since the Bell 412 rotor system imparts substantial moments to the fuselage, it was decided to use a hybrid eight degree-of-freedom model that incorporated rotor flapping effects. The flight mechanics were modeled using SIMULINK to solve the simultaneous non-linear equations of motion.

This chapter presents the structure of the flight mechanics model used to simulate the behavior of the Bell 412HP under various actuator hardover cases. The validation of the model was accomplished by supplying the model 3-2-1-1 inputs (pulse inputs in time ratio of 3:2:1:1) recorded during the evaluation flights of the Bell 412HP and comparing the time histories. The results of that validation are presented as well.

2.2 HELICOPTER FLIGHT MECHANICS

Stability and control are among the most important aspects of the analysis and design of rotary-wing aircraft. As with the airplane, the problem of controlling the vehicle was one of the major obstacles in the development of a successful helicopter. Designing for satisfactory flying qualities remains a major concern in the development of a helicopter with new applications of the vehicle always demanding improved behavior.

Helicopter control requires the ability to produce moments and forces on the vehicle for two purposes: first, to produce equilibrium and thereby hold the helicopter in a desired trim state; and secondly, to produce accelerations and thereby change the helicopter velocity, position, and orientation. Like airplane control, helicopter control is accomplished primarily by producing moments about all three aircraft axes: pitch, roll and yaw. The helicopter has in addition direct control over the vertical force on the aircraft, corresponding to its vertical take-off and landing capability. This

additional control variable is part of the versatility of the helicopter, but it also makes the piloting task more difficult. Usually the control task is eased by the use of a rotor speed governor on the engine throttle to automatically manage the power.

Direct control over moments on the aircraft is satisfactory for trajectory control in forward flight. In hover and at low speed, direct control over the forces would be more desirable, in order to obtain direct command of the helicopter velocity and displacement. Such control is available only for the vertical force, however. The lateral and longitudinal velocities of the helicopter in hover must be controlled using pitch and roll moments about the aircraft center of gravity, which is a more difficult task. The pilot directly commands a change in pitch or roll rate that then produces a longitudinal or lateral force and finally the desired velocity of the helicopter. There usually is significant coupling of the forces and moments produced by the helicopter controls, so that any control application to produce a particular moment will require some compensating control inputs on the other axes as well. Moreover, without an automatic stability augmentation system (SAS), the helicopter is not dynamically or statically stable, particularly in hover. Consequently, the pilot is required to provide the feedback control to stabilize the vehicle, an operation that demands constant attention. The use of an automatic control system to augment the helicopter stability and control characteristics is desirable, and for some applications essential, but such systems increase the cost and complexity of the aircraft.

The rotor is almost universally used to control the helicopter. In forward flight, fixed aerodynamic surfaces such as horizontal stabilizer and elevator surfaces may be used as well.

In forward flight, the dynamics and control of rotary-wing aircraft are similar to those of fixed-wing aircraft. The rotor is, in effect, an augmented wing with a circular platform. However, forward speed is limited by the stalling of the retreating blades and compressibility effects on the advancing blades. The rotor induces severe vibration on the fuselage in forward flight, which is very fatiguing for the crew and passengers. Dynamic modeling is complicated by several effects, such as the impingement of blade vortex-wakes on the other blades, and the flexibility of the rotor blades.

Near hover, the dynamics and control of rotary-wing aircraft are significantly different from those of fixed wing aircraft. The differences are caused by the gyroscopic and torquing effects associated with the rotor, which introduces significant coupling of the lateral and longitudinal motions.

Typically dynamic models of helicopters approximate the fuselage as a rigid body and the rotor as a set of blades of negligible inertia. Thus the rotor tip path plane (TPP) can be tilted “instantaneously” by cyclic pitch changes, and the rotor thrust can be changed instantaneously by collective pitch changes. Tilt of the TPP with respect to

the fuselage provides pitching and rolling moments on the fuselage, since then the thrust axis does not pass through the center of mass. Tilt of the TPP with respect to inertial space provides components of thrust for maneuvering in the horizontal direction, while increasing or decreasing thrust moves the vehicle vertically.

Model fidelity can be increased through the addition of rotor flapping states. The motion of a hinged blade consists basically of rigid body rotation about each hinge. Motion about the hinge lying in the rotor disk plane (and perpendicular to the blade radial direction) produces an out of plane deflection of the blade and is called flap motion.

The mechanical arrangement of the rotor hub to accommodate flap and lag motion of the blade provides a fundamental classification of rotor types as follows:

1. *Articulated rotor*. The blades are attached to the hub with flap and lag hinges
2. *Teetering rotor*. Two blades forming a continuous structure are attached to the rotor shaft with a single flap hinge in a teetering or seesaw arrangement. The rotor has no lag hinges.
3. *Hingeless rotor*. The blades are attached to the hub without flap or lag hinges, although often with a feathering bearing or hinge. The blade is attached to the hub with cantilever root restraint, so that blade motion occurs through bending at the root. This rotor is also called a *rigid rotor*.

The Bell 205 is a teetering rotor helicopter, and as such the fuselage is not subject to large moments generated by the rotor. This facilitates the use of a standard six degree-of-freedom model. The Bell 412 rotor system, however, creates fuselage dynamics that cannot be described by the commonly used linear six degree-of-freedom approach.

2.3 MODEL STRUCTURE

This section is intended to give a general overview of the helicopter model used throughout the development of the CVA. Further detail can be found in appendix A.

Two axis systems are used to describe the dynamics of aircraft. The earth based system, whose origin is fixed at the center of the earth, is used primarily to express gravitational effects, altitude, horizontal distance, and the orientation of the aircraft. Figure 5 displays the set of earth axes used for CVA development. The axis X_E , is chosen to point north, the axis Y_E then pointing east with the orthogonal triad being completed with axis Z_E , pointing down.

Figure 5: Earth Fixed Axes

The aircraft itself has its own axis system: the body axis system, whose origin is fixed at the aircraft's center of gravity. Figure 6 shows the body axis system; in this system X points forward out of the nose of the aircraft, Y points out the starboard (right) side, and Z points down. The angular orientation of the body axis system with respect to the earth axis system depends strictly on the orientation sequence. This sequence is taken as follows:

1. Rotate the earth axes through some azimuthal angle, ψ , about the axis X_E , to reach some intermediate axes X_I, Y_I, Z_I .

2. Next, rotate these axes through an angle of elevation θ , about the axis Y_1 to reach a second, intermediate set of axes, X_2, Y_2, Z_2
3. Finally, the axes X_2, Y_2 , and Z_2 are rotated through an angle of bank, ϕ , about the axis X_2 , to reach body axes X, Y, Z .

The system, as utilized for the validation procedure, is an open loop model. The model consists of three principal components; the small perturbation equations of flight mechanics, extensions to include larger perturbations, and time delay models of actuators.

Figure 6: Aircraft fixed axes

The fundamental dynamics of the helicopter are governed by the following state space equation:

$$\begin{aligned}\dot{x} &= Ax + Bu_c \\ u_c &= [\delta e \quad \delta col \quad \delta a \quad \delta r] \\ x &= [u \quad v \quad w \quad p \quad q \quad r \quad a_1 \quad b_1]\end{aligned}\tag{2}$$

where A is the matrix of stability derivatives, and B is the matrix of control derivatives, u_c is the control vector, consisting of δe , longitudinal cyclic stick, δcol , collective displacement, δa , lateral cyclic stick, and δr , tail rotor (pedal) displacement. The state, x , can be eight dimensional or six dimensional depending on the degree of freedom of the model. The six degree of freedom model has the following state variables, u , v , w the perturbation velocities, and p , q , r the helicopter body rates. The eight degree of freedom model adds simplified rotor dynamics with longitudinal and lateral degrees of flapping (moments), a_1 and b_1 respectively.

Once the state derivative, $\dot{x}$, has been found from the small perturbation equations the values are proportional to the forces X , Y , Z , the moments L , M , N , and flapping rates $\dot{a}_1$ and $\dot{b}_1$. In order to account for gravity forces and Coriolis forces the following equations are employed:

$$\begin{aligned}\int \left(\dot{X} - 9.81 \sin \theta - qw + rv + \dot{u}_0 \right) dt &= u \\ \int \left(\dot{Y} + 9.81 \cos \theta \sin \phi + pw - ru + \dot{v}_0 \right) dt &= v \\ \int \left(\dot{Z} + 9.81 \cos \theta \cos \phi - pv + qu + \dot{w}_0 \right) dt &= w\end{aligned}\tag{3}$$

The initial accelerations are approximated as follows

$$\begin{aligned}\dot{u}_0 &= 9.81 \sin \theta_0 \\ \dot{v}_0 &= -9.81 \sin \phi_0 \cos \theta_0 \\ \dot{w}_0 &= -9.81 \cos \theta_0 \cos \phi_0\end{aligned}\tag{4}$$

In general, the angles Θ and Φ are not simply the integrals of the angular velocity p and q ; in effect, two new motion variables have been introduced and it is necessary to relate them to the angular velocities, p , q , r . The orientation of the aircraft, known as the Euler angles is calculated by the following equations:

$$\begin{aligned}\phi &= \int (r \cos \phi \tan \theta + p + q \sin \phi \tan \theta) dt + \phi_0 \\ \theta &= \int (q \cos \phi - r \sin \phi) dt + \theta_0 \\ \psi &= \int (q \sin \phi + r \cos \phi) \sec \theta dt + \psi_0\end{aligned}\tag{5}$$

To perform parameter estimation using MMLE3 and the hybrid model formulation, the state equations and the observation equations require rotor state information. The original flight test data included no such measurements, and thus a simplified model was used to generate the rotor flapping responses from existing measurements. The simplified differential equations for the longitudinal flapping a_l and lateral flapping b_l are:

$$\begin{bmatrix} \ddot{a}_1 \\ \ddot{b}_1 \end{bmatrix} + \Omega \begin{bmatrix} \gamma/8 & 2 \\ -2 & \gamma/8 \end{bmatrix} \begin{bmatrix} \dot{a}_1 \\ \dot{b}_1 \end{bmatrix} + \Omega^2 \begin{bmatrix} 0 & \gamma/8 \\ \gamma/8 & 0 \end{bmatrix} \begin{bmatrix} a_1 \\ b_1 \end{bmatrix} = \Omega^2 \begin{bmatrix} A1\gamma/8 - 2p/\Omega - \gamma q/8\Omega \\ B1\gamma/8 + 2q/\Omega - \gamma p/8\Omega \end{bmatrix} \quad (6)$$

Where: A1 is the lateral cyclic pitch measured from the hub plane and wind-hub system,

B1 is the longitudinal cyclic pitch measured from the hub plane and wind hub system, and

Ω is the main rotor rotational velocity in rads/s

The previous equation (6) forms a coupled set of differential equations with the damping matrix pre-multiplying the flapping rates $(\dot{a}_1, \dot{b}_1)$, and the stiffness matrix pre-multiplying the flapping angles (a_1, b_1) . Inspection of equation 6, with the appropriate values of $\gamma, \Omega, A1, B1, p$ and q substituted into the equation, shows that the flapping accelerations are relatively unimportant in describing low to mid frequency (up to approximately 20 rad/s) flapping. Consequently, the acceleration terms can be dropped and the rotor flapping equations become²⁴:

$$\begin{bmatrix} \dot{a}_1 \\ \dot{b}_1 \end{bmatrix} + \frac{\Omega}{\gamma^2/64 + 4} \begin{bmatrix} \gamma/4 & \gamma^2/64 \\ \gamma^2/64 & \gamma/4 \end{bmatrix} \begin{bmatrix} a_1 \\ b_1 \end{bmatrix} = \frac{\Omega}{\gamma^2/64 + 4} \begin{bmatrix} A1\gamma^2/64 - B1\gamma/4 \\ A1\gamma/4 + B1\gamma^2/64 \end{bmatrix} \quad (7)$$

For the Bell 412 HP, $\Omega=33.93$ rad/s and $\gamma=5.537$ (blade Lock number)

Equation 7 gives the longitudinal and lateral flapping angles as a function of measured body angular rates, the control inputs, rotor blade rotational speed, and

Lock number, an airfoil section property. The longitudinal and lateral flapping angle time histories were obtained by solving the simplified rotor dynamics equations. With the rotor flapping time histories solved it is possible to identify the rotor parameters in state space form using MMLE3, a time domain based maximum likelihood parameter estimation package.

2.4 PARAMETER IDENTIFICATION

In the fall of 1992, a flight test program²⁵ was conducted on the NRC Bell 412HP helicopter to obtain a set of data suitable for parameter estimation. In general, the maneuvers were performed at a pressure altitude of 2000 ft in calm conditions at a variety of forward airspeeds between hover and 120 knots.

In order to obtain the flight test maneuver time histories, the pilot first established the desired trim conditions. Then the pilot executed a 3-2-1-1 control input (this refers to an input train of alternate step control pulses in time sequence ratio 3,2,1,1 s; see figure 7) to excite the helicopter. The advantages of this input are:

- Adequate flat power spectral density over a wide-frequency bandwidth to excite all the characteristic modes of the aircraft;
- Sufficient high-frequency content, provided by the alternating “strokes” of the input, to improve the estimation of control derivatives;
- Short time of duration as compared to frequency sweep inputs; and

- Manually performed, easy to repeat, and non-critical in shape (i.e.: a “perfect” 3-2-1-1 is not necessary for parameter identification)

The maneuvers flown to identify the Bell 412 were modified 3-2-1-1's in that the magnitude of the control deflection was not constant. Since the 3-2-1-1 is not symmetric it results in a net change in control displacement which can place the helicopter in an undesirable position. To combat this the magnitude of the 3 and 2 portions are reduced. At the end of the maneuver the control input was left constant until the pilot needed to re-trim the aircraft for the next maneuver. If the helicopter was in a slightly unstable condition during this interim phase pulse type input in any axis may have been used to prevent the helicopter from deviating significantly from its trim state.

Figure 7: 3-2-1-1 Control Input

Owing to the simplicity and short duration of the 3-2-1-1 maneuver, tests could be carried out in a serial manner (trim – longitudinal cyclic input – trim – repeat – trim –

lateral cyclic input – trim – repeat – etc...). This process significantly reduced flight test time, reduced the required communication between pilots and flight test crew, and simplified flight test planning.

With the rotor explicitly modeled, many effects arising from the pitch and roll angular rates can be produced by either fuselage rolling moment derivatives (L_p) or rotor lateral moment derivatives (B_{bl}). Without the rotor measurements, the rotor flapping states are correlated with aircraft angular rates, and consequently MMLE3 cannot differentiate between these rotor and fuselage derivatives. Since the aerodynamics of a helicopter are dominated by the rotor it is evident that the fuselage derivatives should be small. Thus, at this stage of analysis, those fuselage derivatives (X_p , X_q , L_p , L_q , M_p , M_q) have been set to zero. Derivatives such as Y_p , and Y_q are not subject to this problem because the tail rotor produces additional aerodynamic effects, leading to dual coefficients that are possibly hard to separate. This argument for the L_p/B_{bl} derivative pairs also holds for the control derivatives (L_{da}/B_{da}). Again, the rotor control derivatives reflect the major effects, and therefore the fuselage control derivatives were also set to zero. A consequence of setting the fuselage derivatives (except for Z and N) to zero is that the rotor derivatives must absorb the small fuselage effects. Since the fuselage is asymmetrical (L_p is not equal to M_q), this decision causes the rotor longitudinal and lateral moment derivatives (A_{al} , B_{bl}) to be unequal.

The N_{bl} and N_{al} are included in the model to represent the effects of flapping on the directional response. This response is presumably related to changes in the main rotor torque. Because of the location of the tail rotor, the tail rotor side force produces a yawing moment; therefore, the N_p and N_q derivatives were retained in the model. Without the explicit modeling of the yawing moment due to engine governor, the N_{do} and the N_{de} derivatives were included in the model.

The model relies on fuselage derivatives only to represent the heave axis. Therefore, Z_{de} , Z_{da} , Z_p and Z_q remain the same form as in model-1, while Z_{al} and Z_{bl} are set to zero.

Table 1: 60 knot 8 DOF stability and control derivatives

60 knot Bell 412 HP stability and control derivatives												
	u	v	w	p	q	r	al	bl	lon	coll	lat	ped
X	-0.3849	-.0200	.08457	0.0	.2670	.2670	9.674	0.0	0.0	.08528	0.0	.08797
Y	.02327	-.1257	-.0381	-.1730	-1.656	-1.656	0.0	-46.84	0.0	.3166	0.0	-.5971
Z	-.09202	-.0219	-1.081	4.914	-1.064	-1.064	0.0	0.0	-2.076	-1.108	-.2399	.00598
L	-.01799	-.0372	.06616	0.0	-.4132	-.4132	-11.16	-13.08	0.0	.07735	0.0	.1911
M	.00047	.01245	-.0260	0.0	.05997	.05997	-4.464	.4745	0.0	.01025	0.0	.04983
N	-.02362	.03395	-.0840	.3890	-.4596	-.4596	14.88	75.50	.03281	-.0230	-3.394	-.5876
A	0.0	0.0	0.0	0.0	1.090	0.0	-18.14	-58.95	-.4484	0.0	2.558	0.0
B	0.0	0.0	0.0	-1.095	0.0	0.0	12.53	-11.03	.5657	0.0	0.3659	0.0

Table 2: 120 knots 8 DOF stability and control derivatives

120 knot Bell 412 HP stability and control derivatives												
	u	v	w	p	q	r	a1	b1	lon	coll	lat	Ped
X	-.1320	-.0045	.1558	0.0	.18	-.6128	3.218	0.0	0.0	.0422	0.0	.2577
Y	-.3361	-.1390	-.0185	-.09	0.0	2.261	-2.592	-5.422	0.0	-.1915	0.0	-.4010
Z	.4704	.0629	-1.094	6.996	5.847	4.959	0.0	0.0	-3.041	-.817	-.6104	.2628
L	0.0	.0313	-.005	-.5	-.013	1.381	-3.349	-2.64	0.0	-.028	0.0	.3851
M	-0.0749	-.0283	-.0351	-0.47	-1.488	-1.943	-5.398	19.96	.077	.021	-.921	-.1031
N	-.07457	.02577	-.0264	-.2	1.4	-2.422	-9.521	26.77	-.6582	-.0477	-1.082	-.6962
A	0.0	0.0	0.0	0.0	1.0	0.0	-13.68	-3.63	-.448	0.0	0.1266	0.0
B	0.0	0.0	0.0	-1.0	0.0	0.0	3.63	-10.50	.1550	0	.3659	0.0

Table 3: Hover 8 DOF stability and control derivatives

Hover Bell 412 HP stability and control derivatives												
	u	v	w	p	q	r	a1	b1	lon	coll	lat	Ped
X	-0.1168	0.0008	0.1624	1.031	-0.0453	0.2155	-19.07	0.0	-1.161	0.0823	0.1589	-0.0208
Y	-0.1645	-0.4267	-0.2775	-12.46	2.299	-1.682	28.74	-49.59	1.103	-.2727	-0.8156	0.6292
Z	-0.0011	-0.0823	-0.5665	0.1011	-0.2314	-0.7115	0.0	0.0	-0.25	-0.8206	-0.1108	0.0047
L	0.0427	-.1078	-0.0740	-0.5	1.464	-.3333	15.12	23.77	0.2925	-0.1025	-1.5540	0.2547
M	0.1662	-0.111	-0.0581	-8.688	-5.57	-1.057	-42.13	-52.3	-0.3842	-0.1649	0.6184	0.2831
N	0.2711	-0.2344	-0.1934	-20.66	-10.41	-2.485	-96.23	-136.7	-1.371	-0.2674	1.7960	0.1965
A	0	0	0	0	-1.0	0	-16.67	-3.63	-0.4484	0	0.1266	0
B	0	0	0	-1.0	0	0	3.63	-10.5	0.1150	0	0.3659	0

2.5 MODEL VERIFICATION

Verification of the Bell 412 models is accomplished through the examination of simulated time history data. Model verification has to be performed using different data to that used for identification. Two sets of 3-2-1-1 inputs were flown in each axis, and only one was used for identification. This left the remaining 3-2-1-1 sequence to be used for verification purposes (dubbed the 'verification 3-2-1-1'). It is possible to get relatively accurate results by simply executing the model with the input data supplied by the DAT file, but this does not take into account the various 'aerodynamic biases' present on the model. Maine and Iliff²⁶ refer to these biases as nuisance parameters since they are unknown parameters of little interest. In order to truly investigate the validity of the model they must be estimated. During the identification process the aerodynamic biases are tuned by the MMLE3 program, but this is a computationally costly procedure. For verification, however, the biases are estimated from the re-constructed flight path data (the re-constructed flight data is the result of removing measurement biases from the flight test data).

The validity of a model is a function of its purpose; for the purposes of the development of the CVA it is desirable to have a good match for a duration of about 2-3 seconds. The model should track the high frequencies well, but not at the expense of low frequency response. The model should match the on-axis response very well, and display a good representation of the off-axis responses. It is only necessary for the model to provide a good match for the first two or three seconds since it is only the

response of hardover failures that are to be simulated (The most severe 4 axis hardover encountered at the FRL had a duration of approximately 5 seconds²⁷).

Results of the Bell 412 model verification can be found in appendix B.

2.5.1 HOVER MODEL

The hover model shows adequate correspondence with the flight test data. Traditionally the hover case has been difficult to identify accurately without measurement and modeling of engine governor and rotor inflow effects.

Longitudinal Cyclic

The results of the longitudinal cyclic verification are presented in figure B1. The profile of the pitch rate response is followed, but there appears to be a parabolic bias. A similar bias is found in the roll rate response. This may be a result of the poor trim state in the 5 seconds prior to the 3-2-1-1 input. The angles appear to be well followed in spite of the rate biases.

Lateral Cyclic

As can be seen in figure B2, roll rate response is very well followed with the model appearing to have slightly more damping than the flight test results. Pitch rate response is well followed with only a slight bias evident in the final half of the

maneuver. The model's yaw rate response does not accurately agree with the flight test data, however it is representative of the general trend.

Tail Rotor

Figure B3 contains the results of the tail rotor verification of the hover model. The rate responses are well tracked, with the only discrepancy occurring on the first 1 of the 3-2-1-1 maneuver. This may be a result of the high amount of control activity at this point. On this step input the rate responses seem overly damped. The angle response, however, is very well matched.

Collective

The collective time histories for the hover model verification can be found in figure B4. Pitch and yaw rate response are well followed with the appearance of a first order bias which is likely a result of incorrect aerodynamic bias terms. Interestingly, the model shows there to be little effect of collective input at hover upon body rates and angles.

2.5.2 60 KNOTS MODEL

From the time histories it can be seen that the on-axis response is quite good, however the off axis responses don't match very well. The overall magnitude and direction of the off-axis responses is matched, but the phase seems to be off. This may have occurred since the test maneuvers were flown on a windy day. The high degree of pitch/roll cross coupling present in the 60 knot model is of interest. From the 3-2-1-1

responses it would appear that the aircraft is more highly coupled at 60 knots forward velocity than it is at hover.

Longitudinal Cyclic

As can be seen in figure B5, the pitch rate response of the model presents a good match with the test data. The model's roll rate response appears somewhat exaggerated however. The attitude response for pitch and roll is well followed although a noticeable bias occurs between the flight test pitch response and the model's pitch response. A sizeable offset is found in the model's yaw response. This is likely due to the integrated effect of yaw rate matching errors.

Lateral Cyclic

The model's rate response, as seen in figure B6, appears out of phase and over damped versus the flight test data. The attitude response is consistent with the effects of integrated rate errors.

Tail Rotor

Figure B7 contains the tail rotor time histories for the 60 knot case. Yaw and pitch rate response is well matched, however roll rate response appears slightly out of phase and has a rate bias. The attitude response is well matched with the exception of the integrated roll rate error.

Collective

As can be seen in figure B8, the model's overall rate response appears more damped than the flight test data. However the model does capture the yaw rate/collective dynamics. Attitude response displays bias terms present in the roll and pitch axes.

2.5.3 120 KNOTS MODEL

At high speeds the tail boom and horizontal stabilizer of the helicopter have a pronounced damping effect on both the lateral and longitudinal axes. This damping is evident upon examination of the 3-2-1-1 response and the identified diagonal parameters. The model corresponds well with the flight test data for both on-axis and off-axis response.

Longitudinal Cyclic

Figure B9 contains the longitudinal cyclic time histories for the 120 knot case. The pitch rate response is very well followed, with only a minor discrepancy occurring in the final second of the maneuver. For the first half of the maneuver the roll rate response is well followed, but grows to become under damped and slightly unstable. The magnitude of the yaw rate response is well followed but the phase is out by approximately 45°.

Lateral Cyclic

The roll rate response is very well followed throughout the duration of the maneuver as can be seen in figure B10. Pitch rate is also well followed with a slight bias

becoming evident in the final third of the maneuver. Roll rate is also well followed, however over the course of the final third of the maneuver the phase is out by approximately 45° .

Tail Rotor

Figure B11 contains the tail rotor time histories for the 120 knot case. Rate response is well followed until the final third of the maneuver where ‘spikes’ in the control signal become evident. This may have been the result of a measurement system malfunction. Yaw rate response is well followed until the final ‘1’ of the 3-2-1-1. Pitch response displays a steadily growing bias, whereas roll and yaw are relatively well matched.

Collective

Collective data for the 120 knot case was unavailable.

2.6 SUMMARY

This chapter has presented the details of the structure of the 8 degree of freedom model used for evaluation of the Bell 412’s response to hardover. The results of the model verification process demonstrate that the hover and 120 knot models present a good match of the flight test data. The 60 knot model however does not display an accurate match of the flight data, especially regarding off-axis responses. This discrepancy need not affect the development of the command validation algorithm, since the algorithm should be sufficiently generalized to account for the possibility of a new model with increased fidelity.

Chapter 3

BELL 412 MODEL STEP RESPONSE

3.1 INTRODUCTION

Linear systems are characterized by their step response, a characteristic commonly used for the identification of such systems. Although the Bell 412 model is non-linear, the core of the model is the linearized solution of the aircraft's aerodynamics. The step response of the system can be used to give a general idea of how the aircraft will behave when subject to actuator hardovers. Since the model is non-linear various magnitudes of hardovers (step inputs) must be simulated in order to fully characterize the step response. In this chapter the effects of step inputs, varying in amplitude from zero to actuator full throw, are examined at several trim conditions.

3.2 EVALUATING THE BELL 412HP'S STEP RESPONSE – METHOD

The step response of the Bell 412HP was evaluated by performing numerous simulations of hardover conditions. The mathematical model used was an eight degree of freedom model of the Bell 412 in flight with a forward velocity of 60 knots. It is anticipated that the dynamics of the helicopter will be highly dependent on its forward velocity, however the basic structure of the command validation algorithm should be independent of speed (obviously, some parameters must be sensitive to speed, but the same set of decisions should be made for the hover case, or flight at 120 knots). With this in mind, the generation of a framework for the command validation algorithm for flight at 60 knots is desired. Extensions to account for flight at other velocities may be added later.

Given the numerous initial conditions and possible combinations of actuator displacement the focus was placed on single axis failures and non-zero initial conditions in single axes (e.g., lateral cyclic failure, at various trim roll angles). The simulations assume that an undesired step disturbance occurs 0.1 seconds into the simulation, and 0.5 seconds later the safety pilot reacts by applying a step (1/2 the magnitude of the hardover step) in the opposite direction. The actuators are modeled

Figure 8: Typical Control Input Sequence

as a first order lag with a corner frequency of 75 rad/s; representative of the low amplitude response of the fly-by-wire actuators chosen for the ASRA²⁸. The pilot is modeled as a first order lag with a corner frequency of 8 rad/s. Figure 8 shows a typical control response for the step response simulation.

In order to characterize the step response, a program was written (in MATLAB) to determine the maximum excursion from trim of each of the aircraft's states ($u, v, w, p, q, r, x, y, z, \theta, \phi, \psi$). The program determines the points at which the output response changes directions and determines the maximum excursions that occur in each of three areas:

1. During hardover (i.e.: seconds)
2. During pilot recovery (i.e.: $0.6 \leq t < 2$ seconds)
- 3* The final point of simulation (i.e.: $t = 2$ seconds)

For most states, the results of the final point of the simulation were discarded due to the fact that the pilot model inputs a step control displacement for 1.4 seconds of the simulation (not very representative of a true pilot's action). The response at $t=2$ seconds is therefore somewhat less than accurate. The primary objective of these simulations is to determine the maximum excursion the ASRA would suffer under a hardover that is detected by the safety pilot, and acted upon. Since the Bell 412 model is non-linear, it responds differently depending on the magnitude of the actuator step

input. In order to investigate this effect the magnitude of actuator displacement was varied from -100% to 100% of full throw in increments of 5%. Table 4 shows the initial trim conditions that have been examined.

Table 4: Initial Conditions Investigated

Initial Condition	-50°	50°	5°
Pitch	-50°	50°	5°
Roll	-90°	90°	5°
Pitch Rate	-40°/sec	40°/sec	5°/sec
Roll Rate	-40°/sec	40°/sec	5°/sec
Fwd. Vel.	-20 m/sec	20 m/sec	2 m/sec
Stbd. Vel.	-20 m/sec	20 m/sec	2 m/sec
Vert. Vel.	-20 m/sec	20 m/sec	2 m/sec

In order to gain a further understanding of the effects of hardover during typical operation of the ASRA, simulations were performed using initial conditions taken from a rapid slalom maneuver. In these simulations the effects of dual axis hardovers (lateral and longitudinal cyclic) was investigated. The maximum and minimum values for pitch and roll were selected quite arbitrarily as values that would be beyond the flight envelope of the ASRA, ensuring that typical operating conditions of the ASRA will have been covered by the simulations. The values of pitch rate and roll rate initial conditions were determined by examining the maximum rates from slalom and 3-2-1-1 time histories.

Figure 9 shows typical results of the program to analyze the Bell 412's step response. Notice the step discontinuity present in the curve of v vs. pitch vs. actuator displacement. This is a direct result of the method used to determine the maximum values; a clearly defined maximum became less defined, and the program selected another maximum. The appearance of these step discontinuities suggests that the helicopter's behavior changes rapidly at one point. However, this is not the case. The response subtly changes from a clearly defined maximum to an extended region where the curve has a slope of zero. Each discontinuity is examined by performing simulations and examining the results manually to determine its cause.

Figure 9: Longitudinal Cyclic Hardover with variation in pitch initial condition

3.2 ANALYSIS OF STEP RESPONSE DATA:

The maximum excursion in rates and attitude (p , q , r , θ , ϕ , y) varied linearly with actuator step displacement. Analysis of the step response data reveals that the maximum rate excursions (p , q , r) of the aircraft have very little deviation from linearity with changes in initial condition. Slight non-linearities become apparent upon examination of the attitude data. Most attitude responses have a slightly second order curve. The curves of disturbance velocities and displacement are more non-

linear, with definite twisting with respect to changes in initial conditions. Variation of trim pitch angle resulted in only minor changes to the maximum excursions, while the trim roll angle had a pronounced effect on the calculated maximum excursion. The trim body rates have a sizable effect on the maximum excursion, as can be expected since when roll rate is 20 deg/s, and a hardover occurs to increase this rate, the result is quite different when the hardover occurs in the opposite direction.

Table 5: Longitudinal Actuator Hardover During Slalom

$P = -12^\circ/\text{s}$	$-48^\circ/\text{s}$	$-18^\circ/\text{s}$	$25^\circ/\text{s}$	-8°	-55°	7°
$Q = 8^\circ/\text{s}$	$-48^\circ/\text{s}$	$-24^\circ/\text{s}$	$15^\circ/\text{s}$	-20°	-34°	5°
$R = -13^\circ/\text{s}$	$-45^\circ/\text{s}$	$-22^\circ/\text{s}$	$21^\circ/\text{s}$	-12.5°	-27°	-11°
$\phi = -42^\circ$	$-48^\circ/\text{s}$	$-22^\circ/\text{s}$	$22^\circ/\text{s}$	-5°	-30°	11°
$\psi = -14^\circ$	$-48^\circ/\text{s}$	$-22^\circ/\text{s}$	$22^\circ/\text{s}$	-12°	-32°	6.5°
$\theta = 0^\circ$	$-48^\circ/\text{s}$	$-22^\circ/\text{s}$	$22^\circ/\text{s}$	-12°	-32°	6.5°
Avg.	$-48.0^\circ/\text{s}$	$-21.6^\circ/\text{s}$	$21.1^\circ/\text{s}$	-11.6°	-35.0°	4.2°

Table 5 shows the results of analyzing the maximum excursion developed in body rates and attitude angles for each component of the slalom initial condition due to a - 100% longitudinal cyclic hardover. Each component of the initial condition was analyzed separately to determine if the maximum displacement from the complete initial condition could be estimated by adding the effects of each component. Similarly each actuator was considered separately to determine if the maximum excursion experienced from a dual axis hardover could be predicted by summing the

effects of two single axis hardovers. Table 6 contains the maximum excursion results for slalom initial conditions due to a 100% lateral cyclic hardover.

Table 6: Lateral Actuator Hardover During Slalom

$P = -12^\circ/s$	$-85^\circ/s$	$-5^\circ/s$	$-20^\circ/s$	-3°	-51°	-12°
$Q = 8^\circ/s$	$-78^\circ/s$	$-4^\circ/s$	$-20^\circ/s$	18°	-40°	-11°
$R = -13^\circ/s$	$-75^\circ/s$	$-5^\circ/s$	$-10^\circ/s$	-6°	-38°	-31°
$\phi = -42^\circ$	$-78^\circ/s$	$-5^\circ/s$	$-21^\circ/s$	-7.8°	-41°	-13°
$\psi = -14^\circ$	$-78^\circ/s$	$-5^\circ/s$	$-21^\circ/s$	-3°	-41°	-11°
$\theta = 0^\circ$	$-78^\circ/s$	$-5^\circ/s$	$-21^\circ/s$	-3°	-41°	-11°
Avg.	$-78.7^\circ/s$	$-4.8^\circ/s$	$-18.8^\circ/s$	-12.4°	-42.0°	-14.8°

The contributions from each initial condition are averaged and this total is summed for both actuators. The initial condition is then added to the total. The result is the predicted maximum excursion. Table 7 presents the results of this procedure for the slalom hardover simulation. The results agree sufficiently well, with the exception of yaw rate r .

Table 7: Comparison of Results

Lon.+Lat Avg	-126.7 °/s	-26.4 °/s	2 °/s	-12.4 °	-77 °	-10.1 °
Lon. + Lat Avg. + I.C	-138.7 °/s	-18.4 °/s	-10 °/s	-12.4 °	-119 °	-24.6 °
Slalom Simulation	-127 °/s	-15.5 °/s	-52 °/s	-19 °	-117 °	-23 °
Relative Error	9.2%	18.0%	80.0%	35.0%	1.7%	7.0%

The predicted yaw rate excursion does not agree well with the simulated dual axis hardover maximum excursion because the yaw rate response from cyclic inputs does not have one clearly defined maximum. Rather, the response contains two definite “spikes” as seen in figure 10. These spikes can be attributed to the ‘dutch-roll’ mode of the helicopter.

Figure 10: Yaw rate response due to longitudinal cyclic input

Figure 11 presents the results of varying the duration of the hardover for flight at 60 knots with a -100% longitudinal hardover. For the longer hardovers (duration > 0.5 seconds) the maximum rate begins to taper. Upon further investigation of the effect of duration of hardover it may be possible to predict not only *if* the current command

will cause the aircraft to exceed its flight envelope, but *when*. The advantage of this would be that if the flight envelope will be exceeded in greater than 0.5 seconds it is likely that the safety pilot would catch the error, and the command can be validated (i.e: The algorithm will not concern itself with slowovers).

Figure 11: The Effect of Varying Hardover Duration (Longitudinal Cyclic -100%)

3.3 IMPROVING PILOT RESPONSE

The maximum excursion of the ASRA's step response is dependent upon the pilot's response to the hardover. In this section the pilot model is expanded from the simple open loop step function to one with position and rate feedback. This will allow the pilot to control the off-axes as well as the hardover axis.

3.3.1 PILOT MODEL BACKGROUND

Taking a broad overview, input-output engineering studies have reminded us that humans as trackers:

- Behave like low pass amplifiers,
- Have a built in reaction time delay,
- Can, in some circumstances, generate lead, or lag characteristics, and
- Behave as if they respond to events about twice a second

For short simulations (under 60 seconds) the pilot model can consider display error as input, and stick position as an output. At longer simulation times pilots will allow themselves a broader ambition than a mere tracking task, requiring the selection of different input and output criteria. Pilots need not treat an instrument as one exclusive information channel just because it is single axis (e.g., a pilot will not control height in isolation just because an altimeter displays nothing else). Also, pilots can use their view outside of the cockpit to obtain critical data.

A block diagram showing the pilot in the loop for a general control system is shown in figure 12. In the figure $G_c(s)$ represents the transfer function of the vehicle being controlled, and $G_p(s)$ the pilot model, which in its simplest form is:

$$G_p(s) = \frac{K_p(\tau_L s + 1)e^{-\tau_s}}{(\tau_I s + 1)(\tau_N s + 1)} \quad (8)$$

Where

K_p = Pilot static gain

τ_L = lead time constant

τ_l = lag time constant

τ_N = neuromuscular lag

τ_e = effective time delay

Figure 12: Pilot Model Block Diagram

The pilot will introduce sufficient lead or lag so that the slope of the open loop Bode plot is -20 dB/decade in the region of the crossover frequency and the phase margin is approximately equal to 90° . The gain K_p is adjusted to position the crossover frequency as required. It appears that the pilot attempts to choose a lead or lag value such that the sensitivity of the closed loop low frequency characteristics to variations in τ_L or τ_l are small, leaving gain and effective time delay as the primary means for adjusting closed loop stability and dominant modes²⁹.

The adjustment rules cannot be simply stated since they depend intimately on interactions of the elements in the man/machine system. The rules can, however, be divided into two categories; adaptation and optimization. Adaptation is the selection by the operator of a specific form (lead-lag, pure lead, pure lag, or pure gain); and optimization is the adjustment of the parameters of the selected form to satisfy some internally generated criteria (e.g.: good closed loop response in roll and pitch axes). The form selected by the adaptation process is one compatible with good low frequency closed loop response, insensitivity of the system to small changes in operator characteristics, and absolute stability of the system. The optimization criteria appear to be generally compatible with the minimization of the rms error³⁰.

3.3.2 PILOT MODEL STRUCTURE

The pilot model structure is detailed in appendix A. Yaw rate is fed directly back to the pilot's tail rotor input. Yaw angle feedback is not necessary, since in most hardover situations the pilot is primarily concerned with getting the aircraft under control (i.e.: low body rates, and low pitch and roll angles) as opposed to controlling heading angle. The collective axis serves a dual role; during forward flight it is used mainly for thrust trim, and in hover it is used to control height rate. The model shown in figure 7 involves the collective as a form of thrust trim. Since thrust is not one of the states of the Bell 412 model the collective axis is left as an open loop step function. A combination of rate and position feedback is used for cyclic control (pitch

and roll). The switch blocks perform the switching from hardover input to pilot response.

3.3.3 PILOTED STEP RESPONSE ANALYSIS

In order to analyze all the initial conditions specified in table 1, a program was written to expedite the process. The program automatically adjusts the initial conditions, and performs the simulations; once the simulation is complete, the state time histories are analyzed to determine the maximum excursions from the trim initial condition. A block diagram of the program structure is shown in figure 13.

Figure 13: Maximum finding algorithm

The code is divided into the following subroutines:

1. pstep, qstep, rstep etc...
2. actstep

3. analyze
4. maxcursion
5. maxanalyze

3.3.3.1 PSTEP, QSTEP, RSTEP ETC...

These functions contain the list of initial conditions and actuator displacements to be simulated.

3.3.3.2 ACTSTEP

This function performs the simulation, assigns state variables and initial conditions, and performs linear regression. After the maximum excursions are determined by the other functions, they are post-processed by actstep to determine if the maximum finding routines have been fooled by a monotonic increase in the state variable. This is accomplished by comparing the chosen maximum with the final value of the state time history; if the two are approximately equal the maximum is replaced with the value zero, signifying that no maximum was found.

3.3.3.3 ANALYZE

This function automates the process of calling the maxcursion and maxanalyze routines for each of the states in the simulation.

3.3.3.4 MAXCURSION & MAXANALYZE

These functions return the critical points from the simulation time history and groups them within the three regions of interest; during hardover, during pilot recovery, and the final point of simulation. The locations of the critical points are found by sequentially analyzing the time history to determine whether the function is increasing or decreasing. When the function changes from increasing to decreasing, or vice-versa, the location of a critical point is found. These points are then grouped according to what region they fall under. A special flag is used for the analysis of rate values. The flag is set when a rate value crosses zero after 0.5 seconds have elapsed. This is used to 'shorten' the associated attitude time history, since the maximum must lie at a point where the rate is zero.

3.4 SUMMARY

This chapter detailed the results and procedure of the analysis of the Bell 412's step response. The step response was evaluated as a response to two separate events; first, the actuator(s) receive the undesired step command from the FCC and proceed to undergo a step displacement and second, once the safety pilot has determined that the current command is dangerous (500 ms after the initial transient) and applies a corrective action. The model used to evaluate the response consists of a linear small disturbance core with trigonometric routines to account for larger disturbances. The evaluation of step response was performed for numerous trim conditions to develop a sense of the degree of non-linearity involved in the model. It was found that although

the model was quite non-linear at extreme trim condition cases the non-linearity was not significant within the expected operational envelope of the Bell 412. As a result of this investigation it was concluded that a linear model of the aircraft would be sufficient to predict its response to a hardover.

Chapter 4

REDUCED MODEL PREDICTION

Careful analysis of the Bell 412 step response data shows that the reaction of the pilot/aircraft system can be described as a function of the current state (initial condition) and the actuator input. Through the use of a simple functional it is possible to predict the maximum excursion from trim that a given command will yield. If this predicted excursion exceeds the desired flight envelope the command is deemed invalid, and control is switched from fly-by-wire mode to conventional mode. This chapter generates an analytical basis for the reduced model prediction method, and shows how this framework was extended to account for non-linearities.

4.1 PREDICTION (FINDING MAXIMA)

Although the Bell 412 model is non-linear, at its core are linearized aerodynamic equations expressed in state space form:

$$\dot{x} = Ax + Bu \quad (9)$$

For an initial state $x(0)$ of zero; the Laplace transform of the state is given by:

$$x(s) = (sI - A)^{-1} Bu(s) \quad (10)$$

If the output is defined by:

$$y(t) = Cx(t) \quad (11)$$

Then its Laplace transform is:

$$y(s) = Cx(s) \quad (12)$$

and,

$$y(s) = C(sI - A)^{-1} Bu(s) \quad (13)$$

The matrix:

$$H(s) = C(sI - A)^{-1} B \quad (14)$$

that relates the Laplace transfer function of the output to the Laplace transfer function of the input is known as the transfer function matrix. For a system with 8 states and 4 inputs (such as the Bell 412 model) $H(s)$ is an 8x4 matrix. The following notation is adopted

$$\begin{aligned} H_{i,j}(s) \\ i = 1, 2, \dots, n \\ j = 1, 2, \dots, m \end{aligned} \quad (15)$$

here, for a system with n states and m inputs, $H(s)_{i,j}$ is the transfer function relating state i to input j . For example, using the Bell 412 model, $H_{4,1}(s)$ is the transfer function relating roll rate to elevator input.

The transfer functions are expressed in zero/pole form, and some approximate cancellation can be performed (typically reducing the order of the transfer function by two or three).

The control input of interest is a sequence of step inputs shown in figure 14.

Figure 14: Input sequence

The transfer function of this input sequence is:

$$\mathcal{L}\{u(t)\} = \frac{1}{s} - \frac{3}{2s} e^{-\frac{1}{2}s} = \frac{1}{s} \left[1 - \frac{3}{2} e^{-\frac{1}{2}s} \right] = U(s) \quad (16)$$

The response of the aircraft to control input $u(t)$ is given by:

$$\begin{aligned}\mathcal{L}\{x_i(t)\} &= \sum_{j=1}^m U_j(s) H_{i,j} \\ i &= 1, 2, \dots, n \\ j &= 1, 2, \dots, m\end{aligned}\tag{17}$$

We are interested in the maximum roll and pitch angles that occur in the aircraft's response. We will drop the Euler terms in the roll and pitch equations for simplicity, resulting in the linear relation:

$$\begin{aligned}p &= \dot{\phi} = \frac{d\phi}{dt} \\ q &= \dot{\theta} = \frac{d\theta}{dt}\end{aligned}\tag{18}$$

The maximum angle reached during the response will occur at the time the rate is equal to zero:

$$x_i(t) = \mathcal{L}^{-1}\{U_j(s)H_{i,j}(s)\} = 0 \quad t = t_{\max}\tag{19}$$

Since $H_{i,j}(s)$ is a polynomial in s of the form

$$\begin{aligned}\frac{a_1 s^c + a_2 s^{c-1} + \dots + a_c}{b_1 s^d + b_2 s^{d-1} + \dots + b_d} \\ d \geq c\end{aligned}\tag{20}$$

and:

$$U_j = \frac{1}{s} \left[1 - \frac{3}{2} e^{-\frac{1}{2}s} \right]\tag{21}$$

by integrating the state the maximum angular displacement from trim can be found from:

$$\theta_{\max}, \phi_{\max} = \frac{1}{s} \left(1 - \frac{3}{2} e^{-\frac{1}{2}s} \right) \left[\frac{a_1 s^c + a_2 s^{c-1} + \dots + a_c}{b_1 s^d + b_2 s^{d-1} + \dots + b_d} \right] \quad (22)$$

$H_{i,j}(s)$ can be expressed in pole-zero form as:

$$H_{i,j}(s) = \frac{(s - z_1)(s - z_2) \dots (s - z_c)}{(s - p_1)(s - p_2) \dots (s - p_d)} \quad (23)$$

The angle equation can be rearranged to give:

$$\begin{aligned} \theta, \phi(s) &= \left(1 - \frac{3}{2} e^{-\frac{1}{2}s} \right) \left[\frac{(s - z_1)(s - z_2) \dots (s - z_c)}{(s - p_1)(s - p_2) \dots (s - p_d)} \right] \\ &= \left(1 - \frac{3}{2} e^{-\frac{1}{2}s} \right) F \end{aligned} \quad (24)$$

Since the denominator of F is a product of linear factors with a repeated root at zero,

F may be expressed in partial fraction expansion form as:

$$F(s) = \frac{J}{s - p_1} + \frac{K}{s - p_2} + \dots + \frac{X}{s - p_d} + \frac{Y}{s} + \frac{Z}{s^2} \quad (25)$$

The time response is then:

$$\begin{aligned} \theta, \phi(t) &= J e^{p_1 t} + K e^{p_2 t} + \dots + X e^{p_d t} + Y + Zt \\ &\quad - u(t) f(t - 0.5) \frac{3}{2} [J e^{p_1 t} + K e^{p_2 t} + \dots + X e^{p_d t} + Y + Zt] \end{aligned} \quad (26)$$

Solving this equation at $t = t_{\max}$ will yield the maximum angular departure from trim,

$\theta_{\max}, \phi_{\max}$ for a unit step actuator displacement. Since the state space model is linear

the angular response is also a linear function of the magnitude of the step input, that

is:

$$\theta_{\max}, \phi_{\max} (\text{actuator displacement}\%) = \frac{\theta_{\max}, \phi_{\max} (\text{unit step})}{\frac{\text{unit step displacement}}{\text{actuator fullthrow}}} * 100\% \quad (27)$$

This results in a reduced order model that predicts the future attitude of the aircraft, assuming that the current command will have a duration of 0.5 seconds, and the pilot's reaction is to apply a step of half the magnitude of the input command in the opposite direction. This assumption is justified by pilot experience; during a hardover failure the stick (or collective or pedals) will suddenly undergo a large displacement, and the pilot's natural reaction is to re-center the stick then apply corrective control action. While the concept of a pilot applying open loop control to a highly coupled system is not desirable from a stability standpoint it does accomplish its purpose; to create a well defined maximum excursion from trim attitude. The command duration of 0.5 seconds is assumed in order to approximate the combined transport delay involved in the pilot/aircraft system. Typically safety pilots are highly motivated and alert, with a maximum neuromuscular delay of about 300 ms. Since the response delay of the Bell 412 is approximately 72 ms the effective pilot plus system delay is 372 ms. The assumption of a 0.5 second hardover duration can be seen as a conservative or 'worst-case' approximation of the system plus pilot response delay.

4.2 LEAST SQUARES METHOD

The preceding section describes a method for predicting the future attitude of the linear Bell 412 model subject to step inputs. This method is adequate for small

displacements, since the equations of motion can be linearized about a small displacement from trim (via the assumption of $\sin\theta \cong 0, \cos\theta \cong 1$), however the assumption breaks down for large displacements. In order to account for these differences the step response data was analyzed to examine maximum angular displacements. Also, the analysis of step response data allowed for the use of a feedback pilot model (as shown in section 3.3). Baseline model data is generated by simulation of 5% increments of actuator displacement with all initial states equal to zero. The reduced model is of the form

$$\theta_{\max}, \phi_{\max} = b_0 + b_1(\% act) \quad (28)$$

where $\% act$ refers to the percentage of actuator displacement. Introducing the notation

$$\angle_{\max}^T = [1 \quad \%act(i)] \quad (29)$$

$$b = [b_0 \quad b_1]$$

$$\%act = [-100\% \quad -95\% \quad \dots \quad 100\%]$$

The parameter b is chosen to minimize the least-squares loss function

$$V(b, \%act) = \frac{1}{2} \sum_{i=1}^{\%act} (\theta_{\max}(i) - \angle_{\max}^T b)^2 \quad (30)$$

The result gives the maximum attitude excursion as a function of the actuator input. This operation is carried out for all four actuators, and the DC terms are ignored, producing the model.

$$\theta_{\max} = b_{\delta e}(\% \delta e) + b_{\delta c}(\% \delta c) + b_{\delta a}(\% \delta a) + b_{\delta r}(\% \delta r) \quad (31)$$

4.3 COMPARISON OF LINEAR AND LEAST SQUARES MODELS

Table 8 shows the parameters identified for both the linear and least squares models of the Bell 412 at hover, 60 knots and at 120 knots.

Table 8: Linear and Least Squares Reduced Order Model Parameters

		Longitudinal Cyclic	Collective	Lateral Cyclic	Tail Rotor
60 knot model					
Roll Angle	Linear Model	.2186	.0682	-.4054	.1468
	Least Squares Model	.267	.0816	-.4144	.089
	Relative Error	18.12%	16.42%	2.17%	67%
Pitch Angle	Linear Model	.1256	.0334	-.0210	.0123
	Least Squares Model	.1136	.0473	-.0189	.0093
	Relative Error	9%	-41%	10%	24%
120 knot model					
Roll Angle	Linear Model	.0231	-.0412	-.2918	.1368
	Least Squares Model	.0177	-.04388	-.3318	-.0988
	Relative Error	30.5%	5.9%	12%	???
Pitch Angle	Linear Model	.1872	.0744	.0015	.0357
	Least Squares Model	.2253	.0806	.0439	.0082
	Relative Error	17%	8%	96%	335%
Hover Model					
Roll Angle	Linear Model	.0101	-.0898	-.3013	.0832
	Least Squares Model	-.0018	-.0846	-.3258	.065
	Relative Error	1500%	6.14%	7.52%	28%
Pitch Angle	Linear Model	.2021	-.0389	.0109	.0132
	Least Squares Model	.2067	-.0467	.0475	.0156
	Relative Error	-2%	17%	77%	-15%

From the table it can be seen that the linear and least squares models compare favorably, especially with regard to on-axis response. Although the relative error for the pitch angle due to lateral cyclic input seems quite high, the absolute error is quite low, when compared with the magnitude of the on-axis response.

Figures 15 and 16 plot the prediction parameters versus forward airspeed. From the plots it can be seen that the on-axis parameters do not widely vary with increasing airspeed. The off axis parameters, however, do seem to vary significantly. This is likely a result of an inaccurate model of the Bell 412 at 60 knots. Once more models of the Bell 412 are identified, the true shape of the parameter vs. airspeed curve may be seen.

Figure 15: Pitch angle parameters vs. forward airspeed

Figure 16: Roll angle parameters vs. forward airspeed

4.4 INITIAL CONDITIONS

The model form shown in equation 31 has no provision for the initial condition of the aircraft. Examination of the step response data shows that the initial body rates have the greatest effect (of all initial conditions) on the maximum attitude excursion suffered during a hardover. The effect of initial rate on the attitude of the aircraft can be expressed as a gain on the rate value since the step response shows rate to have a linear effect. The initial attitude of the aircraft must also be considered. This is accomplished by considering the current attitude as a bias term in the reduced model. The current attitude angle is added to that predicted by the model to predict the maximum attitude excursion that would be suffered if the current command were a hardover. For example, to predict pitch angle excursions the following equation would be used:

$$\theta_{\max} = b_{\delta e}(\% \delta e) + b_{\delta c}(\% \delta c) + b_{\delta a}(\% \delta a) + b_{\delta r}(\% \delta r) + kq + \theta_{curr} \quad (32)$$

Where:

θ_{curr} Current pitch angle of the aircraft

k Static gain on pitch rate as determined from step response

q Current pitch rate

The prediction equation 32 considers control displacements from trim, however ASRA's systems measure absolute control displacement. In order to account for this discrepancy the process shown in the block diagram of figure 16 is used.

Figure 17: Discrete differentiator

The block diagram shown in figure 17 has the following transfer function

$$H(z) = \left(1 - \frac{1}{z}\right) U(z) \quad (33)$$

Equation 33 may be thought of as a discrete differentiator. The transfer function of equation 33 does not truly compensate for the trim discrepancy, however since its rate

and position are input to the prediction equation the aircraft can be considered trimmed for any control input.

4.5 PREDICTION OF ALTITUDE LOSS

In order to be able to determine if a given command will exceed the ASRAs flight envelope it is necessary to predict the future altitude of the aircraft. This is due to the fact that the flight envelope is a function of altitude (obviously, the greater the altitude, the larger the envelope). However, in order to calculate flight path it is necessary to solve 3 simultaneous equations:

$$\begin{aligned}x &= m[\dot{U} + qw - rv + g \sin \theta] \\y &= m[\dot{V} + rv - pw - g \cos \theta \sin \phi] \\z &= m[\dot{W} + pv - qv - g \cos \theta \cos \phi]\end{aligned}\tag{34}$$

The equations are non-linear since they contain terms which comprise the product of dependant variables. Obviously it would not be possible to evaluate the equations 'on-line' without incurring a significant time delay. Instead, the altitude is predicted through similar means as the attitude. By performing successive simulations it is found that the altitude is highly dependant upon attitude of the aircraft. For example, at high roll angles, the use of the lateral cyclic causes greater altitude loss than at low roll angles. This occurs since the aircraft fixed axes (in which aircraft velocities are measured) shift with respect to the inertial axes (in which aircraft displacement is measured), so that at high roll angles, lateral velocity (v) has more effect on altitude than downward velocity (w). To account for this effect, the vertical and lateral

displacement from trim at nose and wings level for given actuator steps was found.

These displacements are converted to altitude loss via the following equation:

$$h = z_{\max} \cos \theta \cos \phi + y_{\max} \sin \phi \cos \theta + k\dot{h} \quad (35)$$

The equation simplifies the non-linear aircraft motion equations into a deterministic form. The values of $z_{\max}$ and $y_{\max}$ are obtained for wings and nose level trim for hover, 60, and 120 knots forward speed.

4.6 SUMMARY

This chapter presented the command validation algorithm proposed for the ASRA. The algorithm is based upon open loop model of the aircraft and pilot's response to uncommanded step displacements of the controls. Through the use of the open loop model it is possible to predict the response of the aircraft under the presumption that the current command will have a duration of 500 ms.

Chapter 5

SIMULATION EVALUATION OF CVA

This chapter describes the simulation of the command monitoring concept, within the **SIMULINK** environment, described in the previous chapter. The focus of these simulations is to assess that the CVA performs its intended function under all circumstances. In order to accomplish this the CVA is tested with various initial conditions corresponding to ADS-33C maneuvers. A brief description of the **SIMULINK** model and the ADS-33 maneuvers is included.

5.1 CVA MODEL STRUCTURE

Figure 18 shows the block diagram of the outer loop of the simulation structure. The basic helicopter model, and pilot model are as previously described. The CVA block acts as a trigger for the switch block, its output is either a one or a zero, corresponding to valid or invalid command respectively. Once an invalid command has been received, control is switched from the FCC (here, the FCC output is simulated by direct applications of step inputs) to the safety pilot. The input and the output of the CVA block are sampled at 64 Hz through the use of zero order holds.

Figure 18: Overall CVA and Bell 412 block diagram

Once the input signals have been sampled at 64 Hz, the command signals (δe , δa , δc , δr) are differentiated as was shown in figure 15. The attitude effect of the current command is calculated according to:

$$\theta_{\max} = b_{\delta e}(\% \delta e) + b_{\delta c}(\% \delta c) + b_{\delta a}(\% \delta a) + b_{\delta r}(\% \delta r) + kq + \theta_{\text{curr}} \quad (36)$$

as was shown in chapter 4. The only difference between this equation and the one shown earlier is that the command input gains (b_{δ_e} , b_{δ_a} , *etc...*) are not static variables, but are functions of forward airspeed. This allows the CVA to function over the full range of airspeeds. For the simulation the command input gains are linearly interpolated between model velocity points (hover, 60, 80, and 120 knots). Once the predicted attitudes have been calculated, the altitude is predicted. A logic circuit is used to ensure that the predicted altitude is never greater than the current altitude. This assumption ensures that the envelope remains conservative while the altitude is low. With the altitude, and attitude predicted, the envelope can be evaluated to validate the commands. For the simulation, the safety envelope was taken as a height bias of 2 meters (i.e.: there will always be at least 2 meters between the rotor blades and the ground) for low-level maneuvers. The ‘up and away’ envelope was taken as 45° maximum absolute pitch angle, and 70° maximum absolute roll angle.

5.2 DESCRIPTION OF ADS-33C AGGRESSIVE MANEUVERS

In this section a brief description of the ADS-33 maneuvers used for simulation testing of the CVA is presented. Initial conditions were chosen from the aggressive ADS-33 maneuvers since they are representative of the typical flight environment for the ASRA. Intuitively, the ‘worst case’ scenario for a helicopter without FCC command validation is an intense hardover at an aggressive initial condition. From the simulation it is desired to see that this threat is eliminated.

5.2.1 ACCELERATION AND DECELERATION

This maneuver, known as the 'quick-stop', is started from a stabilized hover. Power is rapidly increased to maximum, and altitude is maintained by pitching nose down. The collective is held constant during the acceleration up to 50 knots. Upon reaching this airspeed a rapid deceleration is performed by aggressively reducing the power and holding altitude constant by pitching nose up. The peak pitch attitude typically occurs just prior to reaching the final stabilized hover. The objectives of this task are to check the pitch and heave axes handling qualities for aggressive maneuvering, i.e., near the limits of performance. The initial conditions chosen for simulation are contained in table 9. Case 1 represents the maximum pitch attitude case, in the deceleration portion of the maneuver, whereas case 2 represents the maximum pitch rate case during the acceleration phase.

Table 9: Quick Stop Initial Conditions

<i>Case</i>	<i>P_o</i> (deg/s)	<i>Q_o</i> (deg/s)	<i>R_o</i> (deg/s)	<i>θ_o</i> (deg)	<i>φ_o</i> (deg)	<i>ψ_o</i> (deg)	<i>U_o</i> (m/s)	<i>V_o</i> (m/s)	<i>W_o</i> (m/s)
1	-2	0	-2	22	3	0	0	0	0
2	-2	-15	0	5	0	-2	0	0	0

Two failure conditions were considered for the quick-stop; a full longitudinal cyclic pitch up, and a longitudinal cyclic pitch down with full collective down. These failures were chosen as probable 'worst-case' scenarios. For the simulations the hardover occurs at time T=0. The results of the simulations can be found in appendix

C. The predicted attitude and altitude are shown by the dotted lines, whereas the actual altitude and attitude are shown by the solid lines. The graphs display the output of the CVA for the given time, thus explaining the ‘spike’ present at time zero (the time of the hardover). Figure C1 shows the prediction and time histories for attitude, altitude and CVA state. From the figure it can be seen that the predicted pitch angle agrees well with the time history. Conversely, the predicted roll angle is much smaller than the actual roll excursion. This is a function of the non-linear effect of initial conditions. As the initial pitch angle increases, so does the roll attitude excursion, however, the CVA, being linear, predicts the same excursion for any pitch initial condition. The results of the single axis hardover simulation are shown in figure C2. From the figure it can be seen that the pitch and roll attitude excursions are well predicted. Figure C3 shows the results of the two-axis hardover (collective and longitudinal cyclic). Again the pitch and roll attitude excursions are well predicted, and the predicted altitude loss is in the correct general range. Interestingly the pitch attitude excursion never exceeded the safety envelope. This is due to the relatively low cross-coupling of the Bell 412 at hover, and the pitch damping effect of its weight distribution.

5.2.2 RAPID SIDESTEP

Starting from a stabilized hover with the longitudinal axis of the helicopter pointed 90° to a reference line marked on the ground, a rapid lateral translation is initiated, with a bank angle of at least 25° , while maintaining altitude constant using the collective. When the rotorcraft has reached a lateral velocity within 5 knots of its

maximum allowable lateral airspeed an aggressive deceleration to hover is performed. The peak bank angle during the maneuver is usually at least 30° , and occurs just prior to the return to hover. The hover is maintained for 5 seconds, and the maneuver is repeated in the opposite direction. The objective of this task is to examine the lateral/directional handling qualities for aggressive maneuvering. Table 10 shows the initial conditions simulated for the rapid sidestep maneuver. As with the quick-stop, case 1 is representative of the maximum attitude case, and case 2 represents the maximum rate case. The failure condition chosen for the rapid sidestep is a full lateral cyclic hardover in the direction to increase the overall roll angle, or in the direction of the roll rate.

Table 10: Rapid Sidestep Initial conditions

<i>Case</i>	P_o (deg/s)	Q_o (deg/s)	R_o (deg/s)	θ_o (deg)	ϕ_o (deg)	ψ_o (deg)	U_o (m/s)	V_o (m/s)	W_o (m/s)
1	0	0	0	5	30	0	0	0	0
2	30	0	0	8	5	0	0	4	0

Figure C4 displays the simulation results for case 1 with the CVA engaged. Upon the onset of the hardover the CVA predicts an envelope breach, and disallows the command. Figure B5 displays the results of the simulation with the CVA disengaged. The pitch and roll attitude excursions are well predicted, and the command does in fact produce a roll attitude envelope breach. The predicted altitude is again in the correct range. The simulation results for case 2 of the rapid sidestep can be found in

figure C6. As before, the pitch and roll predictions match the simulated results well, and the predicted altitude loss is in the correct range.

5.2.3 RAPID SLALOM

This maneuver is initiated from steady level flight at 60 knots with the aircraft lined up with the centerline of the test course. Turns are performed, in a slalom fashion, to the outside of four reference pylons. This maneuver is performed at a reference altitude below 100 feet. The objective of this task is to examine the handling qualities of the helicopter in aggressive forward flight.

Table 11: Rapid Slalom Initial Conditions

<i>Case</i>	<i>P_o</i> (deg/s)	<i>Q_o</i> (deg/s)	<i>R_o</i> (deg/s)	<i>θ_o</i> (deg)	<i>φ_o</i> (deg)	<i>ψ_o</i> (deg)	<i>U_o</i> (m/s)	<i>V_o</i> (m/s)	<i>W_o</i> (m/s)
1	0	7.5	-12	-2.5	-42	0	30	0	0
2	-35	-2	-7	1	-25	-2	30	0	0

Table 11 shows the initial conditions examined for the rapid slalom maneuver. Case 1 is representative of the maximum roll attitude condition, and case 2 is representative of the maximum roll rate condition. The failure condition evaluated for this maneuver was a full lateral cyclic deflection in the direction of the roll rate and roll angle (for the initial conditions both were negative).

Figure C7 shows the time history for the case 1 failure with the CVA engaged. At the time of failure the CVA predicts an envelope breach and assigns control of the helicopter to the safety pilot. Figure C8 displays the results for the same failure

condition and initial conditions, but with the CVA disengaged. From the figure it can be seen that the CVA predicts the attitude excursion very well. The algorithm is, however, somewhat inaccurate in its prediction of altitude. Figure B9 shows the simulation results for the case 2 failure condition. Again, the CVA predicts a roll envelope breach and cancels the fly-by-wire mode. Figure C10 show the results of the same simulation except with the CVA disengaged. The algorithm predicts the envelope breach well, however the maximum roll attitude attained is slightly greater than the predicted attitude.

5.3 SUMMARY

The simulation evaluation demonstrates that the CVA proposed for ASRA is effective at preventing potentially dangerous hardovers from reaching the actuators. At large pitch angles, however, the prediction of roll angle becomes poor. This is a result of the non-linear effect initial conditions have on the aircraft's response, and the simplification of the pilot response.

Chapter 6

ASRA CVA IMPLEMENTATION ISSUES

The previous chapter addressed the suitability of the command validation algorithm for the ASRA through simulation evaluation. However, in practice some alterations have to take place in order to fully implement the design. The successful operation of the command validation algorithm depends upon the presence of well-defined flight envelope limits. It is likely that FRL will set the ASRA's limits according to their confidence in the control laws. Also, feedback data will have to be filtered in order to prevent signal noise from causing nuisance trips. This chapter presents a basic set of

envelope limits that may be used to test the operation of the CVA, as well as a means of filtering the sensor data.

6.1 HARD HOVER ENVELOPE LIMITS

The hover hard envelope limits are set in order to prevent main or tail rotors from striking the ground. These limits are characterized as ‘hard’ since they are based upon the absolute physical limit of the helicopter at low level flight. Intuitively the hard envelope should become increasingly narrow as altitude drops. The limits are a function of the rotor disc diameter and altitude. The roll envelope limits are described by:

$$|\phi_{\max}| = \sin^{-1}\left(\frac{h}{d}\right) \quad (37)$$

where

h rotor hub height (m)

d rotor diameter (m)

Figure 19 shows the critical dimensions of the Bell 412. The rotor diameter is approximately 14 meters, and the rotor hub height is approximately 3.5 meters.

Figure 19: Bell 412 Critical Dimensions

Figure 20 displays a plot of the absolute roll envelope limits versus rotor hub height.

Figure 20: Absolute roll limit vs. rotor hub height

The pitch envelope is limited by main rotor tail strike for negative pitch angles (nose down) and tail rotor strikes for positive pitch angles.

The pitch envelope limits are:

$$\theta_{\max -ve} = \sin^{-1}\left(\frac{h}{7}\right) \quad \theta_{\max +ve} = \sin^{-1}\left(\frac{h-1.1}{8.3}\right) \quad (38)$$

Figure 21 presents a plot of the absolute pitch angle envelope versus rotor hub height.

Figure 21: Pitch absolute pitch envelope vs. rotor hub height

6.2 UP AND AWAY FLIGHT ENVELOPE

The bulk of the ASRA's high-speed work will be done in what is known as an 'up and away' condition. The flight envelope for this condition can be much broader than that of the hover because of the added safety of having ample recovery altitude. However, there still exists the possibility for a safety failure as it is during the up and away state where the controllers are usually engaged for the first time during their developmental stage. The envelope limits for this condition will be fixed with respect to altitude. Discussions with safety pilots have revealed that a comfortable maximum attitude would be approximately 60° in roll, and 45° in pitch.

6.3 PILOT WARNING SYSTEM

In the event of a CVA trip of the fly-by-wire system, a signal must inform the safety pilot that he has control of the aircraft. This signal would likely consist of both a visual warning, and an audio tone of a particular frequency and duration.

As a fly-by-wire trip warning system is currently installed in the ASRA there is no need to add another system to account for the presence of the CVA. There is no need to differentiate to the safety pilot the source of the trip via separate indicator and/or tone. Instead, the general fly-by-wire trip tone and lamp would come on and the source of the trip would be output to one of the multi-function displays.

6.4 CVA TEST AND VERIFICATION PROCEDURE

Before the CVA can be fully implemented on the ASRA it is necessary to test the algorithm, its code, and its parameters. As a broader experience base is developed on the Bell 412, the CVA can be tuned to provide higher performance with less nuisance trips. Initially the code will be tested to ensure that the proper sign convention, and units have been used. This will be accomplished by flying with the code in a passive mode. The algorithm will receive input from the FCC, but will lack the ability to trip the FBW. The output of the CVA should roughly follow the attitude (and altitude) of the aircraft since it is effectively predicting a future attitude. Each term of the algorithm's equations can be checked by scaling all other factors to zero, and examining the output of the CVA. This allows the algorithm to be tested through the use of one channel.

Once the signs and operation of the code has been assured the fundamental assumptions may be checked. Paramount in the design of the CVA is the assumption

of the pilot response to a step type input. This conjecture can be investigated through an in-flight experiment employing FRL's Bell 205 Airborne Simulator. The experiment would involve the application of a FCC generated step input at a random time within a specific time window (for example say 5 seconds). Once the safety pilot has detected the step they are to recover the aircraft back to a wings and nose level trim condition. Intuitively the magnitude of the step command must be kept within reasonable limits; not so large as to compromise the safety of the helicopter and its crew, and not so small as to prevent the safety pilot from its detection. The results of this test would allow for the verification, and possible modification, of the pilot response assumed in the development of the CVA.

In order to evaluate the effectiveness of the algorithm, failures will have to be simulated. For obvious reasons these failures cannot be simulated at low altitudes. The tests can be carried out at higher altitudes, however, the lack of sufficient pilot cues could distort the results. To compensate for the poor cueing environment afforded by high altitude hover testing, the tests could be carried out with the use of a cloud. Provided the cloud were of a sufficient density and altitude it would be possible to use it as a hover cue. The pilots would be told to treat the cloud as if it were the ground. This would allow further verification of the pilot response, and the attitude excursion suffered for each hardover can be compared against the algorithm's predicted attitude and altitude.

6.5 FEEDBACK SIGNAL FILTERING

Since the helicopter is a noisy platform from which to take measurements it will be necessary to perform some type of complimentary filtering on the feedback signals. The filters used for most control applications in the Bell 205 Airborne Simulator are first order low pass filters. For the rate and attitude channels a breakpoint of approximately 15 rads/sec should be of a sufficient frequency to capture the important dynamics while filtering the noise. The altitude of the helicopter will have to be measured via radar altimeter (RADALT) since pressure altitude does not give an indication of the height above the ground but rather height above sea level. RADALT traces are known to be fairly noisy with peaks generated in the signal periodically as the radar signal is reflected by complex objects. For this reason the RADALT signal should be filtered at approximately 5 rads/sec.

Figure 22 displays an altitude trace taken from a Bell 212 helicopter. It can be seen from the trace that there are numerous 'dropouts' in the RADALT trace. Although the Bell 412 RADALT does not experience dropouts to the same degree as the Bell 212, nevertheless the problem is present.

Figure 22: Bell 212 Radalt and pressure altitude trace

6.6 CODE STRUCTURE AND FLOW

This section documents the structure and order of the software code that will be used to implement the CVA. Figure 23 shows a flow chart of the command validation process. Once the FCC has computed the commands to the actuators they will be passed to the CVA along with the necessary feedback information (i.e.: rates, attitudes, velocities, and altitude). The algorithm will first calculate the predicted excursion as described in chapter 4. The feedback variables and the commanded actuator deflections are then used to determine the altitude change. Since the method used to predict altitude is fairly crude an added measure of safety is provided by ensuring that the predicted altitude is always less than or equal to the current altitude.

In the event that the predicted altitude is greater than the current altitude (as would be the case for a collective up failure) the predicted altitude is set equal to the current altitude. This ensures that the lowest altitude is used to evaluate the flight envelope limits. These limits are evaluated utilizing the predicted altitude as described in section 6.1.

Figure 23: CVA flow chart

6.7 COMPUTATIONAL SAFETY

Some overall considerations have to be taken into account to ensure that the CVA will perform its intended purpose, and not be the cause of potentially dangerous failures. This is accomplished by structuring the CVA such that it works in parallel with the FCC, as opposed to in series. Although it is a subtle difference, if the CVA were to work in series with the FCC its output would be the command signals, allowing for the possibility of the CVA to modify the FCC's commands. Working in parallel with the FCC, the CVA's output would be binary; it either approves or disapproves of the current command. The FCC commands will wait for the output of the CVA, perhaps as a multiplying factor; multiplying the commands by one if they are approved, and zero if they are not.

Chapter 7

CONCLUSIONS AND RECOMMENDATIONS

This chapter documents the lessons learned from the development of the CVA for the ASRA Bell 412. As time restraints have prevented the algorithm from having been tested and implemented the conclusions are based upon the merit of the simulations, and feedback from experts within the field of fly-by-wire research aircraft. The advantages, and distinctive characteristics of the digital CVA are discussed. Moreover, sources of error and possible shortcomings of the algorithm are addressed.

7.1 CONCLUSIONS

The open loop prediction model command validation algorithm is a viable means of preventing FCC commanded hardovers from reaching an aircraft's actuators. The algorithm's use is essentially restricted to the research aircraft domain since production aircraft that operate via fly-by-wire have redundant dissimilar flight control computers and control software and hence no need for an FCC watchdog like the CVA. However, in the research field it is considered advantageous for aircraft to be of a 'single-string' nature. This implies:

- a single set of FBW actuators;
- one, non-redundant flight control computer;
- a single set of aircraft state sensors; and
- a single set of flight control software

These features significantly reduce the maintenance and operating costs associated with the aircraft. Such simplicity of design facilitates the incorporation of software changes without the overhead of multiple coding sources, multiple languages or operating systems and in-depth code validation. All are necessary for production systems but would be overly prohibitive for flexible, time critical research programs. Inherent in the single string architecture is a reliance on a safety pilot to mitigate against system failure or exceedance of the safe flight envelope. Such an operating methodology places high demands on the safety pilot, but affords the advantages of increased flexibility by retaining entire the certified operational envelope of the

aircraft, and allowing the aircraft to maneuver aggressively and take-off/land without restriction with the FBW system engaged.

Single string aircraft are subject to the potential threat of FCC commanded hardovers reaching the actuators and endangering the aircraft. For this reason most single string research aircraft have a safety pilot who can disengage the fly-by-wire system and fly the aircraft with its stock mechanical system. This system has its limitations however, since humans have a finite reaction time, and the possibility of distraction. These two effects, combined with a fast aircraft with high control power can lead to disastrous results. The FRL bore this in mind during the conversion of their Bell 412HP into a fly-by-wire aircraft. For this reason they believe that, with the Bell 412's high control power and low time delays, there must be a supplementary system to prevent hardovers. This system is the CVA described in this thesis. The CVA examines the commands from the FCC and determines their validity by examining the combined effects of the commands, the aircraft state, and the pilot recovery process. Chapter 4 has shown a theoretical basis for the development of the reduced order prediction model, and chapter 3 has shown that while the identified model (like the helicopter itself) is a non-linear system, it can essentially be treated as a linear system (i.e.: subjected to the principle of superposition). Chapter 5 introduced some of the practical implementation issues that have, and will, be encountered when the CVA is installed on ASRA.

7.2 BENEFITS OF DIGITAL CVA

The previous chapters have shown the development, testing, and implementation issues of the CVA. Some of the foreseeable benefits of the CVA include:

- Rate limiting and/or rate tripping can be eliminated,
- The algorithm is easy to evaluate,
- The flight envelope can be fully described/protected,

7.2.1 ELIMINATION OF CONTROL SYSTEM RATE LIMITS

Rate limiting is often used in aircraft control systems to inhibit fast control commands, however rate limiting is widely acknowledged to play a crucial role in aircraft control problems. McRuer³¹ has outlined that the effect of rate limiting is to add phase lag between the pilot command and the aircraft response and to reduce the crossover frequency. This added phase lag can lead to a pilot induced oscillation (PIO), an unwanted and inadvertent closed loop coupling between the pilot and one or more independent response variables of the aircraft. The CVA can be much more effective than a rate limit since the commands are not affected by the algorithm as it operates, only in the event of a possible envelope breach will the CVA affect the command by switching control to the safety pilot. The clear advantage is that the effect of a rate limit on a flight control system is not well defined, whereas since the CVA operates in a binary fashion either allowing or disallowing the command, it has no effect on control system feel.

7.2.2 ELIMINATION OF RATE TRIP LIMITS

Rate trip limits differ from rate limits in that when a command exceeds the rate limit it trips the FBW system as opposed to being set to the rate limit. This can be a source of 'nuisance trips' of the system, especially during aggressive maneuvers such as the quick-stop, and side-step.

For the Airborne Simulator the rate trip limits are set conservatively, and then, in the presence of 'nuisance trips', increased (i.e., allowing for faster responses) according to the pilot's confidence in the control system under evaluation. This system has worked well for the Bell 205 since its dynamics are well understood by the FRL. However, with its increased control power, and quicker response it is doubtful that such a system will work for the ASRA Bell 412, at least until a greater experience base has been generated for the aircraft. Some inherent problems with this approach include the fact that it is based upon the pilot's perceived ability of the aircraft rather than sound mathematics or engineering judgement. The rate limits are increased until the nuisance trips no longer interfere with the experiment, but there remains the problem that the aircraft may now be able to get itself into a dangerous situation within the new limits. One of the prime advantages of the CVA is that it separates the command monitoring problem into two distinct problems; prediction of the aircraft's future state, and evaluation of the flight envelope. Rate limit trips are not set based upon the flight envelope, but rather the 'typical' rates of command As opposed to directly protecting the aircraft flight envelope, they serve to protect the aircraft from 'abnormal' control inputs.

7.2.3 SIMPLICITY

One of the largest drawbacks involved in the use of digital technology for aircraft flight systems is the introduction of computational delays. These delays can lead to the addition of phase delay and its associated PIO problems. The CVA described in this thesis solely relies on simple multiplication and addition to predict the aircraft state and envelope, and comparison to validate the commands. It is expected that the algorithm will be easily executed within the 64 Hz clock cycle of the VME computer of the ASRA. Another practical advantage offered by the algorithm's simplicity is that it allows for easy debugging and coding.

7.2.4 DESCRIPTION OF FLIGHT ENVELOPE

The CVA described in this thesis allows for the description of a flight envelope. This allows for increased freedom of the pilots and test engineers to modify the flight envelope according to confidence in the tested control system and the relative frequency of nuisance trips. The envelope is described as a function of altitude, however, because the prediction interpolants are based upon aircraft velocity and state, it is quite complete. The envelope limits essentially become a function of the aircraft's velocity, rates, attitude, and altitude. Since the CVA predicts the future attitude of the helicopter there is no need to develop complex envelope limits. Typically the envelopes themselves are described as a function of aircraft state and

state rates. These functions do not include any control input information consequently the acceleration terms are weighted heavily to achieve acceptable detection. However, the acceleration states are quite noisy, especially in a turbulent, or high vibration environment. This leads to excessive ‘nuisance trips’ as was found by Schroeder et. al²⁰ during their work with NASA’s VSRA. The algorithm described in this thesis does not have this problem since it uses an open loop model to predict the future attitude and altitude of the aircraft based on the current command.

7.3 SHORTCOMINGS OF OPEN LOOP MODEL BASED CVA

The greatest shortcoming of open loop model based command validation is the fact that the algorithm is based on a model of the aircraft’s performance. The algorithm is limited by the fidelity of the model used for its development. It is in this respect that the popular feedback methods have an advantage over the open loop command validation method. However, generally fly-by-wire aircraft are subjected to extensive parameter identification and modeling and thus the dynamic models are acceptable for the short-term requirements of the algorithm. Since the CVA is implemented in the digital domain using the same hardware as the FCC it will not detect hardovers that are commanded as a result of a bus failure or general system reset of the type experienced by Gubbels and Morgan²⁷. On ASRA separate systems and overall design changes will prevent these types of hardovers from occurring, thus the inability of the CVA to detect them is not an issue. Because the open loop prediction model is first order linear it introduces error as the initial conditions depart from the trim conditions

used for the model's development. In chapter 3 it was shown that these errors do not significantly affect the prediction, however it is possible that in extreme initial conditions that the errors could seriously affect the model's validity.

7.4 POSSIBLE IMPROVEMENTS TO CVA

During the CVA's development, several meetings with the pilots and project engineers took place. One improvement suggested by the pilots is of note; the incorporation of a measure of the available control power for the collective axis into the algorithm. This would effectively narrow the flight envelope when the helicopter is operating with a high collective setting since there would be little room remaining to increase the collective setting in the event of a failure. The algorithm would calculate the attitude and altitude excursions, and determine if the degree of collective required to return to the original altitude. In the event that the predicted attitude and altitude are acceptable, but excessive collective would be required, then the command is deemed not valid.

In recent months Bell helicopter has installed a load monitoring system to one of their Bell 412's. The helicopter was flown through an aggressive set of test maneuvers to determine the amount of stress the airframe endures. This data could provide a basis for development of a rate envelope limit, including attitude rates, and height rate to prevent the aircraft from undergoing structural overload as a result of fly-by-wire commands.

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APPENDIX A

Model Details

This appendix documents the details of the Simulink models used for the development of the CVA.

OPEN LOOP MODEL DETAILS

Figure A1 shows the overall model structure used for the simulation. The input for the model is transcribed from DAT recordings of control inputs, sampled at 64 Hz, and brought into MATLAB's workspace. The block labeled *Actuator Input* consists of the control inputs from the workspace, the actuator time delays, and a change in routing to account for differences in the order in which the actuator inputs are measured, and are required by the model (i.e.: longitudinal cyclic, collective, lateral cyclic, pedals). The control inputs, as received from the DAT, are biased measurements since they record the absolute deflection of the controls as opposed to the control deflection from trim. In order

to account for these differences in control displacements a program was written to reference the measured inputs to trim condition. The program allows the user to define any point as trim, and subtracts this from the all the measured controls.

Once the input signal has been rerouted and delayed it passes to the block labeled *Bell 412 rev* (shown in figure A2); here the small perturbation equations are handled by outer loop of the Bell 412 block diagram, effectively solving:

$$\begin{aligned}\dot{x} &= Ax + Bu \\ u &= [\delta e \quad \delta col \quad \delta a \quad \delta r] \\ x &= [u \quad v \quad w \quad p \quad q \quad r \quad a1 \quad b1]\end{aligned}$$

The aerodynamic biases are accounted for as starting points for the state integrators.

The aircraft orientation is solved by the block labeled *Euler Angles*, which is shown in figure A3.

The aerodynamic forces in longitudinal, lateral, and vertical body axes (X, Y, Z) are required to pass through other blocks to account for gravity forces, and Coriolis forces.

The blocks labeled X, Y & Z *Euler Equations* solve the following simultaneous equations:

$$\begin{aligned}\int \left(\dot{X} - 9.81 \sin \theta - qw + rv + Ax_0 \right) dt &= u \\ \int \left(\dot{Y} + 9.81 \cos \theta \sin \phi + pw - ru + Ay_0 \right) dt &= v \\ \int \left(\dot{Z} + 9.81 \cos \theta \cos \phi - pv + qu + Az_0 \right) dt &= w\end{aligned}$$

The block labeled data acquisition simply stores the values of the states for later analysis.

PILOT MODEL DETAILS

The complete pilot model is shown in figure AX. The aircraft rates, angles, and velocities are used as input to the model. A timed switch is used to simulate the effect of the pilot's recognition of the hardover failure. The model is structured such that control of the lateral and longitudinal cyclic is based primarily upon roll and pitch angles and rates as:

$$\begin{aligned}\delta a &= k_p p + k_\phi \phi \\ \delta e &= k_q q + k_\theta \theta\end{aligned}$$

In the event that the pitch and roll rates and angles were below a specified threshold (usually set to 10 degrees) then control of velocities in addition to rates and attitudes is attempted according to:

$$\begin{aligned}\delta a &= k_p p + k_\phi \phi + k_v v \cos \phi \\ \delta e &= k_q q + k_\theta \theta + k_u \cos \theta\end{aligned}$$

Tail rotor control is based upon yaw rate feedback with the addition of a lead-lag filter to produce the desired crossover frequency properties.

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108

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Figure A2: Bell 412 rev block diagram

Figure A3: Euler angles block diagram

Figure A4: X Euler equation block diagram

Figure A5: Y Euler equation block diagram

Figure A6: Z Euler equation block diagram

Figure A7: Overall piloted Bell 412 block diagram

Figure A8: Pilot model block diagram

APPENDIX B

Flight Model Verification

This appendix contains the time histories of both the FRL eight degree of freedom model, and the flight test data recorded at Mirabel airport September 1992. The flight test data is represented by a solid line and can be easily identified by the amount of noise in the signal. The model time histories are shown by a dashed line. Comparison of the flight test and model rates and attitudes is performed to verify the accuracy of the model. Unfortunately, flight test velocity data was unavailable, as it must be reconstructed through a computationally costly procedure. Control displacements are shown in inches, rates are given in degrees per second and angles are shown in degrees.

Figure B1: Hover longitudinal cyclic

Figure B2: Hover lateral cyclic

Figure B3: Hover tail rotor

Figure B4: Hover collective

***Figure B5:** 60 knots longitudinal cyclic*

Figure B6: 60 knots lateral cyclic

Figure B7: 60 knots tail rotor

Figure B8: 60 knots collective

Figure B9: 120 knots longitudinal cyclic

Figure B10: 120 knots lateral cyclic

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Figure B11: 120 knots tail rotor

APPENDIX C

Simulation Results

This appendix contains the results of the simulations performed to verify the operation of the CVA. The plots show the predicted attitude in degrees as a dotted line, and the actual attitude as a solid line. Predicted altitude (in meters) is shown as a dotted line, whereas actual altitude is shown as a solid line. The CVA status plot reads 1 for valid command, and 0 for an invalid command.

Figure C1: Quick stop case 1

Figure C2: Quick stop case 2

Figure C3: Quick stop case 2, 2-axis failure

Figure C4: Rapid sidestep case 1, CVA engaged

Figure C5: Rapid sidestep case 1, CVA disengaged

Figure C6: Rapid sidestep case 2

Figure C7: Rapid slalom case 1, CVA engaged

Figure C8: Rapid slalom case 1, CVA disengaged

Figure C9: Rapid slalom case 2, CVA engaged

Figure C10: Rapid slalom case 2, CVA disengaged

IMAGE EVALUATION TEST TARGET (QA-3)

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