AN EVALUATION OF
DECELERATING IFR APPROACHES
UTILIZING A HELICOPTER FLIGHT SIMULATOR

L. D. Reid, S. Advani and J. H. de Leeuw

MAY 1991
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Submitted May 1990

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May 1991

UTIAS Report No. 336
CN ISSN 0082-5255
Acknowledgements

The work reported in this document was made possible through the financial support of the Canadian Department of National Defence under contract W2207-7-AF69/01-SS. The authors would like to express their thanks to Mr. P. Sully who acted as the contractor's scientific authority, for his technical input and administrative assistance which was instrumental in helping to bring the project to a successful conclusion.

Due to the complexity of this research and development program a team of engineers, research assistants and pilots was actively involved in producing the reported results. The authors would like to thank the following:

W. O. Graf, P. R. Grant and T. Quach who served as project engineers.
R. J. Soosaar who developed the visual displays.
A. Hoppe, S. S. Mulgund and T. Bell who served as research associates.
M. Morgan and S. W. Baillie of the NAE who provided much needed technical support and piloting skills.
R. Cooper, S. Kereliuk, M. Trainor and P. Winiecki who participated as evaluation pilots.

In addition the authors are indebted to several external organizations for their support. The Flight Research Laboratory of the Canadian National Aeronautical Establishment provided hardware and significant technical assistance. Thanks are due to Dr. S.R.M. Sinclair, the laboratory Head, for his support in providing these services. The U.S. Army Aviation Systems Command, Aeroflightdynamics Directorate very generously provided a copy of the ARMCOP simulation program that formed the basis of the helicopter simulator. The authors would like to acknowledge the assistance of D. Key and M. M. Eshow of that agency, in obtaining the program and getting it running on our system.
Abstract

This project involved the development of a helicopter flight simulator and its application to study IFR landing approaches.

The helicopter simulator was implemented in the rear workstation of the UTIAS Flight Research Simulator. It employed all the major subsystems of that facility including the motion system, visual display system, sound system, and computer system. In addition the flight controls and seat of a Bell 205 helicopter were installed and interfaced to the flight computer. An electronic flight instrumentation system (EFIS) was developed to act as the pilot/helicopter interface. A side arm controller was also included in the simulation. The helicopter flight equations were based on the ARMCOP software package. The system was tuned with the help of an experienced Bell 205 test pilot.

The simulated flight test project was directed towards establishing the ability of the newly developed simulator to duplicate handling qualities results achieved in actual flight. The task selected was that of decelerating IFR approaches using flight control systems and a three cue flight director developed and tested by the National Aeronautical Establishment. A group of 6 evaluation pilots repeated flying tasks previously completed in an actual Bell 205. It was found that good agreement between simulator and test flight data was obtained except for one configuration. This configuration was found to be easier to fly in the simulator. The cause of this was felt to be incorrect simulation of the corresponding flight control system.
## Contents

1.0 INTRODUCTION................................................................. 1

2.0 SIMULATOR CONFIGURATION............................................... 2

2.1 Cockpit Layout and Controls ........................................ 2
2.2 Instrument Displays ...................................................... 3
2.2.1 EFIS ................................................................. 4
2.2.2 HUD ............................................................... 5
2.3 Forward Visual Display ................................................ 5
2.4 Sound System ............................................................. 6
2.5 Motion System ........................................................... 8
2.5.1 Hardware .......................................................... 8
2.5.2 Software .......................................................... 10
2.6 Timing ........................................................................ 11

3.0 ARMCOP ........................................................................ 19

3.1 Mathematical Model ...................................................... 19
3.2 Model Evaluation Against Flight Test Data ....................... 20
3.3 Comparisons of Stability Derivatives ............................... 22
3.3.1 ARMCOP-NAE Comparisons ................................ 22
3.3.2 ARMCOP-NASA Comparisons ................................. 23

4.0 ALTERATIONS AND ADDITIONS TO ARMCOP SOFTWARE .... 24

4.1 Pitch Dynamics ............................................................. 24
4.2 Main Rotor Implementation ............................................. 24
4.3 Landing Gear Ground Contact Model .............................. 24
4.4 Bell Horizontal Stabilizing Bar ....................................... 31
4.5 Rotor Induced Buffet .................................................... 33
4.6 Motion System Software ................................................. 34
4.6.1 Pitch/Surge Channel ................................................ 35
4.6.2 Roll/Sway Channel .................................................. 38
4.6.3 Heave Channel ....................................................... 41
4.6.4 Yaw Channel ........................................................ 43
4.6.5 Adaptive Parameter Limiting .................................... 44
4.7 Visual Display Time Delay Compensation ....................... 45
4.8 Background Motion Vibration ........................................ 48

5.0 PILOT EVALUATIONS OF SIMULATOR ................................ 51

5.1 Overall Evaluation ........................................................ 51
5.2 Motion Algorithm Tuning ................................................. 53
5.2.1 Tuning Without Adaptive Parameter Variations .......... 54
5.2.1.1 Yaw Adjustment ............................................... 55
5.2.1.2 Heave Adjustment ............................................. 56
5.2.1.3 Pitch/Surge Adjustment .................................... 57
5.2.1.4 Roll/Sway Adjustment ...................................... 57
5.2.2 Tuning With Adaptive Parameter Variations ............... 58
5.2.2.1 Yaw Adjustment ............................................... 58
5.2.2.2 Heave Adjustment ............................................. 59
5.2.2.3 Pitch/Surge Adjustment .................................... 59
5.2.2.4 Roll/Sway Adjustment ...................................... 60
5.2.2.5 General Adjustment ........................................ 60
6.0 SIMULATOR MODIFICATIONS BASED ON PILOT EVALUATIONS .......... 64
  6.1 Engine, Governor and Torque................................................. 64
  6.2 Tail Rotor .............................................................................. 67
  6.3 Pitch Attitude During Steady Turns...................................... 67
  6.4 Flight Initiation Sequence.................................................... 67
  6.5 Modifications to Helicopter Dynamics.................................. 68
  6.6 Summary............................................................................... 69

7.0 SIMULATOR ADDITIONS REQUIRED FOR THE FLIGHT TEST PROGRAM ...... 71
  7.1 Stability Augmentation .......................................................... 71
  7.2 Yaw Axis Control Modes....................................................... 71
    7.2.1 Turn Coordination Mode................................................. 72
    7.2.2 Heading Hold Mode....................................................... 73
  7.3 Flight Director....................................................................... 73
    7.3.1 Pitch Flight Director...................................................... 74
    7.3.2 Roll Flight Director....................................................... 74
    7.3.3 Collective Flight Director............................................... 74
  7.4 Flight Director Display.......................................................... 76

8.0 EXPERIMENTAL PLAN ............................................................. 77
  8.1 IFR Landing Approach Task.................................................. 77
  8.2 Simulator Flight Test Procedures............................................ 78
  8.3 Helicopter Configuration During Test Flights........................ 79
  8.4 Simulator Independent Variables.......................................... 79
  8.5 Experimental Sequence....................................................... 80
  8.6 Subjects............................................................................... 81
  8.7 Experimental Measures....................................................... 81
  8.8 Training............................................................................... 83

9.0 RESULTS AND DISCUSSION..................................................... 87
  9.1 Objective Measures............................................................. 87
  9.2 Pilot Evaluations................................................................. 90

10.0 SUMMARY AND CONCLUSIONS.............................................. 94
  10.1 Simulator Development....................................................... 94
  10.2 Experimental Program....................................................... 95

REFERENCES................................................................................. 97

APPENDIX A. PILOT BRIEFING MATERIAL

APPENDIX B. PILOT COMMENTS
### Notation

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>$g$</td>
<td>acceleration due to gravity</td>
</tr>
<tr>
<td>$i_S$</td>
<td>rotor shaft forward tilt angle</td>
</tr>
<tr>
<td>$L$</td>
<td>rolling moment</td>
</tr>
<tr>
<td>$L_i$</td>
<td>$\partial L/\partial i$</td>
</tr>
<tr>
<td>$\mathbf{LAB}$</td>
<td>transformation matrix from frame $F_B$ to frame $F_A$</td>
</tr>
<tr>
<td>$M$</td>
<td>pitching moment</td>
</tr>
<tr>
<td>$M_i$</td>
<td>$\partial M/\partial i$</td>
</tr>
<tr>
<td>$N$</td>
<td>yawing moment</td>
</tr>
<tr>
<td>$p$</td>
<td>roll rate</td>
</tr>
<tr>
<td>$q$</td>
<td>pitch rate</td>
</tr>
<tr>
<td>$r$</td>
<td>yaw rate</td>
</tr>
<tr>
<td>$s$</td>
<td>Laplace variable</td>
</tr>
<tr>
<td>$u$</td>
<td>forward speed</td>
</tr>
<tr>
<td>$v$</td>
<td>lateral speed</td>
</tr>
<tr>
<td>$w$</td>
<td>vertical speed</td>
</tr>
<tr>
<td>$X$</td>
<td>longitudinal force</td>
</tr>
<tr>
<td>$X_i$</td>
<td>$\partial X/\partial i$</td>
</tr>
<tr>
<td>$Y_{MR}$</td>
<td>side force due to main rotor</td>
</tr>
<tr>
<td>$\delta_r$</td>
<td>yaw pedal deflection</td>
</tr>
<tr>
<td>$\theta, \phi, \psi$</td>
<td>Euler angles</td>
</tr>
<tr>
<td>$x$</td>
<td>$dx/dt$</td>
</tr>
<tr>
<td>$\Delta$</td>
<td>matrix</td>
</tr>
<tr>
<td>$\Delta^T$</td>
<td>transpose of $\Delta$</td>
</tr>
<tr>
<td>$[\Delta]_{ij}$</td>
<td>the element of $\Delta$ in row $i$, column $j$</td>
</tr>
<tr>
<td>$a$</td>
<td>column matrix</td>
</tr>
<tr>
<td>$a^j$</td>
<td>the $j^{th}$ component of $a$</td>
</tr>
<tr>
<td>$a_S$</td>
<td>$a$ expressed in reference frame $F_S$ components</td>
</tr>
</tbody>
</table>
1.0 INTRODUCTION

The use of ground-based flight simulators in the training of flight crew is a well established practice. Their application to aeronautical research problems has also been quite successful. However there are relatively few simulators available for such work. In the specialized area of helicopter research only a handful of research simulators exist worldwide. The purpose of the present project was to expand the capabilities of the UTIAS Flight Research Simulator (see Figure 1.1) to include helicopter simulation and to apply it in a study of decelerating IFR helicopter landing approaches. The selection of this latter topic for study resulted from the recent evaluations of this task undertaken by the Canadian National Aeronautical Establishment (NAE)\(^1\) employing an in-flight simulator based on a BELL 205 (see Figure 1.2). We were interested in establishing the differences, if any, in pilot/helicopter performance and handling qualities ratings that would result when a ground-based simulator was utilized in place of an actual flight vehicle.

This report covers the modifications to the UTIAS Flight Research Simulator and the experiment to evaluate decelerating IFR approaches. Chapter 2 describes the configuration of the helicopter simulator. Chapter 3 outlines the ARMCOP flight equations software package and its validation for our BELL 205 configuration. Chapter 4 covers software changes and additions employed when incorporating ARMCOP into a piloted simulator environment. Chapter 5 describes pilot evaluations of the helicopter simulator and the tuning of the motion-base algorithm. Chapter 6 summarizes the modifications carried out as a result of the pilot evaluations. Chapter 7 outlines the additions to the simulation required to duplicate the systems employed by the NAE in their IFR approach study. Chapter 8 gives the details of the experimental plan and the flying task. Chapter 9 presents the results of the experiment. Chapter 10 is a summary of the modifications to the simulator and the experimental study.
FIGURE 1.1 UTIAS FLIGHT RESEARCH SIMULATOR
2.0 SIMULATOR CONFIGURATION

The UTIAS Flight Research Simulator incorporates a cab mounted on a 6 degrees-of-freedom CAE Series 300 motion-base (see Figure 1.1). Two workstations are included inside the cab; a jet transport in the front and a generic helicopter facing rearward. The helicopter workstation is shown in Figure 2.1. The helicopter simulator hardware interconnections are shown in Figure 2.2. The heart of the system is a Perkin Elmer 3250 digital computer. It samples the pilot's control inputs and solves the flight equations using a 33 Hz iteration cycle. Additional processing within each cycle is used to generate commands for the various simulator subsystems. The sound system and the motion system are addressed through a single board computer (SBC) interface. An Ethernet is used to communicate with the instrument display and the forward scene generator. Two Silicon Graphics IRIS 3130 workstations are used to generate the graphics for these displays. The experimenter communicates with the task through several consoles.

The major subsystems are shown in Figure 2.3. Most of these are described below.

2.1 Cockpit Layout and Controls

Because the first applications of the helicopter simulator were anticipated to be closely related to work involving the NAE BELL 205 in-flight simulator, it was decided to install a seat and pilot controls of the same type employed in that aircraft. Used equipment was purchased and installed as illustrated in Figure 2.1. Position transducers were fitted to the primary flight controls and linked to the PE 3250 computer through an analog-to-digital converter with 12 bit resolution (longitudinal and lateral cyclic, collective, yaw pedals and throttle). The controls were counterbalanced to represent the NAE BELL 205 with its control loading system deactivated. At the present time no control loading is provided in the simulator, the only adjustments being friction and pedal fore-and-aft location. The seat is fully adjustable and incorporates a 4 point harness. The seat also has a
right hand armrest containing a side-arm controller (both provided by the NAE) to match one of the common NAE configurations.

The side-arm controller is a Measurement Systems Inc. Model 404-G561 4-axis unit employing an NAE hand grip. It is shown installed on the right hand arm rest in Figure 2.1. Only two axes, pitch and roll, are employed in the present study. The side-arm controller replaced the centre cyclic stick during certain experimental trials.

The side-arm controller is a rigid force sensor (isometric) and does not deflect in response to pilot inputs. The outputs from each of the two active channels are low-pass filtered to remove high frequency hand tremor. A deadband is included to provide a definite centre-zero condition. And finally, a follow-up trim circuit is employed to remove the need for sustained pilot input forces during trimmed steady flight. These items are illustrated in Figure 2.4. The above features are implemented on the PE 3250 computer acting on the sampled stick output.

The side-arm controller gain was measured by applying a known force at a point 4.4 inches above the arm rest. The longitudinal gain was 0.33 volts/lb. and the lateral gain was 0.39 volts/lb. In the computer these were converted to equivalent inches of cyclic stick deflection such that the longitudinal gain was 0.76 inches/lb. and the lateral gain was 0.37 inches/lb. The deadband was taken to be 0.8 oz. longitudinal and 1.6 oz. lateral.

2.2 Instrument Displays

In order to provide the flexibility desired in a research simulator it was decided to employ an electronic flight instrumentation system (EFIS) based on an IRIS 3130 workstation. This provides the flexibility to generate dial instrument displays or modern electronic displays. An additional benefit is the ease with which display changes can be implemented.

In order to demonstrate the flexibility of the simulator, a head up display (HUD) was implemented and superimposed on the forward view display scene.
2.2.1 EFIS

The EFIS was implemented on an IRIS 3130 workstation. The cockpit monitor was the 19 inch unit supplied with the workstation. It had full colour capability and a resolution of 768 horizontal lines and 1024 pixels per line. It was run in the 60 Hz noninterlaced mode and the display was updated at 30 Hz. The location of the display was centred on the pilot's seat as shown in Figure 2.1. The top of the display's cabinet was 23° below the pilot's straight ahead line-of-sight. An Electrographics Model E264 touchscreen was installed on the face of the monitor and used for pilot/display interaction. Three instrument display formats were developed for use on the EFIS. These were the electronic display of Figure 2.5, the moving map display of Figure 2.6 (selectable by touchscreen when the electronic display is employed), and the dial instrument display of Figure 2.7. (Note that the optical distortion in these three figures is due to the camera used to photograph the displays.) Since the electronic display was employed in the present study it will be described in detail below.

Figure 2.8 is a detailed drawing of the electronic display employed in the present study. The symbology is described in Table 2.1. Additional details concerning some of the display features are given below.

The 6° glideslope and localizer displays are given by (27) and (28) in Figure 2.8. The ILS is non-standard in that the transmitters for both the localizer and the glideslope are located at the runway threshold. The displays indicate angular error from the ground-based ILS when the helicopter is more than 1900 ft. from the transmitters. The scaling is 1.5° error per division for the glideslope and 3.0° error per division for the localizer. When closer than 1900 ft. the display switches to linear displacement error with a scaling of 50 ft. per division for the glideslope and 100 ft. per division for the localizer.
2.2.2 HUD

In order to demonstrate the capability of the present system to generate head up displays, the HUD of Figure 2.9 was implemented and superimposed on the forward display scene. No problems were encountered in producing and flying this display.

2.3 Forward Visual Display

The forward visual display scene was generated by a second IRIS 3130 workstation. A full colour display was presented on a 25 inch Mitsubishi Model C-6512 BM high resolution RGB monitor. The resolution was 768 horizontal lines and 1024 pixels per line. It was run in the 33 Hz interlaced mode and the display was updated at 33 Hz. The monitor was mounted on an infinity optics window box of the type shown in Figure 2.10. This produced a virtual image of the monitor screen at optical infinity. The display field-of-view was 29° vertical by 40° horizontal. The display was centred on the pilot's seat with its mid-point depressed 8.4° below the pilot's straight ahead line-of-sight. The window box of Figure 2.10 was directly above the EFIS in Figure 2.1.

A data-base for the visual display system has been produced and it covers a range of scenarios. The general characteristics of these displays include full colour daylight to reduced colour night scenes, proper perspective based on polygon scene elements, and three-dimensional objects. Typical scenarios are shown in Figures 2.11 to 2.14 (actual photographs taken of the display monitor). Figure 2.11 shows the simplified airport employed in the present study. Figure 2.12 shows a heliport and some hills in the background. Figure 2.13 shows a wooded plateau situated in a mountain range. Figure 2.14 shows a Canadian DDH-280 Tribal Class destroyer. All of these scenes have been flown through using the current BELL 205 helicopter simulation.

The scene data-base content was limited by the speed of the IRIS workstations. In order to minimize the time delay in the visual display system the number of polygons to be processed had to be restricted. In practice it was the combination of the size and number of picture element polygons...
in the current displayed scene that limited the allowable scene content. The most critical case occurred when one large polygon obscured a second large polygon in the scene. As a result a decision on the acceptable display time delay must be made before the allowable scene content can be established. Due to the design of the IRIS workstation the present system produced time delays in 15 ms increments. The scene of Figure 2.11 can be run with a 30 ms visual time delay. The scene of Figure 2.12 can be run with a visual time delay of 30 or 45 ms, the larger value holding during the latter phases of a landing approach at the heliport. The scene of Figure 2.13 produces a 60 ms time delay. The scene of Figure 2.14 can be run with a time delay of 45 or 60 ms depending upon the location of the ship. These visual time delays can be reduced in some cases by using time delay compensation software such as that described in Section 4.7. This latter technique was employed in the present study to help eliminate the 30 ms time delay associated with the airport visual display of Figure 2.11. The airport scene was selected because the task was a landing approach involving terminal guidance.

2.4 Sound System

The hardware used to generate the cockpit noises in the BELL 205 simulator is shown in Figure 2.15. A method of playing back digitally sampled sounds in real-time is used. The core of this system is an E-mu Systems Inc. E-max digital sampling keyboard, which can store individual samples in up to sixteen separate channels and play them back in real-time while varying the pitch (or frequency) and amplitude of each channel.

The E-max receives the pitch and amplitude information from the PE 3250 computer via the single board computer and a parallel to serial converter which modifies the incoming signal to conform to a protocol called MIDI, for Musical Instrument Digital Interface. MIDI requires one start bit, eight data bits and a stop bit, and runs at 31.25 kilobaud.

An entire flight in a BELL 205 was recorded on an audio tape, with the flight conditions of the vehicle marked at various intervals. Then five distinct sounds were selected to represent the
simulated BELL 205 in any phase of flight, and the relative volumes of these were driven by flight equation parameters. These sounds were sampled for 3.5 seconds at 28 KHz and loaded into the E-max memory and looped so that they would appear to be continuous. Then the sounds were mapped into separate channels as follows:

| Channel 1       | Low idle on ground, 62% engine/rotor rpm |
| Channel 2       | High idle on ground, 89% rpm              |
| Channel 3       | Hover, 100% rpm                           |
| Channel 4       | Cruise at 60 kts, 100% rpm                |
| Channel 5       | High-torque rotor slap, 100% rpm          |

A FORTRAN program was written to drive the pitch and the volume of each of these channels according to the flight conditions during the simulation. The pitch was scaled to provide the required frequency for each sound as a function of the throttle setting, and hence the engine rpm. A linear fade-out and fade-in of the channels, combined with a smoothly-changing pitch, recreated the BELL 205 cockpit noises.

The pitch and relative volumes of the low and high idle channels were driven by the engine rpm. When it fell below 100%, the high idle sound gradually increased while the hover or cruise sound would decrease. A similar fading occurred between the low and high idle sounds, with only low idle sounds present for engine rpm below 62%.

The hover-to-cruise fading was driven by the airspeed, and the fade occurred between 30 and 40 knots. To represent the reflection of aircraft noise from the ground, the volumes of these channels were increased linearly by 10% between 30 feet above ground, and 0 feet above ground.

The high-torque rotor slap noise was driven by the rotor torque, and was scaled such that 12,000 ft-lbs of torque (or greater) generated full volume, while 3,500 ft-lbs (or less) generated no sound. This resulted in a fairly steep linear fade, providing a sudden burst of the rotor slap sound if the collective were pulled hard.
Two warning tones, generated by an external analog device, were also sampled by the E-max and assigned into the channel map. They were the deceleration profile warning tone, channel 6 — a beeper activated for 5 seconds when the vehicle reached the point of the deceleration profile onset, and a decision height buzzer, channel 7, which would sound when descending between 60 and 50 feet above ground.

2.5 Motion System

The motion system includes both hardware and software components. It is typical of the systems found on many modern training simulators. The purpose of the software is to link the flight equations to the hardware in such a manner that realistic motion cues are generated without causing the hardware to be driven against its physical limits.

2.5.1 Hardware

The hardware consists of a synergistic motion-base with six degrees-of-freedom incorporating hydrostatic bearings. It is a Series 300 CAE system delivered to UTIAS in 1983. The physical configuration can be seen in Figure 1.1 and schematically in Figure 2.16. Each of the six hydraulic actuators can be represented by a vector $\mathbf{l}_j$.

The relevant vectors relating the locations of the upper and lower bearings of the i-th actuator are shown in Figure 2.17. It can be seen that the location of the cab frame $\mathbf{F}_S$ with respect to the inertial frame $\mathbf{F}_I$ is given by

$$\mathbf{s} = \mathbf{B}_i \mathbf{l}_i - \mathbf{A}_i$$  \hspace{1cm} (2.1)

Thus the actuator length vector can be found from
Expressed in \( \mathbf{F}_i \) components Equation 2.2 becomes

\[
\mathbf{f}_i = \mathbf{A}_i + \mathbf{S}_i - \mathbf{B}_i
\]  

(2.2)

where \( \mathbf{A}_i \) \( \mathbf{S}_i \) \( \mathbf{B}_i \) are geometrical constants and \( \mathbf{S}_i \) comes from the output of the motion-base filter algorithm (see Section 4.6) as

\[
\mathbf{S}_i = \mathbf{S}_i(0) + \int_0^1 \mathbf{a}_i \, dt^2
\]

(2.4)

\( \mathbf{S}_i(0) \) is selected so as to start the simulator from a desired location. Usually this will be with all the actuators extended to one half their stroke although in special cases a bias towards some other location may be useful. The actuator length command signal employed for the \( i \)-th actuator is

\[
\mathbf{f}_i = (\mathbf{f}_i^T \mathbf{f}_i)^{1/2} \cdot \mathbf{L}
\]

(2.5)

The physical properties (both static and dynamic) of the simulator's motion-base hardware are fully documented in Reference 3. Table 2.2 lists the physical limits of the motion-base. Figure 2.18 from Reference 3 shows a typical measured frequency response function in heave.

Due to the fact that all six hydraulic actuators are used to produce each of the six degrees-of-freedom it is a property of synergistic motion systems that travel in one degree-of-freedom restricts the allowable travel of the remaining five. Also, as the frequency of the demanded motion increases, the maximum displacement of the resulting motion is reduced due to limits on the velocity and force produced by the actuators. Thus as the demanded motion frequency approaches 10 Hz the peak amplitude of a sinusoidal response approaches zero.
2.5.2 Software

The software linking the flight equations to the motion-base hardware has two conflicting goals:

(i) to provide realistic motion cues,

(ii) to prevent the motion-base from running into its physical limits.

In general this is achieved by the use of washout algorithms that remove the low frequency aircraft motion signals that can lead to large displacements. In the present case we employ a coordinated adaptive washout algorithm of the form described in Section 4.6.

The inputs to the coordinated adaptive washout algorithm (see Figure 4.8) are aircraft specific force $f_{AA}$ and angular velocity $\omega_{AA}$ as measured in the cockpit. These signals are first scaled and limited before they are fed to high-pass washout filters (the HP FILT blocks). The resulting simulator linear acceleration $\mathbf{aSI}$ is then integrated twice to produce simulator linear displacement signals. Because simulator cab tilt is used to represent low frequency longitudinal and lateral specific forces, a cross-feed is present from the specific force channel to the angular rate channel. The tilt angles are added to the outputs of the angular degrees-of-freedom high-pass washout filters to produce the simulator Euler (attitude) angles $\mathbf{U_S}$. The translational and angular simulator signals are then combined to produce the hydraulic actuator length signals $\mathbf{l_i}$. In order to provide extra protection against reaching the mechanical limits of the motion-base, as a last step, software is included to limit the actuator length, velocity and acceleration signals before they are sent to the hardware.

The adaptive nature of this algorithm operates so as to adjust the gains of the various filters in response to the current state of the simulator hardware. As the limits of the system are approached the software reacts by reducing these gains thereby minimizing the likelihood of reaching a physical limit.
2.6 Timing

Because modern flight simulators employ digital computers, they all suffer from some degree of computer-induced transport time delay. The details are quite simulator-specific and this section outlines the timing for the present helicopter simulation.

In order to synchronize the forward visual display with the solution updates of the flight equations on the PE 3250, the buffer swap signal from its IRIS 3130 workstation (indicating the start of a new display scene) is used as the system clock. It is sent to the PE 3250 and initiates the sampling of the pilot's controls and the next solution cycle. This clock runs at 33 Hz in the present simulation. In estimating the visual display time delay from pilot control input to scene display on the monitor, a number of items must be considered:

1. The sampling of the pilot's controls at 33 Hz by the analog-to-digital converter can be represented by a 15 ms time delay,

2. The processing of the flight equations by the PE 3250 and the scene calculations and graphics processing by the IRIS workstation adds 60 ms to the time delay,

3. The time taken to display the scene on the colour monitor can be considered to add 15 ms to the overall time delay,

4. The predictor form of the numerical integration employed in solving the flight equations on the PE 3250 reduces the time delay by 60 ms,

5. The use of visual display time delay compensation (see Section 4.7) on the angular degrees-of-freedom reduces their time delay by a further 30 ms.

Overall, the time delay in the visual display of translational position is 30 ms while there is no time delay on the display of angular attitude.

The instrument display is running at 30 Hz and is not synchronized to the PE 3250 processing cycle. For this reason the time delay for the instruments is not constant. The display of attitude, heading and position have time delays ranging from 15 to 45 ms while those for rate displays range from 45 to 75 ms.
The motion system is more complex than the visual system because the motion drive software contains high-pass and low-pass filters that influence its response. The motion hardware also has a dynamic effect. Measurements performed on the UTIAS Flight Research Simulator indicate that the time delay component of these dynamic effects is typically 50 ms for the angular degrees-of-freedom and 37 ms for the translational degrees-of-freedom. However, when compared with the impact of the motion system software (see Section 4.6) these time delay effects are probably not significant.
<table>
<thead>
<tr>
<th>No.</th>
<th>Symbol</th>
<th>Colour</th>
<th>Function</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>EFIS/MAP touch area</td>
<td></td>
<td>Used to toggle between the EFIS display and the moving map display</td>
</tr>
<tr>
<td>2</td>
<td>Decision height adjust touch area</td>
<td></td>
<td>Used to adjust the decision height. The up arrow increases it, while the down arrow decreases it. Readout of decision height provided by symbol (16).</td>
</tr>
<tr>
<td>3</td>
<td>Course select adjust touch area</td>
<td></td>
<td>Changes the selected course, as indicated by the course pointer (21), and readout (24).</td>
</tr>
<tr>
<td>4</td>
<td>Selected frequency on Nav Aid 1</td>
<td></td>
<td>Indicates the tuned frequency of navigational aid 1 (the green circle, symbol (19)). Displays frequency in MHz (VOR mode) or KHz (ADF mode).</td>
</tr>
<tr>
<td>5</td>
<td>Mode annunciator for Nav Aid 1</td>
<td>Green</td>
<td>Indicates the mode of operation of Nav Aid 1. If it is configured as a VOR receiver, the letters &quot;VOR&quot; are highlighted in green. If it is set up as an ADF receiver, the letters &quot;ADF&quot; are highlighted. The mode of operation can be set by touching either box.</td>
</tr>
<tr>
<td>6</td>
<td>Frequency adjust touch area for Nav Aid 1</td>
<td></td>
<td>Used to increase or decrease the tuned frequency of Nav Aid 1, whether it is configured as an ADF receiver or a VOR receiver.</td>
</tr>
<tr>
<td>7</td>
<td>Selected frequency on Nav Aid 2</td>
<td></td>
<td>Indicates tuned frequency of navigational aid 2 (symbol (20), the blue diamond).</td>
</tr>
<tr>
<td>8</td>
<td>Mode annunciator for Nav Aid 2</td>
<td>Blue</td>
<td>Same operation as item (5).</td>
</tr>
<tr>
<td>9</td>
<td>Frequency adjust touch area for Nav Aid 2</td>
<td></td>
<td>Same as item (6).</td>
</tr>
</tbody>
</table>
### Table 2.1 - Continued

**Electronic Display Symbology**

<table>
<thead>
<tr>
<th>No.</th>
<th>Symbol</th>
<th>Colour</th>
<th>Function</th>
</tr>
</thead>
<tbody>
<tr>
<td>10</td>
<td>Course deviation selector</td>
<td></td>
<td>Identifies which Nav Aid is measuring course or localizer deviation on the EHSI. If the green circle is surrounded by the white rectangle, Nav Aid 1 is measuring deviation. Otherwise, Nav Aid 2 is measuring course deviation. Toggled by touching either the diamond or the circle.</td>
</tr>
<tr>
<td>11</td>
<td>Climb gauge scale</td>
<td>White</td>
<td>Measures rate of climb to ±5000 feet per minute. Upper half of gauge measures positive rate of climb (up).</td>
</tr>
<tr>
<td>12</td>
<td>Climb rate bar gauge</td>
<td>Blue/Red</td>
<td>Indicates rate of climb against (11). If the rate of climb is positive (i.e., up) the bar appears in blue; if the rate of climb is negative, the bar appears red.</td>
</tr>
<tr>
<td>13</td>
<td>Sliding-tape altitude gauge</td>
<td>White</td>
<td>Sliding tape indicates pressure altitude in feet. Referenced against pointer of (14), the altitude readout.</td>
</tr>
<tr>
<td>14</td>
<td>Altitude readout and reference</td>
<td></td>
<td>Displays digital readout of present pressure altitude, and serves as reference to measure altitude using sliding tape.</td>
</tr>
<tr>
<td>15</td>
<td>Radar altitude readout</td>
<td>Blue or Orange</td>
<td>Indicates absolute altitude of aircraft above ground. Appears in blue when absolute altitude exceeds decision height; appears in orange otherwise.</td>
</tr>
<tr>
<td>16</td>
<td>Decision height readout</td>
<td>Blue</td>
<td>Indicates the decision height in feet, as set by the decision height adjust touch area (2).</td>
</tr>
<tr>
<td>17</td>
<td>Heading index and readout</td>
<td></td>
<td>Index for measurement of current aircraft heading, and digital readout of current heading</td>
</tr>
<tr>
<td>18</td>
<td>Compass rose</td>
<td>White</td>
<td>90° arc, aircraft heading measured at top centre against heading index (17).</td>
</tr>
</tbody>
</table>
Table 2.1 - Continued

Electronic Display Symbology

<table>
<thead>
<tr>
<th>No.</th>
<th>Symbol</th>
<th>Colour</th>
<th>Function</th>
</tr>
</thead>
<tbody>
<tr>
<td>19</td>
<td>Nav Aid 1 pointer</td>
<td>Green</td>
<td>Provides bearing to station tuned on Nav Aid 1. If the frequency is not set to that of any existing station, the indicator will point north.</td>
</tr>
<tr>
<td>20</td>
<td>Nav Aid 2 pointer</td>
<td>Blue</td>
<td>Same as (19), for Nav Aid 2.</td>
</tr>
<tr>
<td>21</td>
<td>Selected course indicator</td>
<td>Yellow</td>
<td>Points in direction of selected course, as indicated by digital readout (24). Course deviation is represented by the position of the sliding bar at the centre of the pointer; aircraft is on currently selected VOR radial when this bar is centred. The status of (10) identifies which receiver is being used to measure course deviation.</td>
</tr>
<tr>
<td>22</td>
<td>Slip indicator</td>
<td>Green</td>
<td>Provides measure of side force on aircraft, for use in making coordinated turns.</td>
</tr>
<tr>
<td>23</td>
<td>Rate of turn indicator</td>
<td>Green</td>
<td>Measures rate of turn so that a full-scale deflection corresponds to a rate 1 turn, i.e., 180° per minute.</td>
</tr>
<tr>
<td>24</td>
<td>Selected course readout</td>
<td>Yellow</td>
<td>Selected course in degrees.</td>
</tr>
<tr>
<td>25</td>
<td>Decision height warning</td>
<td>Orange</td>
<td>When radar altitude descends below decision height specified by (16), this annunciator appears and flashes for 5 seconds. After 5 seconds, the indicator remains on the screen until aircraft radar altitude exceeds decision height.</td>
</tr>
<tr>
<td>26</td>
<td>Equivalent airspeed</td>
<td>Blue</td>
<td>Indicates the equivalent airspeed (knots) in the forward direction.</td>
</tr>
<tr>
<td>27</td>
<td>Localizer deviation</td>
<td>Yellow</td>
<td>Indicates lateral displacement from the selected course. Operates in conjunction with the course deviation indicator (21).</td>
</tr>
<tr>
<td>28</td>
<td>Glideslope indicator</td>
<td>Yellow</td>
<td>Indicates aircraft vertical position with respect to a 6° glideslope. If the pointer is above centre then the aircraft is below the glideslope.</td>
</tr>
<tr>
<td>No.</td>
<td>Symbol</td>
<td>Colour</td>
<td>Function</td>
</tr>
<tr>
<td>-----</td>
<td>-------------------------------</td>
<td>--------</td>
<td>------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------</td>
</tr>
<tr>
<td>29</td>
<td>3-cue flight director symbol</td>
<td>Yellow</td>
<td>Vertical location of the symbol from aircraft symbol (32) indicates the required longitudinal cyclic input to establish the ground speed indicated by the pointer (39). The lateral position of the flight director symbol from the aircraft symbol (32) indicates the required lateral cyclic input to return the vehicle to the localizer centre. The vertical bar emerging from above or below the flight director ring indicates the collective input to return the aircraft to the glideslope. When the bar is above the ring, a downward collective input is required.</td>
</tr>
<tr>
<td>30</td>
<td>Sideslip indicator</td>
<td>Green</td>
<td>Indicates the sideslip angle of the aircraft, between the forward velocity vector and the body frame. Positive slips to the right.</td>
</tr>
<tr>
<td>31</td>
<td>Runway symbol</td>
<td>Yellow</td>
<td>Indicates proximity to the runway or ground surface. Driven by the height above ground, this symbol appears at the bottom of the artificial horizon when the altitude is 300 feet, and meets the aircraft symbol (32) when the vehicle touches down. The runway symbol is not affected by pitch, roll or heading.</td>
</tr>
<tr>
<td>32</td>
<td>Aircraft symbol</td>
<td>Green</td>
<td>A fixed symbol used to indicate aircraft attitude: Pitch angle is measured relative to the pitch scale (34), and roll attitude is specified relative to the horizon.</td>
</tr>
<tr>
<td>33</td>
<td>Horizon line</td>
<td>White</td>
<td>Represents the horizon (i.e., a pitch attitude of 0). The horizon line is tied to the blue and brown texture representing the sky and ground.</td>
</tr>
<tr>
<td>34</td>
<td>Pitch scale</td>
<td>White</td>
<td>Used to measure aircraft pitch. Aircraft nose is represented by item (32), and pitch is measured against the centre of this symbol.</td>
</tr>
<tr>
<td>35</td>
<td>Roll scale</td>
<td>White</td>
<td>Fixed scale at top of artificial horizon indicates roll attitude about horizon. Markers are placed at ±10, ±20 and ±30 degrees.</td>
</tr>
<tr>
<td>36</td>
<td>Roll indicator</td>
<td>Yellow</td>
<td>Rotating pointer that measures aircraft roll against symbol (35).</td>
</tr>
</tbody>
</table>
**Table 2.1 - Concluded**

**Electronic Display Symbology**

<table>
<thead>
<tr>
<th>No.</th>
<th>Symbol</th>
<th>Colour</th>
<th>Function</th>
</tr>
</thead>
<tbody>
<tr>
<td>37</td>
<td>Sliding-tape ground speed indicator</td>
<td>White</td>
<td>Ground speed indicator in knots. Referenced against the horizontal line on pointer (38).</td>
</tr>
<tr>
<td>38</td>
<td>Ground speed reference and digital readout</td>
<td>Yellow</td>
<td>Provides a digital readout of the present ground speed in knots. The sliding tape moves &quot;underneath&quot; this box.</td>
</tr>
<tr>
<td>39</td>
<td>Ground speed bug</td>
<td>Yellow</td>
<td>Indicates the commanded ground speed and moves with the ground speed indicator (37).</td>
</tr>
<tr>
<td>40</td>
<td>Distance measuring equipment</td>
<td>White</td>
<td>Indicates distance to the runway threshold.</td>
</tr>
<tr>
<td>41</td>
<td>ADI mode selection touch area</td>
<td>Green</td>
<td>Identifies mode of operation of the ADI. If the word &quot;CRUISE&quot; is highlighted in green, the ADI is in cruise mode. By touching this box, the ADI is put into approach mode, in which case the localizer, glideslope deviation and runway markers appear on the display (27, 28 and 31).</td>
</tr>
<tr>
<td>42</td>
<td>Engine rpm bar gauge</td>
<td>White</td>
<td>Indicates engine rpm percent against scale (43).</td>
</tr>
<tr>
<td>43</td>
<td>RPM% scale</td>
<td>White</td>
<td>Index for bar gauges (42 and 44).</td>
</tr>
<tr>
<td>44</td>
<td>Rotor rpm bar gauge</td>
<td>White</td>
<td>Indicates rotor rpm percent against scale (43).</td>
</tr>
<tr>
<td>45</td>
<td>Engine torque bar gauge</td>
<td>Green</td>
<td>Indicates engine torque in psi against scale (46).</td>
</tr>
<tr>
<td>46</td>
<td>Torque scale</td>
<td>White</td>
<td>Index for torque gauge (45).</td>
</tr>
<tr>
<td>47</td>
<td>Dimmer switch</td>
<td></td>
<td>Increases/decreases intensity of the display in a continuous range.</td>
</tr>
</tbody>
</table>
Table 2.2
Motion Limits for the UTIAS Flight Research Simulator*

<table>
<thead>
<tr>
<th></th>
<th>displacement</th>
<th>velocity</th>
<th>acceleration</th>
</tr>
</thead>
<tbody>
<tr>
<td>Roll</td>
<td>±20.8°</td>
<td>34.4°/s</td>
<td>400°/s²</td>
</tr>
<tr>
<td>Pitch</td>
<td>+21.3°, -19.8°</td>
<td>34.4°/s</td>
<td>400°/s²</td>
</tr>
<tr>
<td>Yaw</td>
<td>±23.7°</td>
<td>34.4°/s</td>
<td>400°/s²</td>
</tr>
<tr>
<td>Surge</td>
<td>+0.61, -0.70 m</td>
<td>0.80 m/s</td>
<td>10 m/s²</td>
</tr>
<tr>
<td>Sway</td>
<td>±0.59 m</td>
<td>0.80 m/s</td>
<td>10 m/s²</td>
</tr>
<tr>
<td>Heave</td>
<td>+0.55, -0.49 m</td>
<td>0.80 m/s</td>
<td>10 m/s²</td>
</tr>
</tbody>
</table>

*Based on motion of or about the centroid of the upper frame. From Reference 3 for pure motion of a single degree-of-freedom.
FIGURE 2.1  HELICOPTER SIMULATOR COCKPIT
FIGURE 2.2 SIMULATOR HARDWARE INTERCONNECTIONS
FIGURE 2.3 SIMULATOR SUBSYSTEMS
FIGURE 2.4 SIDE-ARM CONTROLLER SIGNAL PROCESSING
FIGURE 2.5 ELECTRONIC DISPLAY
FIGURE 2.6 MOVING MAP DISPLAY
FIGURE 2.7  DIAL INSTRUMENT DISPLAY
FIGURE 2.8  EFIS SYMBOLOGY
FIGURE 2.9  HUD DISPLAY
FIGURE 2.10 INFINITY OPTICS WINDOW BOX
FIGURE 2.11 AIRPORT SCENE
FIGURE 2.12  HELIPORT SCENE
FIGURE 2.13    WOODED PLATEAU SCENE
FIGURE 2.14   DDH-280 DESTROYER SCENE
FIGURE 2.15  SOUND SYSTEM
FIGURE 2.16 MOTION-BASE GEOMETRY
FIGURE 2.17  VECTORS FOR A SINGLE ACTUATOR
FIGURE 2.18  HEAVE TRANSFER FUNCTION
3.0 ARMCOP

ARMCOP is a generic helicopter simulation computer code developed by NASA and the U.S. Army Aviation Systems Command. When a copy of the code was kindly made available by the U.S. Army it was decided to employ it to represent the flight equations in the new helicopter simulator. A detailed evaluation of ARMCOP was carried out at the University of Toronto Institute for Aerospace Studies and reported in Reference 4. This section summarizes the results of this process.

The ARMCOP model was compatible with the present application because it included a BELL 205 (UH-1) parameter set and because it used a rotor disk model incorporating tip-path plane dynamics. The latter could be handled in real-time on our simulator's Perkin Elmer 3250 computer. ARMCOP had been used previously for generic handling qualities simulations but had not been correlated with actual flight data in any systematic way. Since the undertaking of this project, a paper has been published which evaluated an Augusta 109 ARMCOP piloted simulation.5 The testing was incomplete yet the results were generally positive. We have evaluated ARMCOP using three different methods: (i) comparisons against actual BELL 205 flight test data from NAE flights; (ii) comparisons of stability derivatives against those calculated from the NAE flights and data presented in Reference 6; and (iii) piloted simulator tests (see Section 5.0).

3.1 Mathematical Model

The mathematical model used by ARMCOP is a nonlinear, total force and moment model. It has ten degrees-of-freedom: six rigid-body, three rotor flapping, and rotor rotation. For helicopters with non-teetering rotors, the three rotor flapping degrees-of-freedom are the longitudinal and lateral tilts of the rotor plane as well as the coning angle. For teetering rotor systems, as is the case with the BELL 205, the coning angle is preset and remains constant. In this case, we are left with a total of nine degrees-of-freedom. The underlying model is described in References 7 to 9.
The rotor model was obtained using rotor disk theory where the blade forces were analytically integrated over the radius. The model is fully described in Reference 9. In developing the rotor blade flapping equation as well as the rotor force and moment equations, the following assumptions were made:

- The rotor blade was rigid in bending and torsion, and the twist of the blade was linear.
- The flapping angle and inflow angle were assumed to be small and the analysis utilized a simple strip theory.
- The effects of the aircraft motion on the blade flapping were limited to those due to the angular acceleration \( \dot{p} \) and \( \dot{q} \), the angular rate \( p \) and \( q \), and the normal acceleration.
- The reversed flow region was ignored and the compressibility and stall effects disregarded.
- The inflow was assumed to be uniform and no inflow dynamics were used.
- The tip-loss factor was assumed to be 1.

### 3.2 Model Evaluation Against Flight Test Data

The flight test data were obtained from BELL 205 flights performed in early 1987 at the NAE in Ottawa. The tests were conducted at an altitude of about 3000 feet under turbulence-free conditions. The data recording began with about 2 seconds of trimmed flight — trim being the aircraft state when \( u_B, \dot{v}_B, \dot{w}_B, \dot{p}_B, \dot{q}_B \) and \( \tau_B \) are all zero. This was followed by a specific control maneuver and a substantial period of controls-fixed flight. The data were originally compiled for the use of the authors of Reference 10. The final controls-fixed phase of the data sampling was to allow the measurement of the aircraft's natural modes.

The NAE BELL 205 had physical modifications performed upon it, some of which were not accounted for in the original ARMCOP test program of Reference 4. The major difference was the tail rotor. The NAE's BELL 205 was outfitted with a more powerful tail rotor (from the BELL 412). This tail rotor was also mounted on the opposite side of the helicopter — the starboard side.
The stabilizing bar model (Section 4.4) was not implemented for the flight test comparisons because the test helicopter had the bar removed. The cyclic coupling to the horizontal stabilizer in the simulation was also removed and the incidence of the stabilizer was fixed at 0 degrees (later in the project this was changed to 5° leading edge down as indicated in Section 6.5).

Figure 3.1 shows the results of a typical longitudinal cyclic input test. The ARMCOP data were generated by using the flight test control input as input following a period of trimmed flight.

The direct pitch rate response is very good. The ARMCOP $q_B$ has a slightly greater magnitude but the shapes of the curves are very close. The small magnitude differences should not create a problem.

The lateral velocity responses are unsatisfactory. ARMCOP does not respond in a manner similar to the NAE curves. The curves head in different directions and changes in direction occur at different times. The concern is how will it affect the fidelity of the simulation. Figure 3.2 is a comparison of the two lateral body frame accelerations; the ARMCOP curve being produced by the program and the flight data curve obtained by filtering and differentiating the flight data lateral velocity. Note that approximately 3/4 of a second has been removed from both the beginning and the end of the flight data acceleration due to end-point instabilities in the filtering and differentiating routine. The resultant flight data acceleration undergoes small changes (minor "jerks") while the ARMCOP acceleration has relatively severe changes. The latter would undoubtedly be felt by the pilot but since the magnitudes of the accelerations themselves are less than 0.1 g, they should not pose a significant problem to the overall simulation.

The cross-coupling response, $p_B$, was initially unsatisfactory. The ARMCOP roll rate response had a similar shape to the flight data curve, but was 3 times greater in magnitude (see Figure 3.3). The root of this problem was in the tilt of the rotor shaft. The ARMCOP data parameters for the UH-1 had a zero shaft tilt when, in fact, this helicopter flies with a physical 5 degree forward tilt of the rotor shaft. Instituting this change reduced the roll rate response to about twice the actual flight response. By increasing the tilt to 10 degrees the $p_B$ response was again
reduced, this time to the point where it was almost identical to the trace produced from the flight data.

Figure 3.3 gives a comparison of the flight data $P_B$ values with the ARMCOP responses at 0, 5 and 10 degree shaft tilts. Figure 3.1 shows the results for 10 degrees only. The result was better than expected because it did not detrimentally affect the other responses at this speed (60 knots). The rolling moment due to the main rotor is calculated as follows:

$$L_{MR} = Y_{MRh} - Q \sin(i_s)$$  \hspace{1cm} (3.1)$$

where $h$ is the moment arm from the rotor hub to the centre of gravity and $Q$ is rotor torque. The result is that the shaft tilt affects the rolling response of the aircraft. This tilt did not substantially affect the trim attitude of the aircraft. The fact that a better response was obtained at an incorrect shaft tilt angle is an indication that there is some aspect of the mathematical model which is not representative of the real helicopter. The increased shaft tilt is compensating for this insufficiency. It was decided to use the correct shaft tilt of 5 degrees in the present simulator.

The yaw rate response is less satisfactory than the pitch and roll rate responses. ARMCOP is more reactive than the flight data. At each change of the cyclic, the reactions are rapid and sharp while the real helicopter's reactions are less abrupt.

The only difference between heave rate ($W_B$) curves is in their level, which was due to an offset in the flight test data.

Further comparisons for the other degrees-of-freedom are contained in Reference 4.

3.3 Comparisons of Stability Derivatives

3.3.1 ARMCOP-NAE Comparisons

Sample $X$-derivatives $(X_u, X_v, X_w, X_{\delta e})$ are presented in Figure 3.4. For each derivative and flight condition, a number of values are given, having been calculated using data from different test
flights. The single ARMCOP value corresponding to the specific flight condition is presented along with all the NAE values. Although the ARMCOP $X_v$ derivatives are generally opposite in sign to the NAE values, the magnitudes are too small to cause concern. The ARMCOP values for the $X_u$ and $X_w$ derivatives are smaller in magnitude but the effects, again, are small. The $X_{\delta_e}$ derivatives are very similar.

The $M$-derivatives ($M_u, M_w, M_q, M_{\delta_e}$) of Figure 3.5 are basically small and generally good agreement is found between the two sets of data.

Additional comparisons can be found in Reference 4.

### 3.3.2 ARMCOP-NASA Comparisons

The NASA stability derivatives for the BELL UH-1H (BELL 205) are presented in Reference 6. The six degrees-of-freedom, quasi-static data were obtained from the manufacturer and condensed, by NASA and AVRADCOM (Aerodynamics Laboratory of the U.S. Army Research and Technology Laboratories), into a form useful for analysis. The data apply to a BELL 205 with a stabilizing bar. These data were therefore compared with the ARMCOP stability derivatives obtained with the stabilizing bar model of Section 4.4 attached. In order to account for differences in polarity conventions, all NASA derivatives involving $\delta_p$ and $\delta_e$ have had their signs reversed herein.

Some $X$-derivatives are shown on Figure 3.6 as a function of forward speed. Discrepancies occur for the $X_v$ derivative. The magnitude of $X_v$ is too small to worry about.

The majority of the $M$-derivatives (shown in Figure 3.7) are small in magnitude. Any errors are also small.

Additional comparisons can be found in Reference 4.
FIGURE 3.1 LONGITUDINAL CYCLIC INPUT
FIGURE 3.2 LATERAL ACCELERATION
FIGURE 3.3 ROLL RESPONSE FOR VARIOUS ROTOR SHAFT TILTS
FIGURE 3.5 SAMPLE M-DERIVATIVES
ARM COP-NAE
FIGURE 3.6 SAMPLE X-DERIVATIVES
ARMCOP-NASA
FIGURE 3.7 SAMPLE M-DERIVATIVES
ARMCOP-NASA
4.0 ALTERATIONS AND ADDITIONS TO ARMCOP SOFTWARE

In order to have ARMCOP operate as the flight model on the UTIAS Flight Research Simulator it had to be linked with the flight controls, the instruments, the motion system, etc. This has been implemented in a standard manner and the details are not presented here. This chapter contains other significant software alterations and additions to ARMCOP that are considered to be of general interest.

4.1 Pitch Dynamics

In Reference 4 it was found that the helicopter pitch dynamics of ARMCOP did not match the NAE flight test data in some maneuvers. A fix was described in Appendix D of that reference whereby high-pass filters were added in series with the stabilizer and main rotor pitching moment calculations. Later tests found that these filters upset the pitch trim conditions and hence they were subsequently deleted. They were not included in the present testing program.

4.2 Main Rotor Implementation

In its original form ARMCOP solved the rotor equations in a wind axis system. This caused problems near hover where the direction of the wind relative to the helicopter body axis frame could change rapidly. The problem was overcome by altering the computer code to represent solving the rotor equations in the helicopter body axis system.

4.3 Landing Gear Ground Contact Model

The landing gear model in ARMCOP has been modified to represent the skids of a BELL 205. This was carried out in order to allow the simulator to be set down following hover and was only
an approximation to the real helicopter skids. If the simulator is ever to be used to study landing operations in detail then a more complex model will be required.

The geometry and reference frames required by this model are depicted in Figure 4.1. The skids are represented by four individual skid pads, here only the left front unit is depicted. They are assumed to be located at the four corners of the original skids. The pads are supported on oleo struts. The struts are aligned with the z-axis of frame $FB$.

Frame $FL$ differs from frame $FB$ in that its origin is fixed to the Earth and is oriented such that $FB$ and $FL$ make the same $\psi$ Euler angle with respect to the inertial frame $FI$. In addition, the $\phi$ and $\theta$ Euler angles of $FB$ with respect to $FL$ and $FI$ are identical.

From Figure 4.1 it follows that the location of the i-th strut pad is given by

$$R_j = R_{CG} + \Delta_j$$  \hspace{1cm} (4.1)

When the strut is in its undisturbed state we add a subscript $(0)$ to the relevant vectors and write

$$R_{0i} = R_{CG} + \Delta_{0i}$$  \hspace{1cm} (4.2)

Define the nominal height of the i-th strut above ground level to be given by

$$H_i = -R_{0iL}^Z$$  \hspace{1cm} (4.3)

Now write Equation 4.2 in $FL$ components

$$R_{0iL} = R_{CGL} + \Delta_{0iL}$$  \hspace{1cm} (4.4)

and define the height of the helicopter centre of gravity above ground level to be
Thus from Equations 4.3 to 4.5

\[ H_i = H_{CG} - A_{oiL}^Z \] (4.6)

where

\[ A_{oiL} = L_{LB} A_{iB} \] (4.7)

Also, since the third rows of \( L_{LB} \) and \( L_{IB} \) are identical (due to the above noted \( \phi \) and \( \theta \) properties) it follows that

\[ A_{oiL}^Z = [L_{iB} A_{iB}]^Z \] (4.8)

Now let \( \xi_i \) represent the amount by which the \( i \)-th strut is extended beyond its undisturbed length, thus

\[ \xi_{iB} = [0 \ 0 \ \xi_i]^T \] (4.9)

Now from geometry, the \( z \)-component of \( \xi_{iL} \) must equal \( H_i \) when the strut is making ground contact. That is,

\[ [L_{LB} \xi_{iB}]^Z = H_i \] (4.10)

or, as in Equation 4.8,

\[ [L_{iB} \xi_{iB}]^Z = H_i \] (4.11)

or

\[ [L_{iB}]_{33} \xi_i = H_i \] (4.12)

and hence

\[ \xi_i = \frac{H_i}{[L_{iB}]_{33}} \] (4.13)
Due to the nature of the formulation of the problem it is found that Equation 4.13 only applies when \( H_i \) is negative, and

\[
L_1 = \frac{H_i}{\lambda_{iB}} \quad \text{for } H_i < 0 \quad (4.14)
\]

\[
L_1 = 0 \quad \text{for } H_i \geq 0 \quad (4.15)
\]

When a strut is making ground contact and moving over the surface both normal forces and skidding forces are generated. The skidding force acts in the direction opposite to the velocity vector representing the motion of the pad along the ground \( (\dot{\mathbf{v}}_i) \). This is shown in Figure 4.2. The normal ground reaction force is in the negative \( z \) direction of the frame \( F_L \) and is of magnitude \( F_{iN} \). The skidding force is in the \( x-y \) plane of \( F_L \) and is of magnitude \( \mu F_{iN} \) where \( \mu \) is the coefficient of skidding friction. These forces acting on the helicopter are expressed as \( F_B \) components by

\[
F_{iB} = [F_i^X \ F_i^Y \ F_i^Z]^T \quad (4.16)
\]

Since the oleo strut is modelled as a massless spring/damper system, it follows that

\[
F_i^Z = F_{iS} \quad (4.17)
\]

where \( F_{iS} \) is the oleo generated force resulting from its compression and it is applied to the airframe, where

\[
F_{iS} = K_1L_i + K_2\dot{L}_i \quad (4.18)
\]

Now represent the velocity of the \( i \)-th pad along the ground by \( \dot{\mathbf{v}}_{iS} \). It follows that

\[
\mathbf{v}_{iSL} = \begin{bmatrix} \dot{\mathbf{v}}_{iSL}^X & \dot{\mathbf{v}}_{iSL}^Y & 0 \end{bmatrix}^T \quad (4.19)
\]
Let the angle between \( V_{iS} \) and the x-axis of \( F_L \) be represented by \( \varepsilon_i \) such that

\[
\sin \varepsilon_i = \frac{V_{iS}^y}{V_{IT}} \tag{4.20}
\]

\[
\cos \varepsilon_i = \frac{V_{iS}^x}{V_{IT}} \tag{4.21}
\]

where

\[
V_{IT} = \left( \frac{V_{iSL}^x}{2} + \frac{V_{iSL}^y}{2} \right)^{0.5} \tag{4.22}
\]

From Equations 4.16, 4.20 and 4.21 it follows that

\[
L_{LB} E_{iB} = - \begin{bmatrix} \mu F_{iN} \cos \varepsilon_i & \mu F_{iN} \sin \varepsilon_i & F_{iN} \end{bmatrix}^T \tag{4.23}
\]

and if we assume that \( \phi \) and \( \theta \) are small angles it follows that

\[
L_{LB} = \begin{bmatrix} 1 & 0 & \theta \\ 0 & 1 & -\phi \\ -\theta & \phi & 1 \end{bmatrix} \tag{4.24}
\]

Thus Equation 4.23 can be approximated by (using Equations 4.16, 4.17 and 4.24):

\[
F_{i}^x + \theta F_{iS} = -\mu F_{iN} \cos \varepsilon_i \tag{4.25}
\]

\[
F_{i}^y - \phi F_{iS} = -\mu F_{iN} \sin \varepsilon_i \tag{4.26}
\]

\[
-\theta F_{i}^x + \phi F_{i}^y + F_{iS} = -F_{iN} \tag{4.27}
\]

The solution to Equations 4.25 to 4.27 (to first order in \( \phi \) and \( \theta \)) is
\[ F_i^x = F_{iS}(\mu \cos \varepsilon_i - \theta - \mu^2 \cos^2 \varepsilon_i \theta + \mu^2 \sin \varepsilon_i \cos \varepsilon_i \phi) \quad (4.28) \]
\[ F_i^y = F_{iS}(\mu \sin \varepsilon_i + \phi + \mu^2 \sin^2 \varepsilon_i \phi - \mu^2 \sin \varepsilon_i \cos \varepsilon_i \theta) \quad (4.29) \]

and \( F_i^z \) is given by Equation 4.17.

From Equation 4.24 it follows that

\[ V_{iSL} = L_{LB} V_{iSB} \]

\[ = \begin{bmatrix} V_{iSB}^x + \theta V_{iSB}^z \\ V_{iSB}^y - \phi V_{iSB}^z \\ V_{iSB}^z - \theta V_{iSB}^x + \phi V_{iSB}^y \end{bmatrix} \quad (4.30) \]

From Equations 4.19 and 4.30

\[ V_{iSB}^z = \theta V_{iSB}^x - \phi V_{iSB}^y \quad (4.31) \]
\[ V_{iSL}^x = V_{iSB}^x \quad (4.32) \]
\[ V_{iSL}^y = V_{iSB}^y \quad (4.33) \]

are the first order in \( \phi \) and \( \theta \). Thus Equations 4.20 to 4.22 can be replaced by

\[ \sin \varepsilon_i = V_{iSB}^y/V_{iIT} \quad (4.34) \]
\[ \cos \varepsilon_i = V_{iSB}^x/V_{iIT} \quad (4.35) \]
where

\[ V_{IT} = \left( V_{ISB}^2 + V_{ISB}^2 \right)^{0.5} \quad (4.36) \]

\[ V_{ISB}^x = u_B + (q_B A_{IB}^x - r_B A_{IB}^y) \quad (4.37) \]

\[ V_{ISB}^y = v_B + (r_B A_{IB}^x - p_B A_{IB}^z) \quad (4.38) \]

and \( u_B \) and \( v_B \) are helicopter ground speed expressed as components in \( F_B \). Thus the ground forces acting on the helicopter are given by Equation 4.16 (repeated here)

\[ \mathbf{E_{IB}} = \begin{bmatrix} F_i^x & F_i^y & F_i^z \end{bmatrix}^T \quad (4.39) \]

and the corresponding moments about the helicopter's centre of gravity are

\[ L_i = F_i^x A_i^y - F_i^y A_i^x \quad (4.40) \]

\[ M_i = F_i^x A_i^z - F_i^z A_i^x \quad (4.41) \]

\[ N_i = F_i^y A_i^x - F_i^x A_i^y \quad (4.42) \]

The total ground forces and moments are found by summing the above equations over the 4 skid pads with \( i \) going from 1 to 4. Required variables not explicitly derived in the above equations come from other parts of the ARMCOP program. Where the same variable is represented by more than one equation, the equation with the highest reference number is used.

The above equations apply as long as the helicopter is in motion. It was found that when landing, as the helicopter's ground speed and angular velocity approached zero, these equations generated unwanted transient motions. To avoid this problem the rate at which the ground speed is...
allowed to approach zero must be restricted. This is achieved by checking $\dot{u}_B$, $\dot{v}_B$, $\dot{p}_B$ and $\dot{q}_B$ to predict when the corresponding individual velocities will drop to zero within the next 0.33 seconds. When this condition is detected for a velocity variable, the equation for that velocity component is replaced by a low-pass filter with a time constant of 0.33 seconds whose output decays to zero starting from the last computed velocity value from the flight equations. This process reverts to the original flight equations if the applied forces and moments would cause the current velocity to approach zero in more than 0.33 seconds if it were not restricted by the low-pass filter.

The final step in bringing the helicopter to rest on the ground is achieved once all 4 skid pads are making ground contact and the resultant ground velocity of all 4 pads is less than 0.2 ft/s. At that moment $u_B$, $v_B$, $p_B$ and $q_B$ are set to zero and the Euler angles $\phi$ and $\theta$ are decayed to zero through a low-pass filter with a 0.5 s time constant. The vertical and yaw degrees-of-freedom are left alone, and still employ the original flight equations at all times.

In order to take off following the above landing process it is necessary to lift at least one skid off the ground by applying lift through the vertical flight equations (this is achieved by a collective control input). Once one of the 4 skids is no longer making ground contact then the simulator reverts to the normal flight equations and the takeoff can be completed.

4.4 Bell Horizontal Stabilizing Bar

This development follows closely that presented in Reference 4. One important dynamic aspect of the BELL 205 not included in the ARMCOP package is the stabilizing bar. This bar is a simple mechanical two-axis stability augmentor.

The bar, as described in Reference 11, pivots on the rotor shaft and is coupled to a viscous damper (see Figure 4.3).
The dynamic equations are developed in Reference 12. The bar senses both pitch and roll rates relative to the shaft-axis of the main rotor. Rate feedback is achieved by gearing the bar directly to the main rotor feathering controls (swashplate). The lag in feedback is a function of the viscous damping between the shaft and the bar. The solution to the dynamic equations (in the Laplace domain) is:

\[
\frac{\Delta A_1}{p_m} = \frac{k}{s + a} \tag{4.43}
\]

\[
\frac{\Delta B_1}{q_m} = \frac{k}{s + a} \tag{4.44}
\]

where \( \Delta A_1 \) is the lateral swashplate deflection, \( \Delta B_1 \) is the longitudinal swashplate deflection, \( p_m \) is the shaft roll rate, \( q_m \) is the shaft pitch rate, \( k \) is the gearing between bar and swashplate, \( a = \frac{1}{T} \) where \( T \) is the time constant of the stabilizing bar, and \( \hat{x} \) = Laplace transform of \( x(t) \).
The value of $k$, as chosen in Reference 12, is 0.16. The time constant, $T$, is equal to 3 seconds, therefore $a = 0.333 \text{ s}^{-1}$.

### 4.5 Rotor Induced Buffet

A rotor buffet signal of frequency equal to twice the rotor rotational frequency is employed. This signal produces a simulator sinusoidal heave acceleration whose amplitude depends upon the flight conditions. The amplitude $A$ is given by

\[
A = 0.3 L(0.25 T + 0.75 B \times F) \text{ m/s}^2
\]  

(4.45)

In Equation 4.45 $L$ is a limiter that prevents large simulator travel when rotor RPM drops below 100 (see Figure 4.4). $T$ is a thrust contribution that varies with rotor thrust as shown in Figure 4.5. $B$ is an acceleration contribution that varies with helicopter acceleration as shown in Figure 4.6. Finally $F$ is a velocity effect, reducing the contribution from acceleration to zero outside the range 15 to 65 kts as shown in Figure 4.7.
4.6 Motion System Software

The motion drive algorithm is a version of the coordinated adaptive washout algorithm detailed in References 13 to 17. This version is developed in Reference 18 and the following material comes from that reference. A block diagram is given in Figure 4.8.

A description of the filter equations, the \( N \) degree cost function and the steepest descent equations is contained in the following sections. It is first necessary to define the scaled inertial accelerations and Euler angle rates which are the inputs to the high-pass filter equations.

\[
a_c = L_{IS} \dot{f}_1 + g_f
\]  
(4.46)

\[
\hat{\beta}_c = T_s \omega_1
\]  
(4.47)

where

\[
L_{IS} = \begin{bmatrix} \cos \theta_S \cos \psi_S & \sin \phi_S \sin \theta_S \cos \psi_S & \cos \phi_S \sin \theta_S \cos \psi_S \\ - \cos \phi_S \sin \psi_S & + \sin \phi_S \sin \psi_S & \\ \cos \theta_S \sin \psi_S & \sin \phi_S \sin \theta_S \sin \psi_S & \cos \phi_S \sin \theta_S \sin \psi_S \\ + \cos \phi_S \cos \psi_S & - \sin \phi_S \cos \psi_S & \\ - \sin \theta_S & \sin \phi_S \cos \theta_S & \cos \phi_S \cos \theta_S \end{bmatrix}
\]  
(4.48)

and

\[
T_s = \begin{bmatrix} 1 & \sin \phi_S \tan \theta_S & \cos \phi_S \tan \theta_S \\ 0 & \cos \phi_S & - \sin \phi_S \\ 0 & \sin \phi_S \sec \theta_S & \cos \phi_S \sec \theta_S \end{bmatrix}
\]  
(4.49)

Using Equations 4.46 to 4.49

\[
a_c^x = f_1^x \cos \theta_S \cos \psi_S \\
+ f_1^y (\sin \phi_S \sin \theta_S \cos \psi_S - \cos \phi_S \sin \psi_S) \\
+ f_1^z (\cos \phi_S \sin \theta_S \cos \psi_S + \sin \phi_S \sin \psi_S)
\]  
(4.50)

\[
a_c^y = f_1^x \cos \theta_S \sin \psi_S \\
+ f_1^y (\sin \phi_S \sin \theta_S \sin \psi_S + \cos \phi_S \cos \psi_S) \\
+ f_1^z (\cos \phi_S \sin \theta_S \sin \psi_S - \sin \phi_S \cos \psi_S)
\]  
(4.51)
\[ a_c^z = -f_t^x \sin \theta_S + f_t^x \sin \phi_S \cos \theta_S + f_t^x \cos \phi_S \cos \theta_S + g \]

\[ \dot{\phi}_c = p_1 + (q_1 \sin \phi_S + r_1 \cos \phi_S) \tan \theta_S \]  
\[ \dot{\theta}_c = q_1 \cos \phi_S - r_1 \sin \phi_S \]  
\[ \dot{\psi}_c = (q_1 \sin \phi_S + r_1 \cos \phi_S) \sec \theta_S \]  

4.6.1 Pitch/Surge Channel

The pitch/surge filter equations are

\[ \ddot{S}_f = P_{x1}a_c^z - P_{x2}S_f - P_{x3}\dot{S}_f \]  
\[ \ddot{\theta}_{SL} = K_{x1}\frac{\dot{f}_t^x}{g} - K_{x2}\dot{\theta}_{SL} - K_{x3}\dot{\theta}_{SH} \]  
\[ \ddot{\theta}_{SH} = P_{x4}\dot{\theta}_c - K_{x3}\dot{\theta}_{SH} \]  
\[ \ddot{\theta}_S = \text{LIN}(\dot{\theta}_{SL}) + \dot{\theta}_{SH} \]

The cost function is given by

\[ J_x = \frac{1}{2N-1} [W_{x0}(a_c^z - \ddot{S}_f)^N + W_{x1}(\dot{\theta}_c - \ddot{\theta}_S)^N + W_{x2}(\dot{S}_f)^N + W_{x3}(\ddot{S}_f)^N + W_{x4}(\ddot{\theta}_S)^N + W_{x5}(\dot{\theta}_S)^N + W_{x6}(P_{xi} - P_{x10})^N + W_{x7}(P_{x2} - P_{x20})^N + W_{x8}(P_{x3} - P_{x30})^N + W_{x9}(P_{x4} - P_{x40})^N] \]  

The function of the multiplier term is to normalize the cost function over the range of -2.0 to 2.0 so that the weighting parameters for different cost function formulations would be of similar order. The first two terms weight the fidelity of the motion and tend to make the simulator motion track \(a_c^z\) and \(\dot{\theta}_c\). The terms weighting \(S_f\), \(\dot{S}_f\), \(\theta_S\) and \(\dot{\theta}_S\) are penalty terms which cause the algorithm to restrict simulator displacements and velocities. The \((P_{xi} - P_{x10})\) terms return the adaptive parameters back to their nominal values as the simulator displacements and velocities are washed out.

Application of the steepest descent technique results in the following expression for the parameter adaptation rate.

\[ \dot{P}_{xj} = -G_{xj} \frac{\partial J_x}{\partial P_{xj}} \]
Where \( G_{xj} \) is the steepest descent step size. Differentiating Equation 4.60 results in

\[
\frac{\partial J_x}{\partial P_{xj}} = \frac{N}{2^{N-1}}[W_{x0}(a^x_0 - \bar{S}^x_j)^{N-1}(\frac{\partial a^x}{\partial P_{xj}} - \frac{\partial \bar{S}^x_j}{\partial P_{xj}}) + W_{x1}(\theta_c - \bar{\theta}_S)^{N-1}(\frac{\partial \theta_c}{\partial P_{xj}} - \frac{\partial \bar{\theta}_S}{\partial P_{xj}}) + W_{x2}(\bar{S}^x_j)^{N-1}\frac{\partial \bar{S}^x_j}{\partial P_{xj}} + W_{x3}(S^x_j)^{N-1}\frac{\partial S^x_j}{\partial P_{xj}} + W_{x4}(\bar{\theta}_S)^{N-1}\frac{\partial \bar{\theta}_S}{\partial P_{xj}} (4.62) + W_{x5}(\theta_S)^{N-1}\frac{\partial \theta_S}{\partial P_{xj}} + W_{x6}(P_{x1} - P_{x10})^{N-1}\frac{\partial P_{x1}}{\partial P_{xj}} + W_{x7}(P_{x2} - P_{x20})^{N-1}\frac{\partial P_{x2}}{\partial P_{xj}} + W_{x8}(P_{x3} - P_{x30})^{N-1}\frac{\partial P_{x3}}{\partial P_{xj}} + W_{x9}(P_{x4} - P_{x40})^{N-1}\frac{\partial P_{x4}}{\partial P_{xj}}] \]

In this derivation it has been assumed that

\[
\frac{\partial}{\partial P_{xj}} (\frac{d^2x}{dt^2}) = \frac{d^2}{dt^2} \left( \frac{\partial x}{\partial P_{xj}} \right) = \frac{d}{dt} \left( \frac{\partial \dot{x}}{\partial P_{xj}} \right) (4.63) \]

The adaptive parameters are assumed to be independent

\[
\frac{\partial P_{xi}}{\partial P_{xj}} = \begin{cases} 0 & i \neq j \\ 1 & i = j \end{cases} (4.64) \]

Evaluating the derivatives

\[
\frac{\partial a^x_c}{\partial P_{xj}} = \frac{\partial a^x_c}{\partial \theta_S} \frac{\partial \theta_S}{\partial P_{xj}} = f_{nx} \frac{\partial \theta_S}{\partial P_{xj}} (4.65) \]

using Equations 4.50 and 4.65

\[
f_{nx} = -f_l^x \sin \theta_S \cos \psi_S + f_l^x \sin \phi_S \cos \theta_S \cos \psi_S + f_l^x \cos \phi_S \cos \theta_S \cos \psi_S (4.66) \]

It is assumed to a first approximation that

\[
\frac{\partial \phi_S}{\partial P_{xj}} = \frac{\partial \psi_S}{\partial P_{xj}} = 0 (4.67) \]

From equations 4.54 and 4.67

\[
\frac{\partial \dot{\theta}_c}{\partial P_{xj}} = 0 (4.68) \]

Now setting

\[
x_1 = S^x_l (4.69) \]

\[
x_2 = \dot{x}_1 (4.70) \]
The above system of Equations can be written as

\[
\dot{x}_1 = x_2
\]  
(4.78)

\[
\dot{x}_2 = P_x a - P_{x2} x_1 - P_{x3} x_2
\]  
(4.79)

\[
\dot{x}_{2P_j} = \frac{\partial P_x}{\partial x} a + P_x f_x \theta_{SP_j} - \frac{\partial P_{x2}}{\partial x} x_1 - P_{x2} x_{1P_j} - \frac{\partial P_{x3}}{\partial x} x_2 - P_{x3} x_{2P_j}
\]  
(4.81)

\[
\dot{\theta}_S = \dot{\theta}_{SH} + LIM(\theta_2)
\]  
(4.82)

\[
\dot{\theta}_{SH} = P_x \dot{\theta}_c - K_{x3} \dot{\theta}_{SH}
\]  
(4.83)

\[
\dot{\theta}_{SP_j} = \frac{\partial P_{x4}}{\partial x} \dot{\theta}_c - K_{x3} \dot{\theta}_{SP_j}
\]  
(4.84)

\[
\dot{\theta}_2 = K_{x1} \frac{f_x}{g} - K_{x1} \dot{\theta}_1 - K_{x2} \dot{\theta}_2
\]  
(4.85)

\[
\dot{\theta}_{2P_j} = -K_{x1} \theta_{1P_j} - K_{x2} \theta_{2P_j}
\]  
(4.86)

Zero initial conditions are assumed for all derivatives so it follows from Equation 4.86 that

\[
\dot{\theta}_{2P_j} = \theta_{2P_j} = \theta_{1P_j} = 0
\]  
(4.87)

therefore

\[
\dot{\theta}_{SP_j} = \dot{\theta}_{SP_j} + \theta_{2P_j} = \dot{\theta}_{SP_j}
\]  
(4.88)
Applying the assumptions of zero initial conditions for the derivatives and that the parameters are independent it follows from Equation 4.84 that

\[
\dot{\theta}_{SHP_j} = \begin{cases} 
0 & j = 1, 2, 3 \\
\dot{\theta}_c - Kx3\theta_{SHP_j} & j = 4 
\end{cases}
\quad (4.89)
\]

From Equations 4.88 and 4.89

\[
\dot{\theta}_{SP_j} = \begin{cases} 
0 & j = 1, 2, 3 \\
\dot{\theta}_{SHP_j} & j = 4 
\end{cases}
\quad (4.90)
\]

The parameter adaptation rate is given by

\[
\dot{P}_{xz} = -G_{xz} \frac{N}{2^{N-1}} [W_{x5}((1 - P_{x5})a^x_c + P_{x5}x_1 + P_{x3}x_2)^{N-1}((1 - P_{x5})f_{nx}\theta_{SPj}) \\
+ \frac{\partial P_{x1}}{\partial P_{xz}}a^x_c + P_{x5}x_1p_{xz} + \frac{\partial P_{x2}}{\partial P_{xz}}x_1 + P_{x3}x_2p_{xz} + \frac{\partial P_{x3}}{\partial P_{xz}}x_2 - W_{x1}(\dot{\theta}_c - \dot{\theta}_S)^{N-1}\dot{\theta}_{SPj} \\
+ W_{x5}(\dot{\theta}_S)^{N-1}\dot{\theta}_{SPj} + W_{x6}(P_{x5} - P_{x10})^{N-1}\frac{\partial P_{x1}}{\partial P_{xz}} + W_{x7}(P_{x2} - P_{x20})^{N-1}\frac{\partial P_{x3}}{\partial P_{xz}} \\
+ W_{x8}(P_{x3} - P_{x30})^{N-1}\frac{\partial P_{x3}}{\partial P_{xz}} + W_{x9}(P_{x9} - P_{x40})^{N-1}\frac{\partial P_{x4}}{\partial P_{xz}}]
\]

\[ \quad (4.91) \]

4.6.2 Roll/Sway Channel

The roll/sway filter equations are

\[
\ddot{S}_I = P_{y1}a^y_c - P_{y2}\dot{S}_I - P_{y3}\ddot{S}_I 
\]
\[ \quad (4.92) \]

\[
\dot{\phi}_{SL} = -K_yf^y_c/g - K_y\phi_{SL} 
\]
\[ \quad (4.93) \]

\[
\dot{\phi}_{SH} = P_{y4}\phi_c - K_y\phi_{SH} 
\]
\[ \quad (4.94) \]

\[
\dot{\phi}_S = LIM(\dot{\phi}_{SL}) + \dot{\phi}_{SH} 
\]
\[ \quad (4.95) \]

The cost function is given by

\[
J_y = \frac{1}{2^{N-1}} [W_{y0}(a^y_c - \ddot{S}_I)^N + W_{y1}(\dot{\phi}_c - \dot{\phi}_S)^N + W_{y2}(\ddot{S}_I)^N + W_{y3}(S_I)^N \\
+ W_{y4}(\dot{\phi}_S)^N + W_{y5}(\phi_S)^N + W_{y6}(P_{y1} - P_{y10})^N + W_{y7}(P_{y2} - P_{y20})^N \\
+ W_{y8}(P_{y3} - P_{y30})^N + W_{y9}(P_{y4} - P_{y40})^N] 
\]
\[ \quad (4.96) \]
Application of the steepest descent technique results in the following expression for
the parameter adaptation rate.

\[ \dot{p}_{ij} = -G_{yj} \frac{\partial J_y}{\partial P_{yj}} \]  

(4.97)

Where \( G_{yj} \) is the steepest descent step size. Differentiating Equation 4.96 results in

\[
\frac{\partial J_y}{\partial P_{yj}} = \frac{N}{2^{N-1}} [W_{y0}(a^y - \bar{S}^y)^{N-1} \left( \frac{\partial a^y}{\partial P_{yj}} - \frac{\bar{S}^y}{\partial P_{yj}} \right) + W_{y1}(\phi_c - \phi_S)^{N-1} \left( \frac{\partial \phi_c}{\partial P_{yj}} - \frac{\partial \phi_S}{\partial P_{yj}} \right)]
\]

\[ + \, W_{y2}(\bar{S}^y)^{N-1} \frac{\partial \bar{S}^y}{\partial P_{yj}} + W_{y3}(\bar{S}^y)^{N-1} \frac{\partial \bar{S}^y}{\partial P_{yj}} + W_{y4}(\dot{\phi}_S)^{N-1} \frac{\partial \dot{\phi}_S}{\partial P_{yj}} \]  

(4.98)

\[ + \, W_{y5}(\phi_S)^{N-1} \frac{\partial \phi_S}{\partial P_{yj}} + W_{y6}(P_{y1} - P_{y0})^{N-1} \frac{\partial P_{y1}}{\partial P_{yj}} + W_{y7}(P_{y2} - P_{y0})^{N-1} \frac{\partial P_{y2}}{\partial P_{yj}} \]

\[ + \, W_{y8}(P_{y3} - P_{y0})^{N-1} \frac{\partial P_{y3}}{\partial P_{yj}} + W_{y9}(P_{y4} - P_{y0})^{N-1} \frac{\partial P_{y4}}{\partial P_{yj}} \]

In this derivation it has been assumed that

\[
\frac{\partial}{\partial P_{yj}} \left( \frac{d^2 y}{dt^2} \right) = \frac{d^2}{dt^2} \left( \frac{\partial y}{\partial P_{yj}} \right) = \frac{d}{dt} \left( \frac{\partial \dot{y}}{\partial P_{yj}} \right) \]  

(4.99)

The adaptive parameters are assumed to be independent

\[
\frac{\partial P_{yi}}{\partial P_{yj}} = \begin{cases} 
  0 & i \neq j \\
  1 & i = j 
\end{cases} \]  

(4.100)

Evaluating the derivatives

\[
\frac{\partial a^y}{\partial P_{yj}} = \frac{\partial a^y}{\partial \phi_S} \frac{\partial \phi_S}{\partial P_{yj}} = f_{ny} \frac{\partial \phi_S}{\partial P_{yj}} \]  

(4.101)

using Equations 4.51 and 4.101

\[
f_{ny} = f_1\left( \cos \phi_S \sin \theta_S \sin \psi_S - \sin \phi_S \cos \psi_S \right) \]  

(4.102)

\[- f_1\left( \sin \phi_S \sin \theta_S \sin \psi_S + \cos \phi_S \cos \psi_S \right) \]

It is assumed to a first approximation that

\[
\frac{\partial \theta_S}{\partial P_{yj}} = \frac{\partial \psi_S}{\partial P_{yj}} = 0 \]  

(4.103)

From Equations 4.53 and 4.103

\[
\frac{\partial \phi_c}{\partial P_{yj}} = \frac{\partial \phi_c}{\partial \phi_S} \frac{\partial \phi_S}{\partial P_{yj}} = h_{ny} \frac{\partial \phi_S}{\partial P_{yj}} \]  

(4.104)
where

\[ h_{ny} = (q_1 \cos \phi_S - r_1 \sin \phi_S) \tan \theta_S \]  

(4.105)

Now setting

\[ y_1 = S_I^y \]  

(4.106)

\[ y_2 = \dot{y}_1 \]  

(4.107)

\[ y_1 P_{yj} = \frac{\partial y_1}{\partial P_{yj}} \]  

(4.108)

\[ y_2 P_{yj} = \frac{\partial y_2}{\partial P_{yj}} \]  

(4.109)

\[ \phi_1 = \phi_{SL} \]  

(4.110)

\[ \phi_2 = \dot{\phi}_1 \]  

(4.111)

\[ \phi_1 P_{yj} = \frac{\partial \phi_1}{\partial P_{yj}} \]  

(4.112)

\[ \phi_2 P_{yj} = \frac{\partial \phi_2}{\partial P_{yj}} \]  

(4.113)

\[ \phi_{SPy} = \frac{\partial \phi_S}{\partial P_{yj}} \]  

(4.114)

The above system of equations can be written as

\[ \dot{y}_1 = y_2 \]  

(4.115)

\[ \dot{y}_2 = P_{y1} a_y^y - P_{y2} y_1 - P_{y3} y_2 \]  

(4.116)

\[ \dot{y}_1 P_{yj} = y_2 P_{yj} \]  

(4.117)

\[ \dot{y}_2 P_{yj} = \frac{\partial P_{y1}}{\partial P_{yj}} a_y^y + P_{y1} f_{ny} \phi_{SPy} - \frac{\partial P_{y2}}{\partial P_{yj}} y_1 - P_{y2} y_1 P_{yj} - \frac{\partial P_{y3}}{\partial P_{yj}} y_2 - P_{y3} y_2 P_{yj} \]  

(4.118)

\[ \dot{\phi}_S = \dot{\phi}_{SH} + LIM(\phi_2) \]  

(4.119)

\[ \dot{\phi}_{SH} = P_{y4} \dot{\phi}_c - K_{y3} \phi_{SH} \]  

(4.120)

\[ \dot{\phi}_{SPy} = \frac{\partial P_{y4}}{\partial P_{yj}} \dot{\phi}_c + P_{y4} h_{ny} \phi_{SPy} - K_{y3} \phi_{SPy} \]  

(4.121)

\[ \dot{\phi}_2 = -K_{y1} \frac{f_y}{g} - K_{y1} \phi_1 - K_{y2} \phi_2 \]  

(4.122)

\[ \dot{\phi}_{2P_{yj}} = -K_{y1} \phi_{1P_{yj}} - K_{y2} \phi_{2P_{yj}} \]  

(4.123)
Zero initial conditions are assumed for all derivatives so it follows from equation 4.123 that

\[ \phi_2 \psi_j = \phi_1 \psi_j = \phi_1 \psi_j = 0 \]  

(4.124)

therefore

\[ \phi_{SP_j} = \phi_{SP_j} + \phi_2 \psi_j = \phi_{SP_j} \]  

(4.125)

Applying the assumptions of the zero initial conditions for the derivatives and that the parameters are independent it follows that

\[ \phi_{SP_j} = \begin{cases} 0 & j = 1, 2, 3 \\ \phi_c - K\phi_{SP_j} + P\phi_{SP_j} & j = 4 \end{cases} \]

(4.126)

From Equations 4.125 and 4.126

\[ \phi_{SP_j} = \begin{cases} 0 & j = 1, 2, 3 \\ \phi_{SP_j} & j = 4 \end{cases} \]

(4.127)

The parameter adaptation rate is given by

\[
P_{ij} = -G_{ij} \sum_{i=1}^{N-1} \left[ W_{j0} \{(1 - P_{ij})a_{ij} + P_{j1}y_1 + P_{j2}y_2\} + \sum_{i=1}^{N-1} \{(1 - P_{ij})f_{ij}\phi_{SP_j} - \frac{\partial P_{ij}}{\partial P_{ij}} \phi_{SP_j} \right] 
\]

(4.128)

\[ + W_{j3}(\phi_S) \phi_{SP_j} + W_{j4}y_1 - P_{j0} + W_{j5}(\phi_S) \phi_{SP_j} + W_{j6}(P_{j1} - P_{j10}) \frac{\partial P_{ij}}{\partial P_{ij}} + W_{j7}(P_{j2} - P_{j20}) \frac{\partial P_{ij}}{\partial P_{ij}} + W_{j8}(P_{j3} - P_{j30}) \frac{\partial P_{ij}}{\partial P_{ij}} + W_{j9}(P_{j4} - P_{j40}) \frac{\partial P_{ij}}{\partial P_{ij}} \]

\[
4.6.3 \textbf{Heave Channel}
\]

The heave filter equation is

\[ \ddot{S} = P_{s1}a_c - P_{s2}S - P_{s3} - P_{s4} \int S \, dt \]  

(4.129)

The cost function is given by

\[
J = \frac{1}{2N-1} \left[ W_{s0}(a_c - \ddot{S}) + W_{s1}S + W_{s2}S \right] 
\]

(4.130)
Application of the steepest descent technique results in the following expression for the parameter adaptation rate.

\[
P_{zj} = -G_{zj} \frac{\partial P_z}{\partial P_{zj}}
\]  

(4.131)

Where \( G_{zj} \) is the steepest descent step size. Differentiating Equation 4.130 results in

\[
\frac{\partial J_z}{\partial P_{zj}} = \frac{N}{2^{N-1}} [W_{z0}(a^z_c - \hat{S}^z)_{N-1}(\frac{\partial a^z_c}{\partial P_{zj}} - \frac{\partial \hat{S}^z}{\partial P_{zj}})] + W_{z1}(\hat{S}^z)_{N-1} \frac{\partial \hat{S}^z}{\partial P_{zj}}
\]

\[
+ W_{z2}(\hat{S}^z)_{N-1} \frac{\partial \hat{S}^z}{\partial P_{zj}} + W_{z3}(P_{z1} - P_{z10})_{N-1} \frac{\partial P_{z1}}{\partial P_{zj}}
\]

\[
+ W_{z4}(P_{z2} - P_{z20})_{N-1} \frac{\partial P_{z2}}{\partial P_{zj}} + W_{z5}(P_{z3} - P_{z30})_{N-1} \frac{\partial P_{z3}}{\partial P_{zj}}
\]

\[
+ W_{z6}(P_{z4} - P_{z40})_{N-1} \frac{\partial P_{z4}}{\partial P_{zj}}
\]

(4.132)

(4.133)

In this derivation it has been assumed that

\[
\frac{\partial}{\partial P_{zj}} (\frac{d^2 z}{dt^2}) = \frac{d^2}{dt^2} (\frac{\partial z}{\partial P_{zj}}) = \frac{d}{dt} (\frac{\partial z}{\partial P_{zj}})
\]

(4.134)

The adaptive parameters are assumed to be independent

\[
\frac{\partial P_{zi}}{\partial P_{zj}} = \begin{cases} 
0 & i \neq j \\
1 & i = j
\end{cases}
\]

(4.135)

It is assumed to a first approximation that

\[
\frac{\partial \phi_s}{\partial P_{zj}} = \frac{\partial \theta_s}{\partial P_{zj}} = 0
\]

(4.136)

From Equations 4.52 and 4.135

\[
\frac{\partial a^z_c}{\partial P_{zj}} = 0
\]

(4.137)

Now setting

\[
z_1 = \int \hat{S}^z dt
\]

(4.138)

\[
z_2 = \dot{z}_1
\]

(4.139)

\[
z_3 = \dot{z}_2
\]

(4.140)

\[
z_{1j}P_{zj} = \frac{\partial z_1}{\partial P_{zj}}
\]

(4.141)

\[
z_{2j}P_{zj} = \frac{\partial z_2}{\partial P_{zj}}
\]

(4.142)
The above system of equations can be written as

\[
\dot{z}_1 = z_2
\]  \hspace{1cm} (4.144)

\[
\dot{z}_2 = z_3
\]  \hspace{1cm} (4.145)

\[
\dot{z}_3 = P_{z1} a^z_c - P_{z2} z_3 - P_{z3} z_2 - P_{z4} z_1
\]  \hspace{1cm} (4.146)

\[
\dot{z}_3 P_{zj} = \frac{\partial P_{z1}}{\partial P_{zj}} a^z_c - P_{z2} z_3 P_{zj} - \frac{\partial P_{z2}}{\partial P_{zj}} z_3 - P_{z3} z_2 P_{zj} - \frac{\partial P_{z4}}{\partial P_{zj}} z_1
\]  \hspace{1cm} (4.147)

The parameter adaptation rate is given by

\[
P_{zj} = -G_z \frac{N}{2N-1} [W_{z0} ((1 - P_{z1}) a^z_c + P_{z2} z_3 + P_{z3} z_2 + P_{z4} z_1) N^{-1} \left\{ \frac{\partial P_{z1}}{\partial P_{zj}} a^z_c + P_{z2} z_3 P_{zj} + \frac{\partial P_{z2}}{\partial P_{zj}} z_3 + P_{z3} z_2 P_{zj} + \frac{\partial P_{z3}}{\partial P_{zj}} z_2 + P_{z4} z_1 P_{zj} + \frac{\partial P_{z4}}{\partial P_{zj}} z_1 \right\} + W_{z1} z_3^{N-1} z_3 P_{zj} + W_{z2} z_2^{N-1} z_2 P_{zj} + W_{z3} (P_{z1} - P_{z10}) N^{-1} \frac{\partial P_{z1}}{\partial P_{zj}} + W_{z4} (P_{z2} - P_{z20}) N^{-1} \frac{\partial P_{z2}}{\partial P_{zj}} + W_{z5} (P_{z3} - P_{z30}) N^{-1} \frac{\partial P_{z3}}{\partial P_{zj}} + W_{z6} (P_{z4} - P_{z40}) N^{-1} \frac{\partial P_{z4}}{\partial P_{zj}}]
\]  \hspace{1cm} (4.148)

4.6.4 Yaw Channel

The yaw filter equation is

\[
\dot{\psi}_S = P_\psi \dot{\psi}_c - K_\psi \int \psi_S dt - K_\psi^2 \psi_S
\]  \hspace{1cm} (4.149)

The cost function is given by

\[
J_\psi = \frac{1}{2N-1} [W_{\psi0} (\dot{\psi}_c - \dot{\psi}_S)^N + W_{\psi1} \dot{\psi}_S^N + W_{\psi2} \psi_S^N + W_{\psi3} (P_\psi - P_{\psi0})^N]
\]  \hspace{1cm} (4.150)

Applying the method of steepest descent

\[
\dot{P}_\psi = -G_\psi \frac{\partial J_\psi}{\partial P_\psi}
\]  \hspace{1cm} (4.151)
Differentiating Equation 4.150

\[
\frac{\partial J_\psi}{\partial P_\psi} = \frac{N}{2^{N-1}} \left[ W_{\psi_0} (\dot{\psi}_c - \dot{\psi}_S)^{N-1} \left( \frac{\partial \dot{\psi}_c}{\partial P_\psi} - \frac{\partial \dot{\psi}_S}{\partial P_\psi} \right) + W_{\psi_1} \dot{\psi}_s^{N-1} \frac{\partial \dot{\psi}_s}{\partial P_\psi} \right] + \left( \frac{\partial \dot{\psi}_s}{\partial P_\psi} \right) + W_{\psi_2} \dot{\psi}_s^{N-1} \frac{\partial \dot{\psi}_s}{\partial P_\psi} + W_{\psi_3} (P_\psi - P_{\psi_0})^{N-1} \right]
\]  
\text{(4.152)}

To a first approximation it is assumed that

\[
\frac{\partial \theta_s}{\partial P_\psi} = \frac{\partial \phi_s}{\partial P_\psi} = 0
\]  
\text{(4.153)}

From Equations 4.55 and 4.152

\[
\frac{\partial \dot{\psi}_c}{\partial P_\psi} = 0
\]  
\text{(4.154)}

Now setting

\[
\psi_1 = \int \psi_S dt
\]  
\text{(4.155)}

\[
\psi_2 = \dot{\psi}_1
\]  
\text{(4.156)}

\[
\psi_1 P_\psi = \frac{\partial \psi_1}{\partial P_\psi}
\]  
\text{(4.157)}

\[
\psi_2 P_\psi = \frac{\partial \psi_2}{\partial P_\psi}
\]  
\text{(4.158)}

The above system of equations can be written as

\[
\dot{\psi}_1 = \psi_2
\]  
\text{(4.159)}

\[
\dot{\psi}_2 = P_\psi \dot{\psi}_c - K_{\psi_1} \psi_1 - K_{\psi_2} \psi_2
\]  
\text{(4.160)}

\[
\dot{\psi}_2 P_\psi = \dot{\psi}_c - K_{\psi_1} \psi_1 P_\psi - K_{\psi_2} \psi_2 P_\psi
\]  
\text{(4.161)}

The parameter adaptation rate is given by

\[
\dot{P}_\psi = -G_\psi \frac{N}{2^{N-1}} \left[ W_{\psi_0} (\dot{\psi}_c - \dot{\psi}_2)^{N-1} (\dot{\psi}_2 P_\psi) + W_{\psi_1} \dot{\psi}_s^{N-1} \dot{\psi}_2 P_\psi \right] + \left( \frac{\partial \dot{\psi}_s}{\partial P_\psi} \right) + W_{\psi_2} \dot{\psi}_s^{N-1} \dot{\psi}_2 P_\psi + W_{\psi_3} (P_\psi - P_{\psi_0})^{N-1} \right]
\]  
\text{(4.162)}

4.6.5 Adaptive Parameter Limiting

As the adaptive parameters can go unstable, maximum and minimum values were defined in terms of the nominal parameter values (Table 4.1) to constrain any unstable behavior. The limiting values were selected so that they would not significantly modify the parameter response based on observations of unlimited parameter variations to the test inputs.
4.7 Visual Display Time Delay Compensation

In order to reduce the time delay experienced by computer generated image display systems one can apply lead filters to their input signals. This has been done in the present simulation by employing the technique described in Reference 19. The approach is outlined below.

The position command signal \( d(t) \) sent to the display system is modified by making use of its corresponding velocity signal \( V(t) \) which is also available in the flight equations computer program. Consider

\[
d_{k+1} = d_k + (f_0 V_{k+1} + f_1 V_k + f_2 V_{k-1}) T_c
\]

(4.163)

where

\[
d_k = d(kT_c), \quad V_k = V(kT_c), \quad (k = 0, 1, 2, \ldots)
\]

(4.164)

and \( T_c \) is the value of the time delay which we wish to overcome. If Equation 4.163 could be implemented and \( d_k \) is the current value of position then \( d_{k+1} \) would represent the value of position \( T_c \) seconds into the future. If \( d_{k+1} \) is then sent to the display system instead of \( d_k \) then the time delay would be cancelled.

The values of \( f_i \) in Equation 4.163 are selected by assuming that a sinusoidal \( d(t) \) represented by

\[
d(t) = e^{jWt}
\]

(4.165)

must satisfy the equality for \( W = 0 \) and \( W = W_0 \) where \( W_0 \) is the 'tuning' parameter. If we define \( U \) by

\[
U = W_0 T_c
\]

(4.166)

then it follows that
\[ f_0 = \frac{U \sin U - 2 \cos U (1 - \cos U)}{2U \sin U (1 - \cos U)} \quad (4.167) \]

\[ f_1 = \frac{2 \sin U (\sin U - U \cos U)}{2U \sin U (1 - \cos U)} \quad (4.168) \]

\[ f_2 = \frac{U \sin U - 2(1 - \cos U)}{2U \sin U (1 - \cos U)} \quad (4.169) \]

Now, because the computer program is running with an update interval of \( \Delta t \) seconds and in general \( \Delta t \neq T_c \), the values for \( V_{k+1} \), \( V_k \) and \( V_{k-1} \) are estimated by fitting the local velocity sequence \( v \) in the program by

\[ v(t) = g_0 + g_1 \sin(W_0(t - n\Delta t)) + g_2 \cos(W_0(t - n\Delta t)) \quad (4.170) \]

Define the following

\[ v_n = v(n\Delta t), \quad (n = 0, 1, 2, ...) \quad (4.171) \]

and

\[ P = W_0\Delta t \quad (4.172) \]

By evaluating Equation 4.170 at the three times producing \( v_n, v_{n-1} \) and \( v_{n-2} \) one can solve for the \( g_i \):

\[ g_0 = \frac{v_n - 2v_{n-1} \cos P + v_{n-2}}{2(1 - \cos P)} \quad (4.173) \]

\[ g_1 = \frac{(1 + 2 \cos P)v_n - 2(1 + \cos P)v_{n-1} + v_{n-2}}{2 \sin P} \quad (4.174) \]

\[ g_2 = \frac{(1 - 2 \cos P)v_n + 2v_{n-1} \cos P - v_{n-2}}{2(1 - \cos P)} \quad (4.175) \]

Thus from Equations 4.164 and 4.170 to 4.175 we can obtain the following estimates for the \( V_i \) required in Equation 4.163:
\[ V_{k+1} = g_0 + g_1 \sin(W_o(k + 1)T_c - n\Delta t)) + g_2 \cos(W_o(k + 1)T_c - n\Delta t)) \quad (4.176) \]

\[ V_k = g_0 + g_1 \sin(W_o(kT_c - n\Delta t)) + g_2 \cos(W_o(kT_c - n\Delta t)) \quad (4.177) \]

\[ V_{k-1} = g_0 + g_1 \sin(W_o((k - 1)T_c - n\Delta t)) + g_2 \cos(W_o((k - 1)T_c - n\Delta t)) \quad (4.178) \]

In the present application we take

\[ kT_c = n\Delta t \quad (4.179) \]

and Equations 4.176 to 4.178 reduce to

\[ V_{k+1} = g_0 + g_1 \sin U + g_2 \cos U \quad (4.180) \]

\[ V_k = g_0 + g_2 \quad (4.181) \]

\[ V_{k-1} = g_0 - g_1 \sin U + g_2 \cos U \quad (4.182) \]

Substituting Equations 4.180 to 4.182 into Equation 4.163 and using Equations 4.167 to 4.169 and
4.173 to 4.175 produces

\[ d_{k+1} = d_k + b_0 V_n + b_1 V_{n-1} + b_2 V_{n-2} \quad (4.183) \]

where

\[ b_0 = \frac{U + \sin U(1 - 2\cos P)}{2W_o(1 - \cos P)} + \frac{(1 - \cos U)(1 + 2\cos P)}{2W_o \sin P} \quad (4.184) \]

\[ b_1 = \frac{(\sin U - U)\cos P}{W_o(1 - \cos P)} - \frac{(1 - \cos U)(1 + \cos P)}{W_o \sin P} \quad (4.185) \]

\[ b_2 = \frac{U - \sin U}{2W_o(1 - \cos P)} + \frac{1 - \cos U}{2W_o \sin P} \quad (4.186) \]
The above sets of equations allow one to generate a leading phase for \( d(t) \) at low frequencies from data contained within the computer algorithm. One must choose values for \( T_{C} \) and \( W_{O} \) that produce the desired effect.

The transfer function between \( d_{k+1} \) and \( d_{k} \) in Equation 4.163 can be computed by representing \( d(t) \) by a pure sine wave and solving for \( d_{k+1} \). Plots of amplitude and phase of the corresponding frequency response function can then be made. This has been done for \( \Delta t = 30 \) ms, \( T_{C} = 30 \) ms and \( W_{O} \) of 10, 15 and 20 r/s and the results are plotted in Figures 4.9 to 4.11. In these plots the solid line represents the measured transfer function and the dashed line the phase associated with 30 ms of pure time advance. A perfect time delay compensation algorithm would have a gain of zero dB and a linearly increasing phase with frequency. It can be seen that the desired effect is achieved over the frequency range 0 to 4 Hz, with the case for \( W_{O} = 20 \) r/s marginally the best. Since the pilot control activity occurs typically in the frequency band 0 to 3 Hz it was decided to use the delay compensation algorithm with \( W_{O} = 20 \) r/s in the present study.

Ideally the algorithm should be applied to 6 signals, the 3 linear and 3 angular displacement variables sent to the IRIS workstation. However there was not enough time available on the PE 3250 to do this. As a compromise it was only applied to the 3 angular degrees-of-freedom since these tended to be the most tightly controlled loops and thus were the most sensitive to time delay effects.

4.8 Background Motion Vibration

As mentioned in Reference 4 the numerical integration scheme employed by ARMCOP tends to be a little noisy, particularly in the translational acceleration degrees-of-freedom. This can only be detected by pilots in the simulator through the motion system response, mainly in the heave degree-of-freedom. This effect can be seen in Figure 4.12 where both the flight equations solution for specific force and the measured simulator specific forces are displayed. No means for removing
this injected noise could be found. However the pilots did not object to its presence, accepting it as reasonable low level helicopter vibrations.
### TABLE 4.1 ADAPTIVE PARAMETER LIMITS

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Minimum Value</th>
<th>Maximum Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>$P_{x1}$</td>
<td>0.0</td>
<td>$P_{x10}$</td>
</tr>
<tr>
<td>$P_{x2}$</td>
<td>$P_{x20}$</td>
<td>$2 \cdot P_{x20}$</td>
</tr>
<tr>
<td>$P_{x3}$</td>
<td>$P_{x30}$</td>
<td>$2 \cdot P_{x30}$</td>
</tr>
<tr>
<td>$P_{x4}$</td>
<td>0.0</td>
<td>$P_{x40}$</td>
</tr>
<tr>
<td>$P_{y1}$</td>
<td>0.0</td>
<td>$P_{y10}$</td>
</tr>
<tr>
<td>$P_{y2}$</td>
<td>$P_{y20}$</td>
<td>$2 \cdot P_{y20}$</td>
</tr>
<tr>
<td>$P_{y3}$</td>
<td>$P_{y30}$</td>
<td>$2 \cdot P_{y30}$</td>
</tr>
<tr>
<td>$P_{y4}$</td>
<td>0.0</td>
<td>$P_{y40}$</td>
</tr>
<tr>
<td>$P_{z1}$</td>
<td>0.0</td>
<td>$P_{z10}$</td>
</tr>
<tr>
<td>$P_{z2}$</td>
<td>$P_{z20}$</td>
<td>$2 \cdot P_{z20}$</td>
</tr>
<tr>
<td>$P_{z3}$</td>
<td>$P_{z30}$</td>
<td>$2 \cdot P_{z30}$</td>
</tr>
<tr>
<td>$P_{z4}$</td>
<td>$0.25 \cdot P_{z40}$</td>
<td>$P_{z40}$</td>
</tr>
<tr>
<td>$P_{\psi}$</td>
<td>0.0</td>
<td>$P_{\psi0}$</td>
</tr>
</tbody>
</table>
FIGURE 4.1 HELICOPTER REFERENCE FRAMES
FIGURE 4.2  LANDING SKID PAD
inclination of mast from vertical

angle of bar relative to mast

damper

FIGURE 4.3 STABILIZING BAR
FIGURE 4.4  BUFFET PARAMETER L

FIGURE 4.5  BUFFET PARAMETER T
FIGURE 4.6 BUFFET PARAMETER B

FIGURE 4.7 BUFFET PARAMETER F
FIGURE 4.8 ADAPTIVE MOTION ALGORITHM
FIGURE 4.9 VISUAL TIME DELAY COMPENSATION TRANSFER FUNCTION, \( W_0 = 10 \, \text{rad/s} \)
FIGURE 4.10 VISUAL TIME DELAY COMPENSATION TRANSFER FUNCTION,
$W_0 = 15 \text{ r/s}$
FIGURE 4.11 VISUAL TIME DELAY COMPENSATION TRANSFER FUNCTION, $W_0 = 20 \text{ rad/s}$
FIGURE 4.12 SPECIFIC FORCE NOISE
5.0 PILOT EVALUATIONS OF SIMULATOR

Once the helicopter simulator was operational and debugged the next step involved its subjective evaluation by an NAE test pilot with considerable experience on the NAE Bell 205 (pilot 1 in Table 8.2). This was an iterative process in which deficiencies noted by the test pilot were corrected and the resulting system evaluated at a subsequent date. The bulk of the items reported in this section were identified and corrected over a 10 month period. This involved 4 days of evaluation flying on the part of the test pilot.

5.1 Overall Evaluation

The initial evaluation process was carried out without simulator motion. This was intended to eliminate gross deficiencies before adding the complexities related to motion. Once the helicopter was performing in an acceptable manner the evaluation and tuning of the simulator's motion system took place (as reported in Section 5.2). This was followed by final adjustments to the complete simulation based on pilot evaluations with all systems operational (including motion). This section describes the deficiencies noted by the test pilot in all areas except simulator motion. The latter is covered in Section 5.2. The many positive evaluation responses are not reported here because no systematic attempt was made to obtain them.

The maneuvers employed during the evaluation process exercised all degrees-of-freedom and were representative of those employed when assessing helicopter handling qualities. Nap-of-the-Earth and hovering evaluations were limited due to the restricted field-of-view provided by the visual display system.

The first trials were carried out with the motion system inoperative and the stabilizer bar on.

The following problems were noted:

(1) The sideslip digital display should also indicate the polarity of $\beta$. 
(2) The ball instrument display was too lively.

(3) Level flight at 60 kts and 90 kts required too much right pedal and the collective lever was physically too low. The torque indication was too low.

At this point the trim positions of the controls were adjusted to the pilot's satisfaction and the pedal gain increased by 50%. The evaluation then proceeded.

(4) During level flight at 120 kts excessive pedal deflection was required to carry out sideslip corrections. The torque indication was too low.

(5) During level flight at 30 kts the pedal gain was too low. The torque indication was incorrect.

(6) During hover the torque indication was incorrect.

(7) During dynamic maneuvering at 60 kts the simulator roll response was not like a Bell 205. The roll rate dropped off during roll response. The phugoid mode was too well damped.

(8) During flight at 80 kts the Dutch roll mode was too powerful.

The stabilizer bar was then removed from the simulation and the testing continued. This seemed to improve the roll response during aggressive maneuvering.

(9) Pitch-roll coupling seemed too strong.

(10) Too much collective was required during steep turns.

(11) Trimming the helicopter for level flight at 30 kts was difficult with the stabilizer bar removed.

(12) During flight at 60 kts the displayed torque response to collective pulses was too fast. Rotor overspeed was noted during maneuvers.

(13) During flight at 100 kts the phugoid mode was found to be a little overdamped. There was a little too much pitch damping. There was not as much speed change during maneuvers as anticipated. The rotor tended to overspeed when offloaded through the collective.

(14) During flight at 40 kts the pitch damping was too low.
At this point the stabilizer bar was reactivated. A series of maneuvers was flown, some near the ground at the airport location. In response to pilot complaints about yaw response, the pedal gain was increased by a further 50%.

(15) Collective to yaw and pitch coupling seemed excessive.

(16) Collective response seemed to be sluggish or delayed.

(17) The pilot experienced problems due to the limited field-of-view while hovering near the ground. This resulted from a confusion of yaw and lateral velocity cues.

(18) Rotor RPM droop following collective increases was too great.

(19) During 30° banked turns performed at 80 kts the steady-state pitch attitude was 5° higher during right-hand turns than during left-hand turns.

The time delay compensation was tested next (see Section 4.7). The pilot found that this visual display delay reduction improved the simulation and improved his handling qualities rating by about one unit.

5.2 Motion Algorithm Tuning

The simulator motion system is described in Sections 2.5 and 4.6. In order to obtain the maximum benefit from its use it is necessary to select the parameters of the motion-drive algorithm to suit the Bell 205 helicopter simulation. This was achieved by starting with the parameter values used in the previous simulation of a Boeing 747. These parameters were first modified during testing by UTIAS project engineers. This produced a reasonable starting point for pilot evaluations by the NAE test pilot.

The motion tuning was carried out with the visual display delay compensation present and the stabilizer bar off. The adaptive feature of the motion-drive algorithm was initially turned off by setting the steepest descent gains $G_{ij}$ to zero. This allowed the initial tuning to be carried out as if the algorithm were classical washout.
When the motion-base was first activated the test pilot found that he was creating a bad lateral PIO vibration. This was found to be caused by some incorrect values for parameters specifying the pilot's location relative to the helicopter's centre of gravity. This was corrected and testing proceeded.

5.2.1 Tuning Without Adaptive Parameter Variations

In order to minimize any motions due to poor choices of the initial values of algorithm parameters, tuning began with all the input scaling factors set to 0.3 (Kfx, Kfy, Kfz, Kωp, Kωq, Kωr). (See Section 4.6 for the meanings of system parameters.) In the material reported below the pilot comments are in italics. We intended to tune systematically channel-by-channel but were distracted from this by the following complaint: *There is a strong kick on roll inputs that shouldn't be there.*

On further questioning and observation this was determined to be a high-frequency, low-amplitude spike in roll at roll entry. This was probably due to the vertical offset from CG to the simulator centroid location causing a spike in lateral acceleration at the centroid location, which was feeding through the low-pass tilt-coordination filter into the roll channel. The roll tilt rate limit was already very low (1 deg/s), so the break frequency of this filter was probably too high, and this relatively high frequency input was not attenuated enough. Therefore K_y1 was changed to alter the break frequency from 8 r/s to 5 r/s. As well, since K_y2 was unchanged, this implied that the damping ratio had gone up from 1 to 1.6. *This feels better — the kick is mostly gone.*

We next wanted to try to bring the damping ratio back down to 1 so as not to change things unnecessarily. K_y2 was changed from 16 to 10 to bring the damping ratio of the roll LP filter back to 1. *A little of that kick is now back.*

We therefore decided to bring the damping ratio back up. K_y2 was changed from 10 to 15 to bring the damping ratio up to 1.5. *There is still a kick though it's been reduced, it should be less.*
In order to reduce the break frequency further, \( K_y1 \) was changed to bring the break frequency down from 5 \( r/s \) to 3 \( r/s \). As well, since \( K_y2 \) is unchanged, the damping ratio was now 2.5. *The kick is now completely gone — it feels more like a helicopter.*

We now wanted to try to reduce the damping ratio as 2.5 seemed exceedingly high. \( K_y2 \) was changed from 15 to 6 to bring the damping ratio down to 1. *A bit of the kick is now back.*

In order to raise the damping ratio, \( K_y2 \) was changed from 6 to 9 to bring the damping ratio up to 1.5. *Feels fine — the kick is gone.*

Now that this roll kick complaint was out of the way, we were able to start our more systematic motion adjustment channel by channel. First, we wanted to boost all the input gains as 0.3 seemed on the low side. All input scaling was set to 0.5 (\( K_{fx}, K_{fy}, K_{fz}, K_{\omega p}, K_{\omega q}, K_{\omega r} \)).

We wanted to keep all scaling in the scaling block and not use the filter coefficients for scaling. \( P_y4 \) was changed from 0.75 to 1.0. *This feels better, the motion on roll inputs is more positive.*

### 5.2.1.1 Yaw Adjustment

We now wanted to start tuning the yaw filter. The first complaint was that there was a lateral motion in the wrong direction on yaw pedal inputs. We speculated that since this was a sway motion and it was of the wrong sign, it might be caused by CG to simulator centroid offset parameters of the wrong sign. We found that due to confusion with frame transformations, these offsets indeed had the wrong sign. This was corrected in the aircraft motion equations. *That's much better — the motion is now in phase with the visuals.* Note that at this point, the pilot was asked whether the motion simplified the flying task (i.e., whether motion cues were helpful to the pilot). The pilot was emphatic in his endorsement of motion and noted that certain tasks would have been impossible for him the previous day (with no motion) such as pirouettes performed around the control tower, but were now quite manageable with motion.
At this point, there were no real complaints with the yaw motion. However we decided to try further adjustments to see if it could be improved without addressing a specific complaint. We decided to loosen the yaw high-pass filter: $K_{\psi 2}$ was changed from 1 to 0.75 (note that this is the break frequency of a first-order low-pass filter when $K_{\psi 1} = 0$). *Yes, it does seem to feel a little better.*

We therefore proceeded to open the filter further. $K_{\psi 2}$ was changed from 0.75 to 0.6. *Can't see much of a difference.*

Since there was no point in generating more motion when it can't be felt, we reversed this step. $K_{\psi 2}$ was changed from 0.6 to 0.75.

### 5.2.1.2 Heave Adjustment

The heave channel was first evaluated with high-frequency motion generated by step changes in the collective. *It feels good but a bit weak.*

We then increased the input scaling on the vertical channel. $K_{fx}$ was increased from 0.5 to 0.75. *I can't really feel the difference.* We were still not close to the motion limits and could go further.

$K_{fx}$ was increased from 0.75 to 1.0. There, now that feels good — it's much more positive.

Although we were still not close to the motion limits and could go further, we were concerned about how this would affect us later in sustained vertical acceleration. The pilot mentioned at this point that even a step input in collective will only produce a pulse in vertical acceleration in a helicopter.

At this point, we tested maneuvers such as steady climbs and drops to try to generate some low-frequency vertical inputs. The aircraft performance was such that a steady vertical acceleration could not be generated (the helicopter quickly settles to a 1000 fpm climb rate). On attempting to
generate steady downward acceleration, the helicopter tended to go unstable due to inaccuracies in the model.

5.2.1.3 Pitch/Surge Adjustment

All pitch/surge maneuvers we attempted elicited no complaints. In fact, there were very few maneuvers that could exercise these degrees-of-freedom. The only way steady pitch angles could be generated was in high-speed forward flight. But these were so modest (5 to 10 degrees on the artificial horizon) that the pilot could not detect whether the simulator pitch attitude differed from these or not. We also tested high frequency longitudinal starts and stops about hover and the pilot was quite satisfied.

5.2.1.4 Roll/Sway Adjustment

The first problem noted here was actually a problem with the vertical filter. During steady banked turns, the body-axis vertical specific force (which is used as input to the washout filters) holds a steady state value not equal to g for as long as the turn is held. Since we are using a second-order vertical filter, this results in unacceptably large offsets from the neutral position during steep banked turns. To improve on this, $P_{z4}$ was changed from 0 to 1.6; $P_{z3}$ changed from 16.0 to 16.8; $P_{z2}$ changed from 8.0 to 8.1. This leaves the second-order high-pass filter unchanged with $\omega_n = 4 \text{ r/s}$ and $\zeta = 1$, but adds a very mild first-order high-pass filter in series (thus yielding a third-order filter) with $\omega_2 = 0.1 \text{ r/s}$. This seemed to bring the motion-base back to neutral quite nicely during the steady banked turns. Some high frequency collective inputs were tried to verify that this did not have a deleterious effect on the vertical motion response and the pilot could not feel any difference.

We next tried some sidestep maneuvers from hover in order to check the response to lateral accelerations, but the simulated aircraft response was quite unstable and unrepresentative of the real aircraft. There was no point in attempting to tune for this unrealistic motion.
We also attempted some descending flights with steady sideslip but encountered the same problem. (This was corrected later.)

We next checked the response to higher frequency roll inputs. At first the pilot liked the response but then noted some anomalous motion just at roll input, and was unsure in which degree-of-freedom this was occurring. From outside the simulator, there was no noticeably wrong motion. This type of problem is typically from one of two sources — the sway channel or the tilt-coordination channel. Since we had already thoroughly deadened the roll tilt-coordination, it was felt that the culprit might rather be the sway channel, so we made that filter more restrictive. \( P_y^2 \) was changed from 16 to 25 and \( P_y^3 \) was changed from 8 to 10. This boosted the break frequency of this high-pass filter from 4 to 5 t/s while keeping the damping ratio at 1. The pilot was very happy with this change as it got rid of the undesirable motion.

This concluded the tuning session for the non-adaptive form of the algorithm. The resulting parameter set is given in Table 5.1 under the heading "Initial Values".

5.2.2 Tuning With Adaptive Parameter Variations

Having tuned the motion-drive algorithm first with the adaptive feature suppressed, the next step was to gradually introduce the adaptive feature.

5.2.2.1 Yaw Adjustment

Although no complaints could be elicited from the pilot concerning yaw motion, we decided to implement some changes to see if they might further improve the motion.

The first change made was neutral as it boosted the input scaling in yaw but lowered the filter gain by the same amount. *Can't really feel any difference, it still feels good.*

In order to bring in the adaptiveness of the filter, \( G_{yy} \) was changed from 0 to 40 (in steps).

*I still can't feel any difference.*
In order to put more emphasis on motion realism, we boosted $W_{\psi_0}$ from 0.1 to 0.8. No, it still feels the same.

In order to get still more adaptiveness out of the filter, $G_{\psi}$ was changed from 40 to 60. Still no change was felt by the pilot but he emphasized that the motion felt fine.

5.2.2.2 Heave Adjustment

We tried out some vertical maneuvers and although they felt alright, the pilot thought he might like more of a kick in response to collective inputs. Therefore, $K_{fx}$ was changed from 1 to 1.5. Yes, that's definitely more positive — let's leave it that way.

In order to bring in the adaptiveness, $G_{z1}$ was changed from 0 to 10 (in steps). Some oscillatory behaviour was apparent at the latter value, so it was reduced to 5 where no instability was noted.

In order to try to reduce vertical excursions, we next changed $W_{z2}$ from 15 to 20. Now there seems to be a ringing in the motion.

The previous motion was driving the filter into a limit cycle. Rather than reduce $W_{z2}$, we decided to increase the weighing on the velocity term as this tends to add a lead term to the weighting. $W_{z1}$ was changed from 7.5 to 10. The ringing is now reduced and at a higher frequency — it feels good — it feels like a structural vibration — let's leave it in.

The pilot performed some general maneuvers at this point. The vertical motion is too strong now — I'm getting too much of a kick in response to collective inputs. $K_{fx}$ was changed from 1.5 to 1.25. Yes, that feels better — not quite as strong.

5.2.2.3 Pitch/Surge Adjustment

We tried the limited number of pitch/surge maneuvers which we had tried with the classical filters and could find nothing wrong with the motion. For this reason no changes were made.
5.2.2.4 Roll/Sway Adjustment

Once again, we found ourselves in the position of having motion which generated no criticism. No further changes to parameter values were made.

5.2.2.5 General Adjustment

We soon discovered a maneuver which generated enough motion to drive the motion-base into its (software) limits. The problem arose upon recovery from a left descending turn (which was performed with about 30 degrees of bank and 800-1000 fpm rate of descent).

This was a transient problem caused by the combination of motions in the heave, yaw and roll degrees of freedom during the turn recovery. We decided to try to correct the problem by using the adaptive parameters on the heave filter. $W_{z2}$ was changed from 20 to 30 in order to increase the penalty on simulator vertical displacement. The maneuver was repeated and the limits were hit again.

In order to increase the lead term, we increased $W_{z1}$ from 10 to 15. There was still no observable effect and the limits were reached again.

We next decided to make the heave filter more restrictive. We changed $P_{z4}$ from 1.6 to 3.2; $P_{z3}$ from 16.8 to 17.6; and $P_{z2}$ from 8.1 to 8.2. If the third-order high-pass filter in heave is decomposed into a second-order filter in series with a first-order one, the second-order filter is left unchanged, while the break frequency of the first-order filter has been increased from 0.1 r/s to 0.2 r/s. This was enough to avoid hitting the soft limits. Although they may be hit in a more severe maneuver of the same type, we were reluctant to make the filters more restrictive just to accommodate one, presumably rare, severe maneuver.

This concluded the motion tuning process. The final parameters are given as "Final Values" in Table 5.1.
Table 5.1
Adaptive Motion Algorithm Parameters

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**Adaptive Motion Algorithm Parameters**

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Table 5.1 - Concluded

Adaptive Motion Algorithm Parameters

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</table>
6.0 SIMULATOR MODIFICATIONS BASED ON PILOT EVALUATIONS

As a result of the pilot evaluation process a number of changes were made to the simulator. Some of these were implemented on-line during testing and are reported in Section 5. Others were quite minor in nature and are not documented in this report. This section details the more significant changes carried out between pilot evaluation sessions. Where it was not possible to completely correct the problem this is indicated. In general it was possible to produce a simulation which was acceptable to the test pilot as long as flights were restricted to moderate maneuvering away from the ground. Severe maneuvering, nap-of-the-earth flying and hovering flight are not adequately represented by the present simulator configuration.

6.1 Engine, Governor and Torque

Several problems were noted in Section 5 relating to the power train. In particular rotor RPM fluctuations were excessive in some cases. Changes to the model to correct this involved the inclusion of a fuel flow controller model and a feedforward from the collective control. In addition, parameter values were adjusted to match the response of the NAE Bell 205 helicopter.

Figure 6.1 gives the block diagram of the revised form of the model. The following notation applies to Figure 6.1:

- $\Omega$: main rotor angular velocity
- $\Omega_{IC}$: the initial value of $\Omega$ at the start of a flight
- $\Omega_{SET}$: rotor angular velocity set point in the governor
- $\Delta\Omega$: rotor angular velocity deviation from $\Omega_{SET}$
- $\Delta W_f$: fuel flow increment as commanded by the governor
- $W_{FSET}$: feedforward fuel flow command from the collective lever setting
- $W_f$: output from the fuel flow controller
developed engine power
engine torque associated with HP
total torque required by the main rotor and the tail rotor to overcome aerodynamic drag effects
residual torque available to accelerate the main rotor, gear box, tail rotor, etc.

The feedforward scheme is based on a fuel flow schedule that approximates the steady-state fuel flow (WFSET) required to fly straight and level at a particular collective lever setting $\delta_c$. This is given by

$$WFSET = (3,379\delta_c - 18,082)Q_I(550K_E) \text{ lb./hr.} \quad (6.1)$$

The intermediate estimate of fuel flow is given by

$$WF = \Delta W_f + WFIC + WFSET \quad (6.2)$$

where $\Delta W_f$ is the output of the RPM governor. In the present case since $\Omega IC = \Omega SET$ the initial value of $\Delta W_f$ is zero. Hence the start-up value of WF is given by

$$WF = WFIC + WFSET \quad (6.3)$$

with WFSET based on selected trim values for $\delta_c$ and $\Omega$. WFIC (constant for a given flight) is the offset of WFSET from the required fuel flow at the start of the run given by (taking values at the initial start-up)

$$WFIC = (QR\cdot\Omega/550K_E) - WFSET \quad (6.4)$$
In general WFIC will be a small correction term. The governor equation is given by

\[
\Delta W_f = G_1 \Delta \Omega + G_2 \Delta \Omega + G_3 \int \Delta \Omega \, dt
\]  

(6.5)

Before passing on the fuel flow to the engine the fuel scheduler carries out several checks, and as a result may modify the fuel flow of Equation 6.2. First it limits the fuel flow to be less than or equal to the flow selected by the throttle position ($\delta_T$) $WFTHR$. Then it limits it to be greater than or equal to the minimum allowed fuel flow $WFMIN$. Thus the output fuel flow $W_f$ is given by

\[
W_f = WF
\]  

subject to

\[
WFMIN \leq W_f \leq WFTHR
\]  

(6.7)

where

\[
WFTHR = \frac{\delta_T}{100} \times WFMAX
\]  

(6.8)

and WFMAX is the maximum allowable fuel flow.

In addition the following limits are placed on the system.

(i) If $\delta_T < 90\%$ then $WFSET = 0$.

(ii) If $WF \geq WFTHR$ and $\Omega < 32.23 \, r/s$ then $G_3 = 0$ in Equation 6.5

(iii) $\Omega$ is limited at all times to $\Omega > 10 \, r/s$.

The parameter values selected for the present application are contained in Table 6.1.

The complaints about the dynamics of the displayed torque on the instrument panel were addressed by changing the displayed torque from rotor torque to engine torque. In addition, the time constant in the engine model was changed to 1.25 s to better match the observed dynamic response of the NAE Bell 205.
The problem with the indicated torque values being too low was corrected by placing a scaling algorithm just upstream of the instrument driver. A third-order polynomial function in airspeed was used. Its coefficients were selected by fitting the scaling function to match NAE flight test values.

6.2 Tail Rotor

The tail rotor on the NAE Bell 205 has been modified by employing a larger one from a Bell 412. This has been accounted for in the ARMCOP simulation by increasing the solidity ratio of the tail rotor from 0.105 to 0.1435.

6.3 Pitch Attitude During Steady Turns

It was determined that the different pitch attitudes noted by the test pilot during steady left and right hand turns were due to the lateral location of the centre of gravity. In ARMCOP it is on the centre line while in the NAE Bell 205 it is displaced laterally. It was decided to leave the ARMCOP value unaltered.

6.4 Flight Initiation Sequence

The flight simulator can be started with the helicopter located at any point within the database of the visual display system. The flight equations are started up in a trimmed flight state and the pilot then flies the simulator away from this state as he wishes. As a result of pilot comments the system has been altered so that the location of the primary flight controls at startup can be determined in one of two ways. In one startup mode the pilot holds the controls where he feels they should be for the trimmed state under study and the simulator accepts these positions as those corresponding to trimmed flight. In the other startup mode the pilot is directed to hold the flight controls in the
positions determined by the ARMCOP trim files. This is accomplished through commands read
from the display console by the simulator operator. In both cases the pilot begins to fly the
simulator with the helicopter and the flight controls in a trimmed state.

6.5 Modifications to Helicopter Dynamics

In response to test pilot comments on the excessive collective/pitch response of the simulator, the
longitudinal response of ARMCOP was examined and compared with NAE Bell 205 flight test data
over the speed range 40-100 kts. The following observations arose from this:

(1) The trimmed pitch attitudes of the ARMCOP simulation were consistently lower than
those of the NAE Bell 205.

(2) The ARMCOP simulation appeared to be less well damped in pitch than the actual
helicopter.

The following changes were carried out on the simulation to address the excessive
collective/pitch coupling in view of the two observations noted above.

(1) The centre of gravity was moved aft by 4 in.

(2) The incidence of the horizontal stabilizer's geometric chord was fixed at 5° leading edge
down (to match the NAE Bell 205).

(3) The dynamic pressure ratio (local/freestream) was reduced from 0.9 to 0.6 at the tail.

(4) The stability derivative $M_w$ was altered by feeding $w$ into the pitching moment
equation. At 60 kts. forward speed the effective $M_w$ was changed from -0.006 to
-0.012.

As a result of flight tests in the simulator studying sideslip problems it was found that the
fuselage sideforce due to sideslip was of the wrong magnitude and polarity. Altering this parameter
(Y1 in ARMCOP) from 11.8 to -118.0 ft²/r cured the problem.
6.6 Summary

Once all of the above changes had been implemented the test pilot judged the simulator to be a good representation of the NAE Bell 205 for modest maneuvering not involving nap-of-the-earth flight and hovering tasks. The configuration employed in these final evaluations and in all subsequent testing consisted of:

1. Bell horizontal stabilizing bar on.
3. Visual display time delay compensation on.
<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value/Units</th>
</tr>
</thead>
<tbody>
<tr>
<td>$G_1$</td>
<td>20</td>
</tr>
<tr>
<td>$G_2$</td>
<td>100</td>
</tr>
<tr>
<td>$G_3$</td>
<td>20</td>
</tr>
<tr>
<td>HP$_{\text{max}}$</td>
<td>1400 HP</td>
</tr>
<tr>
<td>$J$</td>
<td>8,600 slug-ft$^2$</td>
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<tr>
<td>KE</td>
<td>1.75 HP/lb/hr</td>
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<td>WF$_{\text{MAX}}$</td>
<td>800 lb/hr</td>
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<tr>
<td>WF$_{\text{MIN}}$</td>
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<tr>
<td>$\delta_c$</td>
<td>in.</td>
</tr>
<tr>
<td>$\delta_T$</td>
<td>%</td>
</tr>
<tr>
<td>$\varphi_E$</td>
<td>1.25 s</td>
</tr>
<tr>
<td>$\Omega$</td>
<td>r/s</td>
</tr>
<tr>
<td>$\Omega_{\text{IC, SET}}$</td>
<td>33.93 r/s</td>
</tr>
</tbody>
</table>
FIGURE 6.1 ENGINE MODEL
7.0 SIMULATOR ADDITIONS REQUIRED FOR THE FLIGHT TEST PROGRAM

The display and control system developments required to implement the IFR landing approach modes to be evaluated in this study are outlined in Reference 1. Their implementation on the UTIAS Flight Research Simulator is described below. In these tests the longitudinal and lateral stability of the Bell 205 have been augmented in order to represent a modern IFR helicopter. Two flight control modes and a three cue flight director complete the system.

7.1 Stability Augmentation

The stability augmentation involved increasing the pitch and roll damping. This was achieved by incrementing $M_q$ by $-1.71 \, s^{-1}$ and incrementing $L_p$ by $-0.83 \, s^{-1}$.

Following the above changes it was found that the simulator's response to pitch and roll commands was too sluggish. This was corrected by increasing the control power of the cyclic stick by the ratio of the new $M_q(L_p)$ to the old $M_q(L_p)$ based on values corresponding to 60 kts airspeed. This factor was 3.86 for the longitudinal cyclic and 1.60 for the lateral cyclic. The resulting handling qualities were judged to be good.

7.2 Yaw Axis Control Modes

Two different flight control modes were selected for study. Both were intended to allow the pilot to fly landing approaches with their feet on the floor (i.e., no pedal inputs required). In the present tests the yaw pedal inputs were actually set to zero in the simulator.
7.2.1 Turn Coordination Mode

The turn coordination mode was intended to maintain sideslip at an acceptably small value during maneuvering while on the landing approach. It was based on the following reasoning. Consider the lateral equation of motion expressed with airspeed in body frame components in the absence of wind:

\[ \dot{v} = \frac{Y}{m} + g \cos \theta \sin \phi - ru + pw \]  

(7.1)

\[ Y = Y_{\delta r} \cdot \delta r + \Delta Y \]  

(7.2)

(where a positive \( \delta r \) produces a positive yaw rate \( r \)). Here the sideforce \( Y \) has been split into that part due to \( \delta r \) and the rest due to all other sources. Substituting Equation 7.2 into Equation 7.1 produces

\[ \dot{v} = \left[ Y_{\delta r} \cdot \frac{\delta r}{m} + g \cos \theta \sin \phi - ru \right] + \frac{\Delta Y}{m} + pw \]  

(7.3)

Now the feedback scheme of Figure 7.1 is employed to produce a value for \( \delta r \) that approximately drives the term in square brackets in Equation 7.3 to zero. Assuming that \( \frac{\Delta Y}{m} + pw \) is very small, this results in \( \dot{v} = 0 \) and hence if we start the process with \( v = 0 \) and hence sideslip \( \beta = 0 \) it will remain that way as we maneuver and the desired effect will have been achieved.

From Figure 7.1

\[ \delta r = G_{TC} \left[ \frac{E}{lu} \cos \theta \sin \phi - r \right] \]  

(7.4)

Combining Equations 7.3 and 7.4 results in

\[ \dot{v} = \left[ \left( \frac{Y_{\delta r} G_{TC}}{mu} + 1 \right) \left( g \cos \theta \sin \phi - ru \right) \right] + \frac{\Delta Y}{m} + pw \]  

(7.5)
(Note that $Y_\delta$ is negative.) The value of $G_{TC}$ was selected to make

$$\left(\frac{Y_\delta G_{TC}}{mu} + 1\right) = 0 \quad (7.6)$$

for $u = 60$ kts. Thus the desired reduction in $\beta$ to a value near zero will work best for $u = 60$ kts. and will be less effective at other speeds. In addition, when wind is present it is expected that this algorithm will be less effective at minimizing $\beta$.

### 7.2.2 Heading Hold Mode

The heading hold mode was designed to maintain a reference heading to within 5 degrees irrespective of lateral cyclic input. As with the turn coordination mode the pilot is not required to make any yaw pedal inputs while in the heading hold mode. During the experimental trials the yaw pedal inputs were set to zero in the simulator.

The block diagram for the heading hold mode is given in Figure 7.2. This is a UTIAS version that produces performance and handling qualities similar to those of the NAE version reported in Reference 1. It was found that the NAE version would not produce the desired results when implemented on the UTIAS simulator. No reason could be found for this.

### 7.3 Flight Director

A three-axis flight director is used to command pilot inputs on the longitudinal cyclic, the lateral cyclic and the collective controls. Ground speed error is used to command pitch attitude, height error is used to command collective input and lateral position error is used to command bank attitude. The block diagrams for the flight director are given in Figures 7.3 to 7.5. The following descriptions are taken from Reference 1.
7.3.1 Pitch Flight Director

The block diagram of the pitch flight director is given in Figure 7.3. The velocity command is the ground speed profile of Figure 7.6. This represents a decelerating landing approach. The velocity \( u \) is the aircraft ground speed. Measured Bode plots of the transfer function \( FD_\theta/\delta_e \) indicate that this director behaves like a simple integration of the longitudinal cyclic \((k/s)\) over the entire frequency range of pilot control. The pitch limiter in the forward path, set at \( \pm 10 \) degrees around the trim pitch angle of approximately 5 degrees, provides a necessary angular limit to speed corrections when errors become large. Overall scaling of the director was 0.43 inches of symbol movement per 10 kts. of steady state error.

7.3.2 Roll Flight Director

Similar to the pitch flight director, the roll flight director used error \((y-y_c)\) error rate \((\dot{y})\) and aircraft angle \((\phi)\) to create a \(k/s\) transfer function of \(FD_\phi/\delta_a\). The \( Y \) command is the localizer location and \( Y \) error represents lateral deviations from the localizer. Roll angle scaling was set at approximately 1/2 degree per ft. of steady state localizer error. Overall flight director scaling was 0.30 ins. of symbol movement per 100 ft. of steady state localizer error. The limiter in the \( Ky \) path provides a maximum 30 degree heading intercept to the localizer when the localizer error is large. The second limiter provides a \( \pm 30 \) degree boundary on commanded roll angle.

7.3.3 Collective Flight Director

Unlike the pitch and roll flight directors, the collective flight director configuration did not exhibit \( k/s \) behaviour. Initial flight evaluations of the \( k/s \) design showed that this director was exceptionally difficult to centre at moderate to large levels of glideslope error and provided little subjective improvement over a raw glideslope data presentation. While pilots were satisfied with
control of closure rate of the display symbol to zero with pitch and roll axes, pilot technique and comments regarding the collective axis during preliminary evaluations suggested a lower level of attentiveness to collective inputs and led the design of the collective director toward incorporation of substantially more collective lead information. Ground-based simulations in the absence of turbulence and with perfect consistency between all data inputs led to a vertical acceleration feedback path for satisfactory flight director behaviour. This director design could be referred to as a collective position director since the acceleration feedback served to predict the steady state error closure rate for a specific collective input prior to achieving that steady state rate of ascent or descent. Since vertical acceleration is an impractical feedback in the presence of turbulence and aircraft structural noise when high frequency information is required, a washed out collective position signal, with the first order break point roughly corresponding to the aircraft \(Z_w\) derivative, was used in the final design.

The block diagram of the final collective flight director, Figure 7.5, shows the collective position feedback loop and a limiter on the glideslope error term. This limiter bounded the commanded intercept rates of descent to be within 600 ft/min of the steady state trim descent rate, \(\dot{h}_c\). The \(h_c\) command is the glideslope location and \(\dot{h}_c\) for the 6° glideslope is given by

\[
\dot{h}_c = u \tan 6^\circ \tag{7.7}
\]

where \(u\) is the helicopter ground speed. For a constant 60 kts. 6° glideslope approach, nominally requiring a descent rate of 635 ft/min, these limits correspond to aircraft descent rates ranging from 35 ft/min to 1235 ft/min. The Bode plots of the \(F_D \delta_c / \delta_c\) transfer function at 40 and 60 kts. demonstrate a constant amplitude ratio over the normal frequency range of pilot control, consistent with the collective position director concept. Overall scaling of the director was approximately 0.25 ins. symbol deflection per 25 ft. of steady state glideslope error.
7.4 Flight Director Display

The flight director display symbols are depicted in Figure 2.8. The longitudinal and lateral cyclic commands are represented by the displacement of the flight director circle away from the centre of the display. One should fly towards the circle to reduce tracking errors to zero. For example, if the circle is above and to the left of the centre of the display the proper response is to bank to the left and pull up (i.e., left lateral cyclic and aft longitudinal cyclic stick inputs). The stick symbol extending up from the top of the circle (or down from the bottom of the circle) represents the collective command. If the stick is up then reduce (push down on) the collective. The error has been reduced to zero when the stick has shrunk to the edge of the circle symbol.
FIGURE 7.1  TURN COORDINATION SYSTEM

$G_{TC} = 1.92 \text{ in/r/s}$
\[ \Psi_{\text{COM}} + \quad K_\psi \quad + \quad \text{CS}_\text{GAIN} \quad \delta_r \quad \text{BELL-205} \quad \Psi \]

\[ K_\psi = 8.0 \text{ in/r} \]
\[ K_R = 0.04 \text{ in/r/s} \]
\[ \text{CS}_\text{GAIN} = 1.0 \]

**FIGURE 7.2** HEADING HOLD SYSTEM
Figure 7.3 Pitch Flight Director

$K_1 = -1.0$

$K_{FD\theta} = 0.0122$

Units:
- $FD\theta$ — in.
- $u$ — kts.
- $\theta$ — degrees
- $\delta_e$ — in.
Kv limit = \frac{K_y^*}{K_y} u_{GS} \sin 30^\circ

K_y = .0449
K_y' = .55
K_{FD\phi} = .0349

Units
FD\phi — in.
y — ft.
\phi — degrees
\delta_a — in.

FIGURE 7.4 ROLL FLIGHT DIRECTOR
G_1 = .565
G_2 = 1.0
G_3 = 9.6
K_{FDc} = .1344

Units
FDC — in.
h — ft.
δ_c — in.

FIGURE 7.5 COLLECTIVE FLIGHT DIRECTOR
FIGURE 7.6  GROUNDSPEED DECELERATION PROFILE
8.0 EXPERIMENTAL PLAN

The purpose of the test program was to compare the performance and handling qualities ratings achieved during an actual flight test program with those produced using a helicopter flight simulator. The flight test selected was the helicopter IFR approach program described in Reference 1. The simulator employed has been described in the preceding sections of this report.

8.1 IFR Landing Approach Task

The flight evaluations carried out by the NAE and reported in Reference 1 involved capturing and following a 6° linear glideslope while simultaneously decelerating according to the ground speed profile shown in Figure 7.6. An audio alerting signal is included to warn the pilot that he has entered the region where the deceleration profile exists. This audio signal sounds for 3 seconds when the helicopter is 3,550 ft. from the runway threshold.

Two primary types of landing approach techniques were evaluated; the crab technique and the sideslip technique. These techniques are applied in the presence of a crosswind in order to maintain the helicopter on the localizer during the landing approach. In the crab technique the helicopter is flown with zero bank angle and sufficient heading angle (away from the runway heading) to maintain its ground track along the localizer in the presence of the crosswind. In the sideslope technique the helicopter heading is maintained equal to the runway heading. A forward slip is used (with the helicopter banked towards the oncoming crosswind) to generate sufficient airspeed at right angles to the localizer to cancel the effect of the crosswind, thus maintaining the ground track along the localizer. A third technique termed the blending technique was also evaluated in which the landing approach was begun using the crab technique and then changed to the sideslip technique near the end of the maneuver when the ground speed dropped below 40 kts. At the point of transition the heading hold autopilot captures the current helicopter heading and in general this will not be the runway heading. Thus the final portion of the approach is flown with a
combined crab and sideslip. This should prevent excessive values of heading offset and bank angle as the ground speed drops off towards zero.

In these trials the following combinations of approach technique and yaw axis control mode were always employed:

1. Crab technique + turn coordination mode,
2. Sideslip technique + heading hold mode.

8.2 Simulator Flight Test Procedures

At the start of a test flight the pilot is seated in the simulator restrained by a four-point harness. He places his controls to correspond to trimmed level flight at 60 kts. indicated airspeed. The starting location of the helicopter is 500 ft. above ground level (AGL), on the localizer, heading along the runway centreline and 6,800 ft. back from the runway threshold. The helicopter is trimmed for level flight at 60 kts. indicated airspeed. The pilot is informed by the experimenter concerning the current test parameters and the controls are sampled by the simulator program to establish their trimmed positions. At this time the visual display is turned on. When the pilot indicates that he is ready the flight begins in the trimmed state. After 5 seconds the wind fades in linearly over the next 6 seconds (a 15 kt. constant crosswind either 45° from the left or the right; see Figure 8.2). During this portion of the flight the task is to maintain level flight along the localizer and to establish a ground speed of 60 kts. The cyclic commands from the flight director are used to assist in this task. The pilot continues towards the runway at 500 ft. AGL until he captures the glideslope from below at a distance of 4,760 ft. from the runway threshold. He then continues down the glideslope at a ground speed of 60 kts. until he reaches the start of the deceleration curve at 3,550 ft. from the threshold. At this point an audio warning is heard and the decelerating portion of the approach begins. If the flight is being conducted through cloud, the helicopter will break out of the cloud upon descending through 50 ft. AGL and visual contact with the airport will be made. At a height of 40 ft. AGL (or if the ground speed drops below 25 kts.) the run is terminated. This is achieved
by uncoupling the pilot controls and washing out pitch and roll to zero over a 3 second interval. All
other variables are frozen at their final values. The visual display is then greyed out. The details of
the approach are illustrated in Figure 8.1

8.3 Helicopter Configuration During Test Flights

The following configuration selection was present during all of the trials:

(1) Bell horizontal stabilizing bar on.
(2) Adaptive motion-drive algorithm on.
(3) Visual display time delay compensation on.
(4) Lp and Mq augmentation active.
(5) Yaw pedals inactive.
(6) EFIS display.
(7) Crosswind of 15 kts. at ±45° to the runway heading.

8.4 Simulator Independent Variables

Because the purpose of this study was to investigate the influence of flight simulation on
helicopter/pilot performance, several simulator configurations were employed. These were:

<table>
<thead>
<tr>
<th>Condition</th>
<th>Symbol</th>
</tr>
</thead>
<tbody>
<tr>
<td>Motion system on</td>
<td>M</td>
</tr>
<tr>
<td>Motion system off</td>
<td>N</td>
</tr>
<tr>
<td>VFR visibility (clear)</td>
<td>V</td>
</tr>
<tr>
<td>IFR visibility (zero)</td>
<td>I</td>
</tr>
<tr>
<td>Conventional cyclic controls,</td>
<td></td>
</tr>
<tr>
<td>Force stick side arm cyclic controls</td>
<td></td>
</tr>
</tbody>
</table>
8.5 Experimental Sequence

In order to keep the size of the experiment within reasonable bounds it was necessary to restrict the combinations of factors employed. The following 4 flight director/control mode configurations were selected:

- **C1** — (crab technique) + (turn coordination mode) + (3 cue flight director)
- **C2** — (sideslip technique) + (heading hold mode) + (3 cue flight director)
- **C3** — (blending technique) + (blending mode) + (3 cue flight director)
- **C4** — (crab technique) + (turn coordination mode) + (raw display)

In the above, "raw display" refers to a case in which the 3 cue flight director is turned off. The pilot then uses the localizer and glideslope bugs and the commanded ground speed bug (see Figure 2.8).

The visual display options were VFR (clear) (see Figure 2.11) and IFR (zero visibility) with a greyed-out visual display.

The matrix of 20 test cases selected is given in Table 8.1

It was decided that each pilot subject would perform 4 replicates of each of the cases listed in Table 8.1. The order of presentation of the cases was randomized but the 4 replicates were carried out one after the other with a minimum of delay between runs. This was done in order to help the pilot to produce a good handling qualities rating for each case.

Each block of 4 replicates took approximately 12 minutes to run including the time consumed in providing pilot ratings. The production runs for each pilot were run in 4 one-hour blocks. Only 2 one-hour blocks (with a half-hour break between them) were performed on any one day.
8.6 Subjects

Six male subjects took part in the experiment. Two (S1 and S2) were test pilots, three (S3, S4 and S5) were civilian helicopter pilots and one (S6) was a project engineer. Their flight experience is detailed in Table 8.2.

8.7 Experimental Measures

Six objective measures and three pilot evaluations were employed.

The objective measures were based on glideslope and localizer error measured in feet and ground speed error measured in knots. The objective measures were obtained after each run. The first three objective measures were the standard deviations of these variables computed from the time the helicopter passed the nominal glideslope capture point (4,760 ft. from the end of the runway) until the end of the run when it descended through 40 ft. AGL (or the ground speed dropped below 25 kts). The final three objective measures were the values of glideslope, localizer and ground speed error at the end of the run.

The pilot evaluations were those employed in Reference 1: Cooper/Harper handling qualities ratings (see Reference 20 and Figure 8.3), a modified Cooper/Harper scale for workload rating (see Figure 8.4), and a certification related assessment (see Figure 8.5). Pilot evaluations were obtained after each block of four identical runs.

The pilots were asked to refer to Figures 8.3 to 8.5 (provided in the simulator) while making their decisions, and to follow the prescribed decision-making process before giving each response.

When rating the handling qualities, the pilots would refer to Figure 8.3, the Handling Qualities Rating Scale. Beginning from the lower left of this flowchart, the following questions were considered relating to the aircraft characteristics:

(1) Is it controllable?
(2) Is adequate performance attainable with a tolerable pilot workload?

(3) Is it satisfactory without improvement?

The questions would determine if the handling qualities were good, adequate, or deficient. The result from such questions would place the handling qualities in one of four categories, with category 1 representing a configuration which was satisfactory without improvement; category 2, adequate without improvement; category 3, requiring improvement; and category 4, unacceptable. Within each category a further breakdown of the degree of desirability (or deficiency) would be addressed, unless the aircraft were uncontrollable. As a result, a rating from 1 to 10 would be determined, with 1 being the most favourable, and 10 being unacceptable.

While answering the questions relating to the handling qualities, the pilots were prompted by considering the following:

(1) How much compensation was necessary to perform the approach while maintaining the glideslope, localizer and ground speed targets?

(2) Was there any unpredictable or annoying behaviour like Dutch roll oscillations which complicated the task?

(3) How much compensation was required to terminate the run within the acceptable limits of ±25 ft. localizer error, ±25 ft. glideslope error and -1 to +5 kts. ground speed error? Note that the pilots were required to repeatedly meet these three conditions before the actual experiment could be performed.

The Workload Rating from the Modified Cooper/Harper Scale (Figure 8.4) was obtained in a manner similar to the Handling Qualities Rating. By beginning from the lower left of the figure, decisions were made regarding the mental effort required to maintain the required flight profile and to terminate the approach within the specified bounds.

With these in mind, three questions were answered:

(1) Even though errors may be large or frequent, can the instructed task be accomplished most of the time?

(2) Are errors small and inconsequential?
(3) Is mental workload level acceptable?

The response to the third question would determine whether or not the level of mental effort should be reduced by some means, while a negative response to the second question would strongly recommend a major redesign of the system. A negative response to the first question would require a system redesign, indicating that the difficulty level as too great.

The IFR certification assessment was obtained from the test subjects by asking them to refer to Figure 8.5. The pilots were required to begin from the top, and to consider each category. Important points to consider while making the appropriate choice were as follows:

(1) In a high density IFR environment, would you be able to perform the decelerating approach in this configuration and still be able to carry out non-essential tasks (monitoring the engines, communicate with air traffic control or other aircraft, follow a landing check list)?

(2) Would it be advantageous if another crew member performed some or all of these duties?

The subjects would provide a Certification Related Assessment between 1 and 4. Fractions (1.5, 2.5, 3.5) were also permitted.

8.8 Training

The pilots were trained by having them fly the experimental cases listed in Table 8.1. Training included the giving of pilot ratings according to the experimental plan. A pilot was considered to be trained when

(1) he felt trained, and

(2) the experimenter felt he was trained, and

(3) he consistently achieved localizer and glideslope errors of less than ±25 ft. and ground speed errors in the range -1 to +5 kts. at the end of the approach when using the flight director.
(Note that the flight control system is such that perfect tracking of the flight director results in a
ground speed at the end of an approach that is 2 kts. above the nominal profile.)

When a pilot subject begins his training he is given a copy of the Guidelines for Pilot
Evaluations of Landing Approach Trials in the UTIAS Bell 205 Flight Research Simulator (see
Appendix A). After reading the guidelines, the simulator cockpit is demonstrated to the pilot and
any questions clarified. The pilot then flies the sequence of trials listed in Table 8.1 until he is
considered to be fully trained. Typically six 2-hour training sessions were employed.
## Table 8.1
### Test Cases

<table>
<thead>
<tr>
<th>Case</th>
<th>Configuration</th>
<th>Stick</th>
<th>Motion</th>
<th>Visuals</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>C1</td>
<td>Centre</td>
<td>Off</td>
<td>IFR</td>
</tr>
<tr>
<td>2</td>
<td>C1</td>
<td>Centre</td>
<td>On</td>
<td>IFR</td>
</tr>
<tr>
<td>3</td>
<td>C1</td>
<td>Centre</td>
<td>Off</td>
<td>VFR</td>
</tr>
<tr>
<td>4</td>
<td>C1</td>
<td>Centre</td>
<td>On</td>
<td>VFR</td>
</tr>
<tr>
<td>5</td>
<td>C2</td>
<td>Centre</td>
<td>Off</td>
<td>IFR</td>
</tr>
<tr>
<td>6</td>
<td>C2</td>
<td>Centre</td>
<td>On</td>
<td>IFR</td>
</tr>
<tr>
<td>7</td>
<td>C2</td>
<td>Centre</td>
<td>Off</td>
<td>VFR</td>
</tr>
<tr>
<td>8</td>
<td>C2</td>
<td>Centre</td>
<td>On</td>
<td>VFR</td>
</tr>
<tr>
<td>9</td>
<td>C2</td>
<td>Side arm</td>
<td>Off</td>
<td>IFR</td>
</tr>
<tr>
<td>10</td>
<td>C2</td>
<td>Side arm</td>
<td>On</td>
<td>IFR</td>
</tr>
<tr>
<td>11</td>
<td>C2</td>
<td>Side arm</td>
<td>Off</td>
<td>VFR</td>
</tr>
<tr>
<td>12</td>
<td>C2</td>
<td>Side arm</td>
<td>On</td>
<td>VFR</td>
</tr>
<tr>
<td>13</td>
<td>C3</td>
<td>Centre</td>
<td>Off</td>
<td>IFR</td>
</tr>
<tr>
<td>14</td>
<td>C3</td>
<td>Centre</td>
<td>On</td>
<td>IFR</td>
</tr>
<tr>
<td>15</td>
<td>C3</td>
<td>Side arm</td>
<td>Off</td>
<td>IFR</td>
</tr>
<tr>
<td>16</td>
<td>C3</td>
<td>Side arm</td>
<td>On</td>
<td>IFR</td>
</tr>
<tr>
<td>17</td>
<td>C4</td>
<td>Centre</td>
<td>Off</td>
<td>IFR</td>
</tr>
<tr>
<td>18</td>
<td>C4</td>
<td>Centre</td>
<td>On</td>
<td>IFR</td>
</tr>
<tr>
<td>19</td>
<td>C4</td>
<td>Centre</td>
<td>Off</td>
<td>VFR</td>
</tr>
<tr>
<td>20</td>
<td>C4</td>
<td>Centre</td>
<td>On</td>
<td>VFR</td>
</tr>
</tbody>
</table>
Table 8.2
Pilot Experience

<table>
<thead>
<tr>
<th>Pilot No.</th>
<th>Aircraft Hours</th>
<th>Simulator Hours</th>
<th>IFR Rating</th>
<th>Aircraft Hours (Total)</th>
<th>Simulator Hours</th>
<th>Bell 205 Hours</th>
<th>IFR Rating</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>6,500</td>
<td>300</td>
<td>Yes</td>
<td>1,500</td>
<td>75</td>
<td>1,000</td>
<td>Yes</td>
</tr>
<tr>
<td>2</td>
<td>8,200</td>
<td>250</td>
<td>Yes</td>
<td>1,500</td>
<td>60</td>
<td>1,000</td>
<td>Yes</td>
</tr>
<tr>
<td>3</td>
<td>500</td>
<td>20</td>
<td>No</td>
<td>550</td>
<td>0</td>
<td>0</td>
<td>No</td>
</tr>
<tr>
<td>4</td>
<td>200</td>
<td>10</td>
<td>No</td>
<td>1,200</td>
<td>21</td>
<td>0</td>
<td>No</td>
</tr>
<tr>
<td>5</td>
<td>20</td>
<td>7</td>
<td>No</td>
<td>74</td>
<td>0</td>
<td>0</td>
<td>No</td>
</tr>
<tr>
<td>6</td>
<td>50</td>
<td>30</td>
<td>No</td>
<td>0</td>
<td>50</td>
<td>0</td>
<td>No</td>
</tr>
</tbody>
</table>
FIGURE 8.1 LANDING APPROACH DETAILS
**FIGURE 8.3 HANDLING QUALITIES RATING SCALE**
**FIGURE 8.4 MODIFIED COOPER/HARPER SCALE (WORKLOAD RATING)**

*(FROM REFERENCE 1)*
BASED ON YOUR SHORT EVALUATION, IN WHICH OF THE FOLLOWING CATEGORIES WOULD YOU PLACE THIS CONFIGURATION:

1. The helicopter has good flying qualities and could be operated safely in a high-density IFR environment by one pilot without the assistance of additional crew members.

2. The helicopter has marginal flying qualities for operations in a high-density IFR environment by one pilot without the assistance of additional crew members.

3. The helicopter has flying qualities deficiencies which make it unsuitable for single-pilot operations in a high-density IFR environment, however it could be operated safely within such an environment if the pilot-in-command were relieved of all non-control tasks by an additional qualified crew member.

4. The helicopter has major flying qualities deficiencies which make it unsuitable for operation within a high-density IFR environment.

FIGURE 8.5 CERTIFICATION RELATED ASSESSMENT (FROM REFERENCE 1)
9.0 RESULTS AND DISCUSSION

In all, 480 landing approaches were flown. This produced 480 sets of objective measurements and 120 sets of pilot evaluations. In this section samples of raw data are presented along with processed summaries. The statistical analysis employed is the analysis of variance (ANOVA). The purpose of this presentation is to compare the present simulator findings with those from the flight tests of Reference 1.

9.1 Objective Measures

Sample traces of glideslope error (high-positive) localizer error (right-positive) and ground speed error (fast-positive) along the 6° landing approach path for the present experiment (C3, blending technique, centre stick, I/M) are given in Figures 9.1, 9.3 and 9.5. One sample trace for each of the 6 pilots is included. Figures 9.2, 9.4 and 9.6 show data from Reference 1 for the same case. In the blending technique the approach is flown using the crab technique/turn coordination mode down to a ground speed of 40 kts. (an altitude of approximately 170 ft.) at which point a switch is made to the sideslip technique/heading hold mode for the remainder of the approach. From Figures 9.1 and 9.2 it can be seen that glideslope tracking in the simulator was similar to that in flight with perhaps better average performance. From Figures 9.3 and 9.4 it would appear that similar localizer tracking performance was achieved in both cases. Figures 9.5 and 9.6 indicate that the simulator tracking of ground speed commands was more consistent than that achieved in flight, although the magnitudes of the errors were comparable.

The means and standard deviations of the objective measurements are presented in Figures 9.7 to 9.42. Each data point is based on 4 replicates by all 6 subjects. The plots contain data for IFR visibility with simulator motion on (I/M), IFR visibility with simulator motion off (I/N), VFR visibility with simulator motion on (V/M) and VFR visibility with simulator motion off (V/N).
Several ANOVA's were performed on subsets of the objective measurement data and these are presented in Tables 9.1 to 9.18.

The data set employed in Tables 9.1 to 9.6 was generated from 4 replications by the 6 subjects of tasks incorporating 4 treatments (the simulator conditions I/M, I/N, V/M and V/N described above) and 3 cases (flight director/control modes C1, C2 and C4 described in Section 8.5). The centre stick was used in all tasks.

The data set employed in Tables 9.7 to 9.12 was generated from 4 replications by the 6 subjects of tasks employing 4 treatments (the simulator conditions I/M, I/N, V/M and V/N) and 2 cases (centre stick and side arm controller). The flight director/control mode employed was C2.

The data set employed in Tables 9.13 to 9.18 was generated from 4 replications by the 6 subjects of tasks involving 2 treatments (the simulator conditions I/M and I/N) and 2 cases (flight director/control modes C2 and C3). The centre stick was used in all tasks.

In assessing the results from the ANOVA tests the following assumptions about the significance of the results have been made. (Here \( P(x > F) \) represents the probability that an \( F \) value as large as that found could have occurred by chance given that the factor under consideration has no effect on the measured performance.)

\[
\begin{array}{ll}
P(x > F) & \text{Significance} \\
0 - 0.01 & \text{Very significant trend} \\
0.01 - 0.05 & \text{Significant trend} \\
0.05 - 0.10 & \text{Mildly significant trend}
\end{array}
\]

Consider the ANOVA results in Tables 9.13 to 9.18. This analysis was carried out to check on the performance differences associated with C2 — (sideslip with heading hold) and C3 — (blending mode). This can be determined by checking the \( P(x > F) \) values for cases. It is seen that the effect of cases is significant to very significant for the standard deviations along the approach (although the differences are small) but not significant to mildly significant for the final values at the

Next consider the ANOVA results in Tables 9.7 to 9.12. This analysis was carried out to check on the performance differences associated with the centre stick and the side arm controller. This can be determined by checking the $P(x > F)$ values for cases. It is seen that the effect of cases is very significant except for the final localizer error. The performance with the side arm controller was generally slightly poorer than that with the centre stick. This can be seen by comparing the results shown in Figures 9.8 and 9.10, Figures 9.14 and 9.16, Figures 9.20 and 9.22, Figures 9.26 and 9.28, Figures 9.32 and 9.34, and Figures 9.38 and 9.40.

Finally, consider the ANOVA results in Tables 9.1 to 9.6. This analysis was carried out to check on the performance differences associated with subjects, simulator conditions and flight director/control modes (C1, C2 and C4). In the tables "treatments" refers to simulator conditions (I/M, I/N, V/M, and V/N) and "cases" refers to the flight director/control modes. The corresponding performance data are plotted in Figures 9.7 to 9.9, 9.13 to 9.15, 9.19 to 9.21, 9.25 to 9.27, 9.31 to 9.33, and 9.37 to 9.39. Along the approach it is seen that subject and flight director/control mode effects are all very significant while simulator conditions are only significant (very) for localizer tracking. When final values of the objective measures are examined it is found that subject effects are only significant (very) for ground speed error, simulator conditions are only significant (very) for glideslope error and flight director/control mode effects are very significant for ground speed error and mildly significant for glideslope error.

Overall it appears that the subjects' performance was quite individualistic except when forced to conform to an acceptable window at the end of the landing approach which constrained them all to generate similar small final spatial tracking errors. There appears to be less concern about final ground speed errors.

The effect of simulator condition on performance appears to be modest overall. Significant effects do not occur in any pattern. From Figures 9.13, 9.14 and 9.15 it appears that the simulator
conditions with the VFR display improved pilot performance (for the standard deviation of localizer error) on the raw data display (C4). From Figures 9.25, 9.26 and 9.27 it appears that the VFR display tended to degrade performance (for final glideslope error). The influence of allowing an outside scene to be present during the evaluation of an IFR task in the simulator is thus seen to be variable. Its impact spans the spectrum from helpful through neutral to distracting. In Figure 9.27 it can be seen that for the C4 flight director/control mode (which produces the worst performance) the presence of simulator motion improves performance (for the final glideslope error) when the VFR display is on. In general, however, simulator motion has little effect on performance in the present study.

The influence of the flight director/control mode on performance is even more significant than that due to subject effects. In all instances it is clear that while C1 and C2 produce similar performance, C4 has by far the worst. This results from the fact that C4 has no flight director symbol to guide the pilot. This forces the pilot to employ raw glideslope, localizer and ground speed data (see Section 2.2.1). In the present IFR task this produces the observed performance decrement. Although the pilots are able to retain control of the helicopter, precision tracking is not possible in the present task.

Overall, the above findings relating to flight director/control modes are in general agreement with those of Reference 1.

### 9.2 Pilot Evaluations

The pilot evaluations (see Figures 8.3 to 8.5) produce results in the form of a single number for each application. There is some uncertainty about the validity of performing statistical operations on these numbers (such as averaging, ANOVA, etc.). For this reason all of the raw data are presented in this report and no ANOVA has been carried out on them. The handling qualities ratings are presented in Figures 9.43 and 9.44. The workload ratings are presented in Figures 9.45
and 9.46. The IFR certification assessments are presented in Figures 9.47 and 9.48. Pilot comments obtained at the same time as the evaluations are collected in Appendix B.

Consider the handling qualities results of Figures 9.43 and 9.44. Three horizontal lines have been drawn across those cases employing the centre stick. They represent the mean and extreme values reported in Reference 1 based on 5 evaluation pilots. The first thing to note is the reasonable agreement between the present experiment and Reference 1 in the cases of flight director/control modes C1 and C4. Overall, the simulator ratings are about one point lower (better) than those achieved in flight. There could be many reasons for this but it is suspected that two factors dominate:

1. In flight, uncontrolled atmospheric effects due to turbulence and wind shear would tend to degrade handling qualities ratings and increase workload.

2. In flight the signals used to generate the flight director display were obtained from physical sensors having the usual imperfections while the simulator employed clean and exact signals. This would tend to make the simulator easier to fly.

In the case of C2 and C3 it appears that the current tests produced significantly better (lower) handling qualities ratings. Since both these configurations employ the heading hold mode it is suspected that the present heading hold implementation is easier to fly than the one that the NAE employed during their flight tests. As mentioned in Section 7.2.2 the current version of the heading hold control system did not duplicate that of Reference 1. This can be seen by comparing Figure 7.2 with the Reference 1 configuration shown in Figure 9.50. We were unable to make the version shown in Figure 9.50 work on the present simulator. From the handling qualities results it is apparent that the version of Figure 7.2 produces handling qualities ratings that are approximately 2 points better than that of Reference 1. This probably represents a shortcoming in the simulation of this control mode.
Of the 6 subjects employed in the present study only S1 and S2 were experienced in giving handling qualities ratings. From the data of Figures 9.43 and 9.44 it appears that the ratings provided by the other 4 subjects were only slightly more variable for cases involving the centre stick.

There is no clear trend in the handling qualities ratings as the simulator conditions are changed. This is consistent with the performance results reported in Section 9.1.

In general the side arm controller increased the handling qualities ratings by 1 or 2 points. This could be expected, given that the stick was rigid. The use of a compliant side arm controller would probably improve these ratings.

The workload ratings of Figures 9.45 and 9.46 follow the same trends noted above for the handling qualities ratings.

The raw IFR certification assessment data of Figures 9.47 and 9.48 indicate that the non-IFR rated pilots from time to time had difficulty in providing consistent ratings. Overall, however, for cases employing the centre stick, the data were fairly uniform. Figure 9.49 superimposes the present results on a figure from Reference 1 giving the percentage of pilot ratings in each of 3 categories. The following transformation between rating and certification level was used:

<table>
<thead>
<tr>
<th>Rating</th>
<th>Certification Level</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>1-pilot certifiable</td>
</tr>
<tr>
<td>2,3</td>
<td>2-pilot certifiable</td>
</tr>
<tr>
<td>4</td>
<td>uncertifiable</td>
</tr>
</tbody>
</table>

It can be seen that for the case presented (I/M matches most closely the flight trials) that for flight director/control modes C1 and C4 there is perfect agreement between the present experiment and Reference 1. C2 and C3, as expected, produced widely different results with the present simulation indicating single pilot certifiability. This is a direct result of the corresponding handling qualities and workload levels.
If one were to only look at the results produced by the IFR rated pilots (S1 and S2) they rated all centre stick cases, except the raw display (C4), as being single pilot IFR certifiable. Based on Figure 9.49 (and excluding C2 and C3 as having been implemented imperfectly in the simulator) these results for C1 are not incompatible with those of Reference 1. In the case of the raw data display C4, the IFR pilot results of Figure 9.47 rated the helicopter uncertifiable for 50% of the cases and 2-pilot certifiable for 50% of the cases. This is 17 percentage points more favorable in rating than the results of Reference 1 shown in Figure 9.49. This could have been caused by the same factors that produced slightly better handling qualities ratings as noted above.
Table 9.1

ANOVA FOR THE STANDARD DEVIATION OF GLIDESLOPE ERROR WITH CENTRE STICK

(TREATMENTS → SIMULATOR CONDITIONS; CASES → FLIGHT DIRECTOR/CONTROL MODES)

<table>
<thead>
<tr>
<th>Effect</th>
<th>Degrees of Freedom</th>
<th>Sum of Squares</th>
<th>Mean Square</th>
<th>F Value</th>
<th>P (\chi &gt; F)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Subjects</td>
<td>5</td>
<td>472.6462</td>
<td>94.5293</td>
<td>4.7824</td>
<td>0.0004</td>
</tr>
<tr>
<td>Treatments</td>
<td>3</td>
<td>86.3064</td>
<td>28.7688</td>
<td>1.4555</td>
<td>0.2277</td>
</tr>
<tr>
<td>Sub. X Trim</td>
<td>15</td>
<td>463.6841</td>
<td>30.9123</td>
<td>1.5639</td>
<td>0.0856</td>
</tr>
<tr>
<td>Cases</td>
<td>2</td>
<td>5427.7890</td>
<td>2713.8950</td>
<td>137.3015</td>
<td>&lt;0.0001</td>
</tr>
<tr>
<td>Subj X Cas</td>
<td>10</td>
<td>994.0002</td>
<td>99.4000</td>
<td>5.0289</td>
<td>&lt;0.0001</td>
</tr>
<tr>
<td>TrtmtsxCas</td>
<td>6</td>
<td>323.7649</td>
<td>53.9608</td>
<td>2.7300</td>
<td>0.0141</td>
</tr>
<tr>
<td>S X T X C</td>
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<td>664.8752</td>
<td>22.1625</td>
<td>1.1212</td>
<td>0.3122</td>
</tr>
<tr>
<td>Residual</td>
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<td>4269.4450</td>
<td>19.7659</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Total</td>
<td>287</td>
<td>12702.5100</td>
<td></td>
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<td></td>
</tr>
</tbody>
</table>
TABLE 9.2
ANOVA FOR THE STANDARD DEVIATION OF LOCALIZER ERROR WITH CENTRE STICK

(TREATMENTS → SIMULATOR CONDITIONS; CASES → FLIGHT DIRECTOR/CONTROL MODES)

<table>
<thead>
<tr>
<th>Effect</th>
<th>Degrees of Freedom</th>
<th>Sum of Squares</th>
<th>Mean Square</th>
<th>F Value</th>
<th>P(x &gt; F)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Subjects</td>
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<td>5604.0120</td>
<td>1120.8020</td>
<td>4.1211</td>
<td>0.0013</td>
</tr>
<tr>
<td>Treatments</td>
<td>3</td>
<td>3873.3530</td>
<td>1291.1180</td>
<td>4.7473</td>
<td>0.0032</td>
</tr>
<tr>
<td>Sub. X Trtm</td>
<td>15</td>
<td>5523.6210</td>
<td>368.2415</td>
<td>1.3540</td>
<td>0.1724</td>
</tr>
<tr>
<td>Cases</td>
<td>2</td>
<td>41554.8800</td>
<td>20777.4400</td>
<td>76.3972</td>
<td>&lt;0.0001</td>
</tr>
<tr>
<td>Subj X Cas</td>
<td>10</td>
<td>7624.1640</td>
<td>762.4165</td>
<td>2.8034</td>
<td>0.0028</td>
</tr>
<tr>
<td>TrtmtsXCas</td>
<td>6</td>
<td>8720.0780</td>
<td>1453.3460</td>
<td>5.3439</td>
<td>&lt;0.0001</td>
</tr>
<tr>
<td>S X T X C</td>
<td>30</td>
<td>10369.4100</td>
<td>345.6470</td>
<td>1.2709</td>
<td>0.1677</td>
</tr>
<tr>
<td>Residual</td>
<td>216</td>
<td>58744.6600</td>
<td>271.9661</td>
<td></td>
<td></td>
</tr>
<tr>
<td><strong>Total</strong></td>
<td><strong>287.</strong></td>
<td><strong>142014.2000</strong></td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
### TABLE 9.3

ANOVA FOR THE STANDARD DEVIATION OF GROUNDSPEED ERROR WITH CENTRE STICK

(TREATMENTS → SIMULATOR CONDITIONS; CASES → FLIGHT DIRECTOR/CONTROL MODES)

<table>
<thead>
<tr>
<th>Effect</th>
<th>Degrees of Freedom</th>
<th>Sum of Squares</th>
<th>Mean Square</th>
<th>F Value</th>
<th>P(x &gt; F)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Subjects</td>
<td>5</td>
<td>36.6713</td>
<td>7.3343</td>
<td>8.9171</td>
<td>&lt;0.0001</td>
</tr>
<tr>
<td>Treatments</td>
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<td>4.2030</td>
<td>1.4010</td>
<td>1.7034</td>
<td>0.1673</td>
</tr>
<tr>
<td>Sub X Trtm</td>
<td>15</td>
<td>22.8226</td>
<td>1.5215</td>
<td>1.8499</td>
<td>0.0298</td>
</tr>
<tr>
<td>Cases</td>
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<td>319.4355</td>
<td>159.7178</td>
<td>194.1866</td>
<td>&lt;0.0001</td>
</tr>
<tr>
<td>Subj X Cas</td>
<td>10</td>
<td>78.2767</td>
<td>7.8277</td>
<td>9.5170</td>
<td>&lt;0.0001</td>
</tr>
<tr>
<td>TrtmtsxCas</td>
<td>6</td>
<td>8.1745</td>
<td>1.3624</td>
<td>1.6564</td>
<td>0.1331</td>
</tr>
<tr>
<td>S X T X C</td>
<td>30</td>
<td>40.3690</td>
<td>1.3456</td>
<td>1.6360</td>
<td>0.0248</td>
</tr>
<tr>
<td>Residual</td>
<td>216</td>
<td>177.6592</td>
<td>0.8225</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

Total: 287. 687.6118
TABLE 9.4
ANOVA FOR THE FINAL GLIDESLOPE ERROR WITH CENTRE STICK
(TREATMENTS → SIMULATOR CONDITIONS;
CASES → FLIGHT DIRECTOR/CONTROL MODES)

<table>
<thead>
<tr>
<th>Effect</th>
<th>Degrees of Freedom</th>
<th>Sum of Squares</th>
<th>Mean Square</th>
<th>F Value</th>
<th>P (x &gt; F)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Subjects</td>
<td>5</td>
<td>715.3564</td>
<td>143.0713</td>
<td>1.1556</td>
<td>0.3322</td>
</tr>
<tr>
<td>Treatments</td>
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<td>1480.4470</td>
<td>493.4822</td>
<td>3.9861</td>
<td>0.0086</td>
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TABLE 9.5
ANOVA FOR THE FINAL LOCALIZER ERROR WITH CENTRE STICK
(TREATMENTS → SIMULATOR CONDITIONS;
CASES → FLIGHT DIRECTOR/CONTROL MODES)

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<th>P(x &gt; F)</th>
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TABLE 9.6
ANOVA FOR THE FINAL GROUND SPEED ERROR WITH CENTRE STICK
(TREATMENTS → SIMULATOR CONDITIONS;
CASES → FLIGHT DIRECTOR/CONTROL MODES)

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### TABLE 9.7
ANOVA FOR THE STANDARD DEVIATION OF GLIDESLOPE ERROR WITH C2
(TREATMENTS → SIMULATOR CONDITIONS; CASES → STICK)

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TABLE 9.8
ANOVA FOR THE STANDARD Deviation OF LOCALIZER ERROR WITH C2
(TREATMENTS → SIMULATOR CONDITIONS; CASES → STICK)

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TABLE 9.9
ANOVA FOR THE STANDARD DEVIATION OF GROUND SPEED ERROR WITH C2
(TREATMENTS → SIMULATOR CONDITIONS; CASES → STICK)

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<th>P(x &gt; F)</th>
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TABLE 9.10
ANOVA FOR THE FINAL GLIDESLOPE ERROR WITH C2
(TREATMENTS → SIMULATOR CONDITIONS; CASES → STICK)

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### TABLE 9.11
ANOVA FOR THE FINAL LOCALIZER ERROR WITH C2
(TREATMENTS $\rightarrow$ SIMULATOR CONDITIONS; CASES $\rightarrow$ STICK)

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<td>F Value</td>
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TABLE 9.13

ANOVA FOR THE STANDARD DEVIATION OF GLIDESLOPE ERROR WITH CENTRE STICK

(TREATMENTS → SIMULATOR CONDITIONS;
CASES → FLIGHT DIRECTOR/CONTROL MODES)

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<th>Mean Square</th>
<th>F Value</th>
<th>P(x &gt; F)</th>
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TABLE 9.14
ANOVA FOR THE STANDARD DEVIATION OF LOCALIZER ERROR WITH CENTRE STICK
(TREATMENTS → SIMULATOR CONDITIONS; CASES → FLIGHT DIRECTOR/CONTROL MODES)

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<th>Degrees of Freedom</th>
<th>Sum of Squares</th>
<th>Mean Square</th>
<th>F Value</th>
<th>P(x &gt; F)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Subjects</td>
<td>5</td>
<td>216.4962</td>
<td>43.2992</td>
<td>19.5942</td>
<td>&lt;0.0001</td>
</tr>
<tr>
<td>Treatments</td>
<td>1</td>
<td>21.1291</td>
<td>21.1291</td>
<td>9.5615</td>
<td>0.0028</td>
</tr>
<tr>
<td>Sub. X Trtm</td>
<td>5</td>
<td>49.4034</td>
<td>9.8807</td>
<td>4.4713</td>
<td>0.0013</td>
</tr>
<tr>
<td>Cases</td>
<td>1</td>
<td>73.6656</td>
<td>73.6656</td>
<td>33.3358</td>
<td>&lt;0.0001</td>
</tr>
<tr>
<td>Subj X Cas</td>
<td>5</td>
<td>3.4466</td>
<td>0.6893</td>
<td>0.3119</td>
<td>0.9043</td>
</tr>
<tr>
<td>Trtm X Cas</td>
<td>1</td>
<td>13.9576</td>
<td>13.9576</td>
<td>6.3162</td>
<td>0.0142</td>
</tr>
<tr>
<td>S X T X C</td>
<td>5</td>
<td>9.8661</td>
<td>1.9732</td>
<td>0.8929</td>
<td>0.4905</td>
</tr>
<tr>
<td>Residual</td>
<td>72</td>
<td>159.1058</td>
<td>2.2098</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Total</td>
<td>95</td>
<td>547.0703</td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
# Table 9.15

ANOVA for the Standard Deviation of Ground Speed Error with Centre Stick

(TREATMENTS → SIMULATOR CONDITIONS; CASES → FLIGHT DIRECTOR/CONTROL MODES)

<table>
<thead>
<tr>
<th>Effect</th>
<th>Degrees of Freedom</th>
<th>Sum of Squares</th>
<th>Mean Square</th>
<th>F Value</th>
<th>P(x &gt; F)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Subjects</td>
<td>5</td>
<td>0.3472</td>
<td>0.0694</td>
<td>7.3805</td>
<td>&lt;0.0001</td>
</tr>
<tr>
<td>Treatments</td>
<td>1</td>
<td>0.0335</td>
<td>0.0335</td>
<td>3.5646</td>
<td>0.0631</td>
</tr>
<tr>
<td>Sub x Trtm</td>
<td>5</td>
<td>0.2092</td>
<td>0.0418</td>
<td>4.4469</td>
<td>0.0014</td>
</tr>
<tr>
<td>Cases</td>
<td>1</td>
<td>0.0323</td>
<td>0.0323</td>
<td>3.4338</td>
<td>0.0680</td>
</tr>
<tr>
<td>Sub x Cas</td>
<td>5</td>
<td>0.0945</td>
<td>0.0189</td>
<td>2.0095</td>
<td>0.0875</td>
</tr>
<tr>
<td>Trtm x Cas</td>
<td>1</td>
<td>0.0393</td>
<td>0.0393</td>
<td>4.1821</td>
<td>0.0445</td>
</tr>
<tr>
<td>S x T x C</td>
<td>5</td>
<td>0.0587</td>
<td>0.0117</td>
<td>1.2480</td>
<td>0.2959</td>
</tr>
<tr>
<td>Residual</td>
<td>72</td>
<td>0.6773</td>
<td>0.0094</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

Total 95 1.4921
### TABLE 9.16
ANOVA FOR THE FINAL GLIDESLOPE ERROR WITH CENTRE STICK

(TREATMENTS → SIMULATOR CONDITIONS; CASES → FLIGHT DIRECTOR/CONTROL MODES)

<table>
<thead>
<tr>
<th>Effect</th>
<th>Degrees of Freedom</th>
<th>Sum of Squares</th>
<th>Mean Square</th>
<th>F Value</th>
<th>P(x &gt; F)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Subjects</td>
<td>5.</td>
<td>798.6370</td>
<td>159.7274</td>
<td>47.6591</td>
<td>&lt;0.0001</td>
</tr>
<tr>
<td>Treatments</td>
<td>1.</td>
<td>0.0662</td>
<td>0.0662</td>
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<td>0.8886</td>
</tr>
<tr>
<td>Sub. X Trtms</td>
<td>5.</td>
<td>18.7737</td>
<td>3.7547</td>
<td>1.1203</td>
<td>0.3574</td>
</tr>
<tr>
<td>Cases</td>
<td>1.</td>
<td>3.0687</td>
<td>3.0687</td>
<td>0.9156</td>
<td>0.3418</td>
</tr>
<tr>
<td>Subj X Cas</td>
<td>5.</td>
<td>24.0208</td>
<td>4.8042</td>
<td>1.4335</td>
<td>0.2227</td>
</tr>
<tr>
<td>TrtmtsXCas</td>
<td>1.</td>
<td>2.6060</td>
<td>2.6060</td>
<td>0.7776</td>
<td>0.3808</td>
</tr>
<tr>
<td>S X T X C</td>
<td>5.</td>
<td>87.9225</td>
<td>17.5845</td>
<td>5.2468</td>
<td>0.0004</td>
</tr>
<tr>
<td>Residual</td>
<td>72.</td>
<td>241.3050</td>
<td>3.3515</td>
<td></td>
<td></td>
</tr>
<tr>
<td><strong>Total</strong></td>
<td>95.</td>
<td><strong>1176.4000</strong></td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
TABLE 9.17

ANOVA FOR THE FINAL LOCALIZER ERROR WITH CENTRE STICK

(TREATMENTS \(\rightarrow\) SIMULATOR CONDITIONS;
CASES \(\rightarrow\) FLIGHT DIRECTOR/CONTROL MODES)

<table>
<thead>
<tr>
<th>Effect</th>
<th>Degrees of Freedom</th>
<th>Sum of Squares</th>
<th>Mean Square</th>
<th>F Value</th>
<th>P(x &gt; F)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Subjects</td>
<td>5.</td>
<td>504.0903</td>
<td>100.8181</td>
<td>4.1913</td>
<td>0.0021</td>
</tr>
<tr>
<td>Treatments</td>
<td>1.</td>
<td>31.5817</td>
<td>31.5817</td>
<td>1.3129</td>
<td>0.2557</td>
</tr>
<tr>
<td>Sub. X Trtm</td>
<td>5.</td>
<td>237.2473</td>
<td>47.4495</td>
<td>1.9726</td>
<td>0.0930</td>
</tr>
<tr>
<td>Cases</td>
<td>1.</td>
<td>79.3216</td>
<td>79.3216</td>
<td>3.2976</td>
<td>0.0735</td>
</tr>
<tr>
<td>Subj X Cas</td>
<td>5.</td>
<td>33.5352</td>
<td>6.7070</td>
<td>0.2788</td>
<td>0.9233</td>
</tr>
<tr>
<td>TrtmtsXCas</td>
<td>1.</td>
<td>11.7597</td>
<td>11.7597</td>
<td>0.4889</td>
<td>0.4867</td>
</tr>
<tr>
<td>S X T X C</td>
<td>5.</td>
<td>207.5229</td>
<td>41.5046</td>
<td>1.7255</td>
<td>0.1397</td>
</tr>
<tr>
<td>Residual</td>
<td>72.</td>
<td>1731.9010</td>
<td>24.0542</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Total</td>
<td>95.</td>
<td>2836.9590</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Effect</td>
<td>Degrees of Freedom</td>
<td>Sum of Squares</td>
<td>Mean Square</td>
<td>F Value</td>
<td>P(x &gt; F)</td>
</tr>
<tr>
<td>------------------</td>
<td>--------------------</td>
<td>----------------</td>
<td>-------------</td>
<td>---------</td>
<td>----------</td>
</tr>
<tr>
<td>Subjects</td>
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<td>10.7446</td>
<td>2.1489</td>
<td>23.2459</td>
<td>&lt;0.0001</td>
</tr>
<tr>
<td>Treatments</td>
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<td>1.8612</td>
<td>1.8612</td>
<td>20.1335</td>
<td>&lt;0.0001</td>
</tr>
<tr>
<td>Sub. X Trtm</td>
<td>5</td>
<td>2.3166</td>
<td>0.4633</td>
<td>5.0118</td>
<td>0.0005</td>
</tr>
<tr>
<td>Cases</td>
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<td>0.0087</td>
<td>0.0087</td>
<td>0.0946</td>
<td>0.7593</td>
</tr>
<tr>
<td>Subj X Cas</td>
<td>5</td>
<td>0.1960</td>
<td>0.0392</td>
<td>0.4240</td>
<td>0.8306</td>
</tr>
<tr>
<td>TrtmtsXCas</td>
<td>1</td>
<td>0.0176</td>
<td>0.0176</td>
<td>0.1909</td>
<td>0.6635</td>
</tr>
<tr>
<td>S X T X C</td>
<td>5</td>
<td>0.1886</td>
<td>0.0377</td>
<td>0.4081</td>
<td>0.8417</td>
</tr>
<tr>
<td>Residual</td>
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<td>6.6559</td>
<td>0.0924</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Total</td>
<td>95</td>
<td>21.9893</td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
FIGURE 9.1  C3 GLIDESLOPE ERROR VS. ALTITUDE
(PRESENT STUDY)

FIGURE 9.2  C3 GLIDESLOPE ERROR VS. ALTITUDE
(REFERENCE 1)
FIGURE 9.3  C3 LOCALIZER ERROR VS. ALTITUDE
(PRESENT STUDY)

FIGURE 9.4  C3 LOCALIZER ERROR VS. ALTITUDE
(REFERENCE 1)
FIGURE 9.5  C3 GROUND SPEED ERROR VS. ALTITUDE  
(PRESENT STUDY)

FIGURE 9.6  C3 GROUND SPEED ERROR VS. ALTITUDE  
(REFERENCE 1)
FIGURE 9.7 STANDARD DEVIATION OF GLIDESLOPE ERROR (C1 WITH CENTRE STICK)

FIGURE 9.8 STANDARD DEVIATION OF GLIDESLOPE ERROR (C2 WITH CENTRE STICK)
FIGURE 9.9 STANDARD DEVIATION OF GLIDESLOPE ERROR (C4 WITH CENTRE STICK)

FIGURE 9.10 STANDARD DEVIATION OF GLIDESLOPE ERROR (C2 WITH SIDE ARM CONTROLLER)
FIGURE 9.11 STANDARD DEVIATION OF GLIDESLOPE ERROR (C3 WITH SIDE ARM CONTROLLER)

FIGURE 9.12 STANDARD DEVIATION OF GLIDESLOPE ERROR (C3 WITH CENTRE STICK)
FIGURE 9.13 STANDARD DEVIATION OF LOCALIZER ERROR (C1 WITH CENTRE STICK)

![Bar chart showing standard deviation of localizer error for I/M, I/N, V/M, and V/N with center stick.]

FIGURE 9.14 STANDARD DEVIATION OF LOCALIZER ERROR (C2 WITH CENTRE STICK)

![Bar chart showing standard deviation of localizer error for I/M, I/N, V/M, and V/N with center stick.]

- S.D. LOC ERROR
- FEET
- I/M I/N V/M V/N
- μ±10°
FIGURE 9.15 STANDARD DEVIATION OF LOCALIZER ERROR (C4 WITH CENTRE STICK)

FIGURE 9.16 STANDARD DEVIATION OF LOCALIZER ERROR (C2 WITH SIDE ARM CONTROLLER)
FIGURE 9.17 STANDARD DEVIATION OF LOCALIZER ERROR (C3 WITH SIDE ARM CONTROLLER)

FIGURE 9.18 STANDARD DEVIATION OF LOCALIZER ERROR (C3 WITH CENTRE STICK)
FIGURE 9.19 STANDARD DEVIATION OF GROUND SPEED ERROR (C1 WITH CENTRE STICK)

FIGURE 9.20 STANDARD DEVIATION OF GROUND SPEED ERROR (C2 WITH CENTRE STICK)
FIGURE 9.21 STANDARD DEVIATION OF GROUND SPEED ERROR (C4 WITH CENTRE STICK)

FIGURE 9.22 STANDARD DEVIATION OF GROUND SPEED ERROR (C2 WITH SIDE ARM CONTROLLER)
FIGURE 9.23  STANDARD DEVIATION OF GROUND SPEED ERROR
(C3 WITH SIDE ARM CONTROLLER)

FIGURE 9.24  STANDARD DEVIATION OF GROUND SPEED ERROR
(C3 WITH CENTRE STICK)
FIGURE 9.25 FINAL GLIDESLOPE ERROR
(C1 WITH CENTRE STICK)

FIGURE 9.26 FINAL GLIDESLOPE ERROR
(C2 WITH CENTRE STICK)
FIGURE 9.27 FINAL GLIDESLOPE ERROR (C4 WITH CENTRE STICK)

FIGURE 9.28 FINAL GLIDESLOPE ERROR (C2 WITH SIDE ARM CONTROLLER)
FIGURE 9.29 FINAL GLIDESLOPE ERROR (C3 WITH SIDE ARM CONTROLLER)

FIGURE 9.30 FINAL GLIDESLOPE ERROR (C3 WITH CENTRE STICK)
FIGURE 9.31 FINAL LOCALIZER ERROR (C1 WITH CENTRE STICK)

FIGURE 9.32 FINAL LOCALIZER ERROR (C2 WITH CENTRE STICK)
FIGURE 9.33 FINAL LOCALIZER ERROR
(C4 WITH CENTRE STICK)

FIGURE 9.34 FINAL LOCALIZER ERROR
(C2 WITH SIDE ARM CONTROLLER)
FIGURE 9.35 FINAL LOCALIZER ERROR  
(C3 WITH SIDE ARM CONTROLLER)

FIGURE 9.36 FINAL LOCALIZER ERROR  
(C3 WITH CENTRE STICK)
FIGURE 9.37 FINAL GROUND SPEED ERROR (C1 WITH CENTRE STICK)

FIGURE 9.38 FINAL GROUND SPEED ERROR (C2 WITH CENTRE STICK)
FIGURE 9.39  FINAL GROUND SPEED ERROR  
(C4 WITH CENTRE STICK)

FIGURE 9.40  FINAL GROUND SPEED ERROR  
(C2 WITH SIDE ARM CONTROLLER)
FIGURE 9.41 FINAL GROUND SPEED ERROR (C3 WITH SIDE ARM CONTROLLER)

FIGURE 9.42 FINAL GROUND SPEED ERROR (C3 WITH CENTRE STICK)
FIGURE 9.43 HANDLING QUALITIES RATINGS
FIGURE 9.44 HANDLING QUALITIES RATINGS
FIGURE 9.45 WORKLOAD RATINGS

Turn Coordination C1 with Centre Stick

Heading Hold C2 with Centre Stick

Raw Display Turn Coordination C4 with Centre Stick
FIGURE 9.46 WORKLOAD RATINGS

Heading Hold C2
with Side Arm Controller

Blend C3
with Side Arm Controller

Blend C3
with Centre Stick

Reference 1
--- Maximum
--- Average
--- Minimum
FIGURE 9.47 IFR CERTIFICATION ASSESSMENTS
Heading Hold C2
with Side Arm Controller

Blend C3
with Side Arm Controller

Blend C3
with Centre Stick

FIGURE 9.48 IFR CERTIFICATION ASSESSMENTS
FIGURE 9.49 IFR CERTIFICATION ASSESSMENTS (I/M CASE)
FIGURE 9.50 REFERENCE 1 HEADING HOLD CONTROLLER
10.0 SUMMARY AND CONCLUSIONS

The work reported in this document consists of the development of a helicopter simulator and its application in a study of helicopter landing approaches. The project goals were to bring on-line a Canadian helicopter research simulator and to compare ground-based simulator results in the handling qualities area with findings obtained in an actual flight environment.

10.1 Simulator Development

The helicopter simulator was built into the rear workstation of the UTIAS Flight Research Simulator. As such it makes use of many of that facility's subsystems.

(1) A generic helicopter simulator has been implemented based on a slightly modified version of the ARMCOP software package.

(2) Parameters for a Bell 205 helicopter have been employed in order to configure the simulator to represent the NAE in-flight simulator.

(3) The simulator employs a CAE series 300 motion-base and a visual display run by a Silicon Graphics IRIS 3130 workstation.

(4) The instrument panel is a computer-generated display run by a second IRIS 3130 workstation.

(5) Instrument displays have been developed for dial instruments, an EFIS and a HUD.

(6) A number of visual display data bases have been generated.

(7) Both conventional controls and an isometric side arm controller have been incorporated into the simulator.

(8) Tuning of the helicopter was carried out with the help of NAE Flight Research Laboratory personnel. In its present state the simulator has been judged to be a good representation of a Bell 205 for mild-to-moderate maneuvering. It is not suitable for hover and nap-of-the-earth flight due to the limited display field-of-view (30° x 40°).
(9) The current weak points of the simulator are its limited field-of-view and the lack of a control feel system.

10.2 Experimental Program

The experimental program was designed to duplicate the one on decelerating IFR helicopter landing approaches described in Reference 1. The latter was an actual flight test program carried out in the NAE in-flight simulator (based on a Bell 205).

(1) The UTIAS helicopter simulator was modified by the addition of stability augmentation, a flight director system and suitable flight control modes to match the NAE experimental configuration.

(2) The configuration was tuned with the help of NAE Flight Research Laboratory personnel.

(3) Six pilot subjects were trained to fly the 4 helicopter configurations selected for evaluation.

(4) The simulator conditions selected for evaluation were: motion on or off, visual display on or off, centre stick or side arm controller.

(5) Results from the test program included objective performance measures, pilot evaluations and pilot comments.

(6) It was found that simulator conditions did not have a significant impact on the experimental results.

(7) Excellent agreement between the present experiment and Reference 1 was found for flight director/control modes C1 (turn coordination) and C4 (blend) with C1 having satisfactory handling qualities while C4 had handling qualities deficiencies.

(8) Results for cases C2 and C3 (involving heading hold) from the present experiment indicated better handling qualities than those found in Reference 1. This was attributed to differences in the implementation of the heading hold controller.
(9) The isometric side arm controller produced poorer performance and less favorable pilot evaluations than the centre stick.

(10) For cases not involving the heading hold controller, the present on-ground simulator experiment produced findings similar to those of the in-flight tests of Reference 1. In the case of the heading hold controller, differences in its implementation in the two experimental programs produced corresponding differences in its evaluation by the pilots.
REFERENCES


APPENDIX A

PILOT BRIEFING MATERIAL
Guidelines for Pilot Evaluations
of Landing Approach Trials
in the UTIAS Bell-205 Flight Research Simulator

Sunjoo K. Advani
22 March, 1990
1. **PURPOSE OF THIS STUDY:**

The purpose of this experiment is to obtain pilot evaluations of the handling qualities and certifiability of various configurations related to low minima decelerating approaches in the UTIAS Bell-205 simulator. These IFR approaches will be performed using an EFIS display with a three-cue flight director.

You will be asked to perform a number of short flights, each an approach to the landing site. Various autopilot modes will be employed, and the flight director display given in either a three-cue, or zero-cue (off) condition, to assess its impact.

Following each approach series, your evaluation will be requested. You will be asked questions relating to the helicopter's handling qualities, the workload required to carry out the task, and the IFR certifiability of the helicopter. Please note that we are only interested in these qualities of the vehicle during the approach, and not in any other aspects of the simulation.

Your control activity and vehicle flight path will also be analyzed to determine how the various conditions presented to you influenced your performance. The altitude error, airspeed error and heading error will be assessed in this manner to generally determine which autopilot/flight director configurations are most suitable. These results will not be attributed to the pilot who produced them.

2. **EVALUATION TRIALS:**

The course to be flown is shown in Figure 1. The helicopter will begin in a trimmed state at an airspeed of 60 knots, an altitude of 500 feet, a heading of 0-degrees, and 1.1 nautical miles from the landing site. During approximately half the trials, the visual system simulates heavy cloud cover above 50 feet, so that IFR flight will be required above this height. The cloud layer disperses, and clear visibility results below the 50-foot decision height. During the remaining trials, clear visibility will be simulated.

A constant wind velocity of 15 knots, from either 45 degrees to the left, or 45 degrees to the right will also be encountered. The EFIS display, shown in Figure 2, indicates the airspeed and groundspeed to assist in making the approach.

The initial segment of the task will require that level flight at 60 knots groundspeed be maintained. The flight director will guide you to establish this condition. It establishes lateral position by commanding appropriate lateral cyclic control, and ground speed through the longitudinal cyclic control.

The second segment of the flight involves a descent along the 6-degree glide slope, and deceleration to 35 knots groundspeed. The flight director will initially command the descent at 60 knots ground speed. Shortly thereafter, a warning tone will be heard, indicating that deceleration is required. Again, the flight director will command the deceleration. The deceleration profile is shown in Figure 3.

When descending between 60 and 50 feet, a decision height warning tone will sound. The second segment will terminate as soon as the altitude reaches 40 feet, at which
point control of the vehicle will be taken over by the program, gently bringing the pitch and roll to zero before terminating.

2. EVALUATION TRIALS (continued):

SEGMENT 1 - Straight and Level Fight

<table>
<thead>
<tr>
<th>Initial</th>
<th>Airspeed</th>
<th>60 kts</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Distance</td>
<td>1.1 nm</td>
</tr>
<tr>
<td></td>
<td>Altitude</td>
<td>500 feet</td>
</tr>
<tr>
<td></td>
<td>VOR radial</td>
<td>0 deg.</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Final</th>
<th>Ground Speed</th>
<th>60 knots</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Distance</td>
<td>0.8 nm</td>
</tr>
<tr>
<td></td>
<td>Altitude</td>
<td>500 feet</td>
</tr>
<tr>
<td></td>
<td>VOR radial</td>
<td>0 deg.</td>
</tr>
</tbody>
</table>

NOTE: Fly at a constant altitude (500 ft.) until you intercept the glide slope.

SEGMENT 2 - Decelerating Approach

<table>
<thead>
<tr>
<th>Initial</th>
<th>Ground Speed</th>
<th>60 knots</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Distance</td>
<td>0.8 nm</td>
</tr>
<tr>
<td></td>
<td>Altitude</td>
<td>500 feet</td>
</tr>
<tr>
<td></td>
<td>VOR radial</td>
<td>0 deg.</td>
</tr>
</tbody>
</table>

<table>
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<th>Final</th>
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<th>30-35 kts</th>
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<td></td>
<td>Altitude</td>
<td>40 feet</td>
</tr>
<tr>
<td></td>
<td>VOR radial</td>
<td>0 deg.</td>
</tr>
</tbody>
</table>

Deceleration command beeper will be heard when distance from threshold = 0.6 nm

Three autopilot modes can be selected, which allow the pilot to completely disregard the yaw axis and fly with "feet on the floor". The first is a Turn Co-ordination system. With this mode activated in a crosswind approach, flying the flight director will guide the vehicle in a "crabbed" approach. The second is a Heading Hold mode which maintains to within 5 degrees the selected heading. Since we will begin with a 0 degree heading, this will be the direction nominally flown in the Heading Hold mode. In a crosswind approach, the vehicle will be held in a sideslip to maintain the correct heading.
The final augmentation configuration is called the Blend Mode. Initially, the helicopter flies a crabbed approach with the Turn Co-ordinator on. When a ground speed of 40 knots is achieved on the decelerating approach, the heading at that instant is set as the reference heading, and the Heading Hold is quickly activated. Note that this will require the helicopter to crab and slightly slip during the very last segment of the approach. A ground speed of 40 knots is typically achieved at about 80 to 100 feet altitude.

3. SIMULATION DESCRIPTION:

(a) General:
- Airframe similar to Bell-205 with Bell-412 tail rotor. The vehicle has lateral roll rate and longitudinal pitch rate stability augmentation.
- Weight 8,000 lbs

(b) EFIS Instruments:
- HSI (artificial horizon)
- Ground speed centered on moving tape display
- Digital readout of airspeed below ground speed tape display
- Target ground speed bug on moving tape display
- Vertical speed bar graph
- Altitude bar graph, and digital readout
- Compass
- Localizer Scaled at 3 degrees per division shown when aircraft is outside a 1900 foot radius from the threshold, and becomes 100 feet of lateral displacement per division when inside this radius. (This radius corresponds to the distance when a 100 foot lateral offset subtends a 3 degree localizer error.)
- Glide Slope Six degree approach. Instrument is scaled at 1.5 degrees per division outside a 1900 foot radius from the threshold, becoming 50 feet of vertical displacement per division inside this range. The glide slope and localizer bars will not be displayed if the EFIS is in "Cruise" mode. The modes (Cruise/Approach) can be selected from the touch screen.
- Turn rate indicator
- Side force indicator (ball), shows specific side force
- Engine rpm (percent)
- Rotor rpm (percent) This will closely match the engine rpm, since the governor is on.
- Flight director: Lateral and longitudinal commands are similar to a tracking task: If the centre of the circle is above and to the left of the HSI centre mark, pull back and roll left to center the two. The collective axis shows a vertical bar below the circle if more collective displacement is required.

(c) Controls:
Lateral, longitudinal cyclic.
Rudder pedals. These will be disabled when flying the approach trials from the commencement of the flight.

Collective, with throttle wheel.

Two-axis side-arm controller.

This is a force-sensitive device, which can be used to replace the conventional lateral/longitudinal cyclic stick. It may be used as a means of minimizing the inertial force feedback caused by lateral motions of the simulator, which tend to destabilize the system.

The side-arm controller incorporates a follow-up trim technique, which integrates the applied force. This allows the input force to be relieved whenever a new trim condition is established. In effect, the pilot needs only to apply a force when a change from the trim position is required.

4. INSTRUCTOR INTERFACE

The simulator operator will select the various flight scenarios required to evaluate the autopilot and flight director models. The parameters to be varied are:

1. Selection of wind condition, 15 knots from either 045 or 315 (randomly set between trials).
2. Autopilot mode to be used: turn co-ordination, heading hold or blend.
3. Selection of conventional cyclic stick or side-arm controller.
4. Flight director mode: A full three-cue system is the default setting. Shutting this device off completely is also possible.
5. Simulator motion on or off.
6. VFR or IFR weather conditions.

You will be informed of these conditions before the start of each trial set, comprised of four runs under the same conditions.

You will be asked to fly the following approach configurations:

1. Turn co-ordination mode 3-cue flight director
2. Heading hold mode 3-cue flight director
3. Blend mode 3-cue flight director
4. Turn co-ordination Raw display (glide slope, localizer, ground speed)
5. ACCEPTABLE PERFORMANCE LEVELS:

The pilot will be given sufficient time to practice these approaches until he and the
evaluator feel that there is reasonable consistency. The baseline limits are 25 feet of error
on the glide slope and localizer, and 3 knots of groundspeed error at the termination of the
run. Note that the glide slope and localizer errors of 25 feet are represented by 1/4 division
on the localizer, and 1/2 division on the glide slope displays.

6. SIMULATOR SAFETY PROCEDURES

The simulator hardware and the software which runs this experiment have been
designed to work safely and reliably. To ensure your safety, observe the following rules:

1. No smoking
2. Always use the five-point harness. Remain seated and strapped in at all times while
   the simulator is moving.
3. When the experiment is over, remain seated and strapped in until the lights come on
   and the experimenter tells you that you may unstrap.
4. Do not open the door until the gangway has reached the DOWN position and shut
   itself off as indicated by the blue light beside the door.
5. The experimenter is always in contact via the intercom. Contact him if you need
   instructions or advice.

In the unlikely event of an emergency such as a fire, there are a number of features built
into the simulator which will allow you to exit quickly and safely. Follow these steps:

1. Press the large red panic button on the right forward panel to shut down the motion
   base.
2. Tell the experimenter what is wrong.
3. Remain seated and strapped in until the motion base has stopped moving and the
   experimenter tells you that you may unstrap.
4. If there is no immediate danger, wait for the gangway to come down and exit on it.
   If you need to exit immediately, open the door and climb down using the rope
   stowed above the door.
5. If the exit door will not open, exit using the escape window next to the right-hand
   pilot's seat, behind the helicopter station. Climb down using the rope stowed in the
   compartment directly below the escape window.
6. In case of fire, your first priority is to get out of the cab safely. If the fire is small
   and exposed, you may use the fire extinguisher.
1. Flight begins with aircraft in trimmed level flight at 60 kts, 500 feet altitude, 1.1 nm from runway threshold.

2. Establish 60 kt groundspeed and maintain as wind fades in. Pitch flight director (if used) will provide pitch command for 60 kt ground speed. Maintain 500 ft altitude.

3. Intercept six degree glide slope. Nominally occurs when distance to threshold is 0.8 nm. Collective flight director (if used) will command correct descent on the glide slope.

4. Warning tone indicates onset of deceleration cue when distance to threshold is 0.6 nm. Follow flight director (if used), or groundspeed bug to maintain this profile.

5. Fog breaks at 50 ft. (if IFR weather).

6. Program terminates at 40 ft.

Figure 1. Course Profile for Bell 205 Evaluation Trials
FIGURE 2
EFIS DISPLAY
Warning tone indicates onset of deceleration cue when distance to threshold is 0.6 nm.

Based on $V_{kts} = 1.007 \sqrt{\text{DME ft.}}$

$a_x \approx 0.045 \text{ G}$

FIG. 3: DECELERATION PROFILE
APPENDIX B

PILOT COMMENTS
Pilot Comments (Ordered by Cases)

1. Turn Coordination C1, Centre Stick, IFR, No Motion

Little compensation is required to fly the task. (S3)

2. Turn Coordination C1, Centre Stick, IFR, Motion

Torque indicator's scale is low and difficult to read. Should be moved to left side of display under VSI. Collective flight director is sluggish. (S1)

3. Turn Coordination C1, Centre Stick, VFR, No Motion

Collective flight director is sluggish and doesn't provide the desired lead. (S1)

Slightly unfavorable handling qualities due to lateral tracking task. Workload is higher without motion (and in the absence of heading hold). (S2)

External cues help in lateral tracking by using peripheral vision. (S3)

VFR distracting in peripheral view. (S4)

Not much difference between IFR and VFR. (S6)

4. Turn Coordination C1, Centre Stick, VFR, Motion

Turn coordination has slightly worse handling qualities than heading hold because of large crab angles which result in a strong pitch coupling. The pitch errors become more frequent and larger. There is a significant stepiness in the outside display. (S1)

5. Heading Hold C2, Centre Stick, IFR, No Motion

Collective flight director is sluggish. (S1)
IFR case is easier and less distracting than VFR, because the pilot does not try to compensate between flight director and visual cues. (S2)

6. **Heading Hold C2, Centre Stick, IFR, Motion**
   Torque gauge location is not so bad as previously thought. Cross-check with VSI is not so important in the presence of the flight director. This task is effortless. May not be enough collective washout gain. (S1)

   A very easy task. (S2)

7. **Heading Hold C2, Centre Stick, VFR, No Motion**
   No problems with handling qualities or with workload. (S1)

   Decrease in control response without motion. Motion cues help. (S2)

   No problem with this task. (S3)

8. **Heading Hold C2, Centre Stick, VFR, Motion**
   Collective flight director is not ideal: still feels sluggish. (S1)

   Don't need to know wind direction with heading hold. More important with turn coordination. (S3)

   No real compensation required. Good handling qualities. (S5)

   At interception of glideslope, collective flight director moves very rapidly, and is otherwise quite slow and creeps up with little warning. There appears to be a 5 Hz pitch oscillation which is easily induced if yoke friction is tight. The buffet amplitude seems very low. (S6)

9. **Heading Hold C2, Side Arm Controller, IFR, No Motion**
   A little bit of pitch compensation required. Slightly lower handling qualities than with yoke (3 and 2 respectively). (S1)
Heading hold/side arm controller places handling qualities in level 1 quite easily. (S2)

Considerably more difficult with side arm. Would take a lot of getting used to. (S4)

10. **Heading Hold C2, Side Arm Controller, IFR, Motion**

Uncomfortable oscillatory behaviour with sensitive side arm controller, however the tracking is good. (S2)

Requires considerably more compensation and more input than with centre stick. (S5)

Discrepancy between collective flight director and glideslope pointer is slightly annoying. (S6)

11. **Heading Hold C2, Side Arm Controller, VFR, No Motion**

Stiff side arm gives some abruptness in its response which degrades HQ. Non-instinctive conscious inputs are required — higher workload. (S1)

Visual scene distracting because flight director is good. Without motion cues and with VFR display, visual scene is too dominant with runway bobbing. Unnatural feedback from the system with side arm controller/no-motion. (S2)

Side arm controller is twitchy and not as responsive as the centre stick. (S5)

12. **Heading Hold C2, Side Arm Controller, VFR, Motion**

Since the collective flight director is sluggish, pilot must use knowledge of the power required characteristics to provide lead for the collective flight director. (S1)

Both pitch and roll gains are high. High gain pilots would have difficulty with the resultant handling qualities. Oscillations increase workload. (S2)

Acceleration cues from motion help greatly with side arm controller active. (S4)
Side arm controller gives roll oscillations with its high sensitivity. (S5)

Pitch is sensitive. With friction on collective control set a certain way, a pilot induced oscillation is easy to start. (S6)

13. **Blend C3, Centre Stick, IFR, No Motion**

No big difference with motion off since there is no turbulence modelling. Otherwise, this comment would be void. (S1)

Motion would really help the handling qualities. (S2)

Fade-in of wind is difficult with blend. (S3)
Motion would help to reduce workload/improve handling qualities. (S5)

14. **Blend C3, Centre Stick, IFR, Motion**

Flight director symbology is good: Ring size is good and makes small errors obvious (more obvious than NAE flight director). High gain system with pilot in the loop. Collective flight director is the only minor problem, however the task would be easy even without the collective flight director by use of the raw glideslope display. (S1)

Complacency of pilot due to ease of task lowers gain. Too easy; could build up errors quickly. Might be different in reality - radio communication, etc., would increase overall workload slightly. (S2)

Mild heave oscillations induced by pitching motions. Feels a bit strange. (S6)

15. **Blend C3, Side Arm Controller, IFR, No Motion**

Dynamics change during the transition to heading hold. Dynamics are less intuitive in this mode, though it's not a major problem. The airflow noise over the cabin in a sideslip is an important cue which is missing. Effort (workload) is more physical than mental. (S1)

Flight is more difficult in turn coordination phase than in later heading hold mode. (S3)
Some difficulty with side arm controller might diminish after considerable practice. (S5)

Collective flight director muddled with the pitch attitude bars. Collective flight director scale is a bit small. (S6)

16. Blend C3, Side Arm Controller, IFR, Motion

Collective flight director is sluggish for large errors. (S1)

Turn coordination phase more difficult than heading hold phase. (S3)

Pitch oscillation is annoying; attributed to side arm controller sensitivity. However, roll sensitivity could be increased to improve lateral tracking. (S5)

17. Raw Display C4, Centre Stick, IFR, No-Motion

Clutter in localizer bug vicinity makes final phase of flight difficult when low altitude runway symbol appears. (S1)

Large excursions in tracking. Maximum compensation required. Performance inadequate. (S2)

Poor tracking. Deficiencies may be due to lack of IFR experience. (S4)

18. Raw-Display C4, Centre Stick, IFR, Motion

Confidence in repeatability is low. Excessive errors. High mental effort required to maintain stability. (S1)

Can monitor some secondary instruments, though this is very difficult. (S5)

19. Raw Display C4, Centre Stick, VFR, No Motion

Adequate performance with moderate errors. (S1)
VFR helps lateral task; lack of motion makes it difficult. (S2)

20. **Raw Display C4, Centre Stick, VFR, Motion**

Very objectionable handling qualities. Extensive compensation and effort required for minimal control. Not a repeatable task; inconsistent performance. Hard to recover the errors. Thinking required all the time. (S1)

Considerable mental integration required to reduce lateral errors. Coupling of controls such as collective to pitch becomes important. Must be very quick in tracking and observing all three bugs. Glideslope is easiest to follow, and the other two require significant effort to track. The ground speed quickly drops off when the flight is about to terminate, probably due to the attitude of the vehicle (nose up), and the pitch sensitivity of the system. (S6)
This project involved the development of a helicopter flight simulator and its application to study IFR landing approaches.

The helicopter simulator was implemented in the rear workstation of the UTIAS Flight Research Simulator. It employed all the major subsystems of that facility including the motion system, visual display system, sound system, and computer system. In addition the flight controls and seat of a Bell 205 helicopter were installed and interfaced to the flight computer. An electronic flight instrumentation system (EFIS) was developed to act as the pilot/helicopter interface. A side arm controller was also included in the simulation. The helicopter flight equations were based on the ARMCOP software package. The system was tuned with the help of an experienced Bell 205 test pilot.

The simulated flight test project was directed towards establishing the ability of the newly developed simulator to simulate handling qualities results achieved in actual flight. The task selected was that of decelerating IFR approaches using flight control systems and a three cue flight director developed and tested by the National Aeronautical Establishment. A group of 6 evaluation pilots repeated flying tasks previously completed in an actual Bell 205. It was found that good agreement between simulator and test flight data was obtained except for one configuration. This configuration was found to be easier to fly in the simulator. The cause of this was felt to be incorrect simulation of the corresponding flight control system.

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