HUMAN ROLE IN THE CONTROL-LOOP
OF THE AUTOMATIC LANDING AIRCRAFT

BY

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ABSTRACT

The object of this thesis is to study what kind of information
the pilot can obtain from three different kinds of windshield displays
during automatic approaches and landings, and if he is able to detect
the possible malfunctions in the automatic system from his display.
In the study, an airplane landing was simulated by using analogue
computer and two degree of freedom moving-base simulator. The picture
of the runway projected to the pilot was his only visual information
source in the cockpit.

The tests were performed by feeding step and ramp disturbances
into roll rate or pitch rate integrators in the analogue computer, and
the roll, yaw and pitch response times were calculated from the recordings.

It can be seen that the roll control only the most simple picture
of the runway, two lines representing the runway boundary lines are
enough in most cases, but the yaw control lacks accuracy and damping
in this case. Adding the horizon to this picture improves a little
yaw control by allowing the pilot to add yaw rate feedback to his
control.
The pilot was not able to obtain enough information from these two simplified pictures of the runway to control the longitudinal axis and several crash landings were recorded during this part of testing.

The flight path marker provided enough information to the pilot for controlling also the longitudinal axis, and the response time to moderate pitch rate disturbances was less than half of the response time without the flight path marker. No crashes were recorded when the flight path marker was used.

In the tests, both moving-base and fixed-base simulation were made, and it was found that the fixed-base lateral response times were about twice as long as the respective moving-base values.

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SYMBOLS

C = mean aerodynamic cord.
S = wing area.
W = weight.
N = coefficient in yaw equation of motion, particularized by subscript.
P = coefficient in roll equation of motion particularized by subscript.
Y = coefficient in side force equation of motion particularized by subscript.
M = coefficient in pitch equation of motion particularized by subscript.
Z = coefficient of lift equation of motion particularized by subscript.

\[ I_{xx} \]
\[ I_{yy} \]
\[ I_{zz} \]
\[ I_{xy} \]
\[ I_{xz} \]
\[ I_{yz} \]

= moment of inertia.

\[ I_{xy} \]
\[ I_{xz} \]
\[ I_{yz} \]

= products of inertia.

\[ C_{yf} \]
\[ C_{y\delta} \]
\[ C_{l_{\phi}} \]
\[ C_{l_{\psi}} \]
\[ C_{l_{\phi}} \]
\[ C_{l_{\delta}} \]
\[ C_{l_{\delta}} \]

= change in side force coef. with variation in \( \alpha \).
= change in side force coef. with variation in \( \delta \).
= change in rolling moment coef. with variation in \( \phi \).
= change in rolling moment coef. with variation in \( \psi \).
= change in rolling moment coef. with variation in \( \phi \).
= change in rolling moment coef. with variation in \( \delta \).
= change in rolling moment coef. with variation in \( \delta \).
\( C_{1\delta_r} \) = change in rolling moment coef. with variation in \( \delta_r \).

\( C_{n\dot{\psi}} \) = change in yawing moment coef. with variation in \( \dot{\psi} \).

\( C_{n\beta} \) = change in yawing moment coef. with variation in \( \beta \).

\( C_{n\dot{\beta}} \) = change in yawing moment coef. with variation in \( \dot{\beta} \).

\( C_{n\phi} \) = change in yawing moment coef. with variation in \( \phi \).

\( C_{n\delta_{\alpha}} \) = change in yawing moment coef. with variation in \( \delta_{\alpha} \).

\( C_{n\delta_{\alpha}} \) = change in yawing moment coef. with variation in \( \delta_{\alpha} \).

\( C_{L\alpha} \) = change in lift coef. with variation in \( \alpha \).

\( C_{L\delta_{\alpha}} \) = change in lift coef. with variation in \( \delta_{\alpha} \).

\( C_{m\dot{\alpha}} \) = change in pitching moment coef. with variation in \( \dot{\alpha} \).

\( C_{m\dot{\phi}} \) = change in pitching moment coef. with variation in \( \dot{\phi} \).

\( C_{m\delta_{\alpha}} \) = change in pitching moment coef. with variation in \( \delta_{\alpha} \).

\( C_{m\alpha} \) = change in pitching moment coef. with variation in \( \alpha \).

\( C_{m_o} \) = change in pitching moment with zero angle of attack.

\( b \) = wing span.

\( d \) = distance of pilot's eye from windshield.

\( g \) = gravity acceleration.

\( h \) = altitude.
l = length of the runway.
m = mass.
p = roll rate.
q = pitch rate, dynamic pressure.
r = yaw rate.
s = Laplace operator.
v = velocity.
x = distance to the end of the runway.
y = lateral displacement from the centerline of the runway.
\alpha = angle of attack.
\beta = angle of sideslip.
\gamma = flight path angle.
\delta_a = aileron deflection.
\delta_e = elevator deflection.
\delta_r = rudder deflection.
\theta = pitch angle.
\phi = roll angle.
\gamma = yaw angle.

SUBSCRIPTS

l = rolling moment.
L = lift force.
m = pitching moment.
n = yawing moment.
y = side force.
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CHAPTER 1

INTRODUCTION

This is a study about a pilot's ability to get information from a head-up display for an automatic landing. Independent of the increase of size, speed, flight altitude, cost and complication of aircraft, the take-off and landing of an aircraft is still greatly dependent on the weather conditions. Considerable effort has been made during the past two decades to make these essential parts independent of weather, and now one can see partial improvements in this field\(^1\). British European Airlines have already made automatic landings in scheduled passenger flights between Paris and London, and several military planes are equipped with all-weather capable systems\(^1\). All-weather landings done by these landing systems are only partially successful, although no fatal crashes have occurred. The pilot, the most adaptive and important part of the whole system is neglected, which is probably the reason why none of the systems have been internationally accepted to date. The pilot is still in charge of his plane and load, but he does not know well enough what is happening around him to make judgments dealing with his passengers' safety. He must interpret his situation by using data from his instrument panel and mentally picture what is happening around him\(^1,7\).
In most cases when everything goes as planned, the picture in the pilot's mind is similar to those he is used to during visual approaches and landings. However, in case of a malfunction in some part of the automatic system, is he still able to form a true picture in his mind just by using the data from his instrument panel?

At present, if the pilot finds something unusual happening in the instrument panel, his first reaction is to interrupt the approach and initiate a go-around maneuver and then to find the reason for the unexpected information. By using up-to-date instrumentation, the go-around command need not necessarily be just a malfunction in the control system\(^7\). Being aware of the possibility of malfunctions, the pilot may also interpret many other facts as a malfunction if he is tired, nervous, or not aware what is happening. Also, he lacks the ability to estimate the importance of a malfunction and may tend to overestimate them\(^7\).

There is another approach to the instrument landing technique, where an artificial picture of the runway is presented to the pilot during the approach and landing, so that he has a visual basis for his decisions.

In this case, the pilot can use all his natural experience to make the vital decisions for landing: Should I continue with my automatic control or revert to a manual mode of control? Should I attempt to land or initiate a go-around maneuver?
If the artificial picture of the runway is used during all landings, the pilot can always compare it to the actual appearance of the runway during visual landings. The interpretation of this picture is then easier during low visibility automatic landings. It is not known if the solution to the display problem significantly eases the automatic landing problem. By giving the pilot the possibility of understanding what is happening around him, he accepts the concept of the automatic landing system as a helper to relieve his workload.

This work concentrates on the evaluation of the minimum visual display of the runway during automatic landing and finding out how much and how accurate information the pilot can obtain from such displays. Three basic types of display forms were studied:

1. Two lines, representing the runway boundary lights.

2. Three lines, two of them representing the runway boundary lights and the third, the horizon.

3. Two lines and one dot, the two lines representing the runway boundary lights and the dot showing the spot where the velocity vector of the aircraft intersects the ground.

The purpose of this work is not to study what kind of parameters are necessary in the display for completing the landing and how there are generated. This area is rather well covered by other researches. In Chapter II the conventional cockpit panel instrumentation is discussed. It is found that the pilot is not able to obtain the inform-
ation fast enough by using that kind of instrumentation to form a picture in his mind to follow the landing procedure. A more natural way of giving the necessary information to the pilot is a windshield projection system which is described next and finally the basic system analysis is given.

In Chapter III, the control of an airplane during landing is discussed and the change of the dynamics of the aircraft due to changes of velocity are derived. Then the performance of the manual and automatic control during landing are compared, and the effects of certain possible failures are discussed.

To find out the pilot's ability to get information from three different display pictures, a test program is set up. In Chapter IV, the test set up and experimental equipment is described, and justification for selecting the display pictures is given.

Chapter V describes the basis of the tests and how they are performed, and in Chapter VI the results are given. From the results, it is found that there is a great difference in the pilot's response if the tests are made on the fixed base or on the moving base simulator. The moving base simulation results show that in order to be able to control roll, a very simple picture is needed, and for that, two runway boundary line rows are sufficient. In order to control yaw also, the display picture must have some information so that the pilot can put lead to the lightly damped yaw-loop. For pitch control, the flight
path marker is superior over two other displays, and only by using the flight path marker, the pilot was able to control the plane in pitch.
CHAPTER II

2.1 Instrumentation of the Cockpit for Landing

Current cockpit panel instrumentation makes it possible for the pilot to obtain attitude, heading, glidepath deviations, altitude, airspeed, rate of climb and flight director command information in various particular and integrated forms. Information on the panel is generally presented in isolated bits, and the form of the instruments and change of the parameters does not always relate to the real world. This kind of information is enough for cruise navigation, because the pilot has time to form a picture about his situation, when parameters deviate slowly from their desired positions and long delay time is acceptable in corrections.

The display situation changes completely when decisions must be made very fast. Then the pilot must be aware of the situation, where several parameters change simultaneously and tolerances on variables are very narrow. The piecemeal flow of the panel information effectively prevents the pilot from responding as easily and naturally as during VIR (visual-flight-regulations) conditions, when flow of information comes from the field of vision. Limitations in the pilot's capability to perform the desirable control movement causes decreases in the pilot's gain in that control loop, which is vitally important during approaches and landings. A fast decision followed by a fast
correction is essential in the case of correcting for the effects of the atmospheric turbulences or some malfunctions in the control loop.

By using his regular channels of information like feeling of linear and angular accelerations, the pilot can make natural corrections as is also shown in the chapter VI of this report. The low gain acceleration feedback is not enough to measure the prime variables and the pilot finds it impossible to separate small malfunction accelerations from aerodynamic turbulence accelerations. The way the human pilot collects information to form a picture of the outer world for flying is not entirely clear and the layout of the instruments on the instrument panel is more traditional than based on thorough research\(^\text{13}\), hence, the pilot is not able to collect and synthesize the information he gets from the instrument panel as easily as he can interpret and analyze the picture he sees through the windshield.

One notices, that during the instrument approaches, the pilot has to see the approach lights before the weather minimums are reached, and naturally he wants to see them as early as possible. That is why he desires to get all of his early information through the windshield, because during the final approach and flare-out, he gets them through the windshield anyway. Focusing eyes back to the windshield after looking at the instrument panel takes a certain time, which forms a natural delay and the concentration decreases the effectiveness of estimations and decisions. On the other hand, that makes the pilot less willing to move his eyes to the instrument panel. On the pilot's side, the panel instrumentation is not the most effective during landings.
2.2 Windshield Display

The importance of windshield projection of the instrument values was already noticed after World War II. The development of collimator gunsights made it possible to project the image of the sight-picture in front of the pilot. By developing this kind of system\(^1\), it is possible to project any kind of picture to the windshield and move it arbitrarily in certain regions.

Figure 1 shows a gunsight used as windshield-projecting device with an attached cathode ray tube (CRT). This configuration will project, for example, a picture shown in figure 2\(^1\). When the picture is projected to the windshield so that the light-source and image are in the focus plane of the lense-system\(^1\), the light coming through the image reflects from the windshield and is collimated. That is why the position of the pilot's eyes is not important to see the right size picture, but in order to see the relative size of the image picture with respect to the real picture, his head's distance from the windshield must be kept constant. Small lateral displacements does not change the size of the image picture.

The picture generated by the gunsight and shown in figure 2, contains two parts. A circle and a dot are generated by upper part of the system and they can be arbitrarily displaced by turning the mirrors a and b in figure 1. By turning the mirror a, the picture moves vertically and by using the mirror b, deflection takes place in the horizontal plane. Turning of the mirror a does not affect to the position of the CRT picture.
Figure 1 Gunsight as a Windshield Projector
Figure 2 A Projected Picture

Figure 3 Angles of the Flight Path
If the CRT picture contains the information about the runway, for example, runway lights and is properly located with respect to the body axis of the aircraft, the pilot sees these dots on the windshield at the same places as he sees the real runway lights when they appear. Since the picture of the runway can be generated independently of weather conditions, the pilot can have a "real world picture" in front of him. The image projected on top of the runway-picture can be moved independently. If it is moved, in the vertical plane by amount of $\theta - \alpha$ and in horizontal plane by amount of $\beta$, in figure 3, the dot portion represents the direction of the velocity vector of the aircraft. This is the way the test picture No. 3 in chapter VI has been generated. In this simple picture, there is still one display parameter available, namely the diameter of the circle. Very naturally one relates the diameter of the circle to the airspeed of the aircraft, because the velocity is one of the most important parameters during landing that the pilot currently follows. In this case, the pilot need not look down to the instrument panel to obtain that information.

Using these techniques, described in more detail in reference 1, there are at least six company projects and several other suggestions, giving all the information the pilot needs for landing. One such picture shown in figure 4 is proposed by Sperry Gyroscope Company.

Unfortunately, the generation of a reliable picture or obtaining the data for generation of it is not yet fully solved in any of the proposed systems, and no one has yet used in the regular service.
2.3 System Analysis

A solution to all-weather landing problem demands a system approach in the broad sense. Both men and equipment are involved in building the whole system. The performance of the equipment must satisfy the cockpit crew, and the equipment must respond to the commands imposed by the crew in a manner, that the crew considers satisfactory. Figure 5 shows a block diagram of a man-aircraft system. The display is in the feedback path and so the performances of the system depend on the performances of the display. The pilot's information includes two paths in real flight case. The primary path is the display, but a very important one to be considered is his "motion sensing" detection of both linear and angular accelerations. This is the reason for using both fixed and moving base simulations in this work.
Figure 4 Windshield Display Configuration for Final Approach by Sperry Gyroscope Company

Figure 5 Block Diagram of the Man-Vehicle System
CHAPTER III

3.1 Control of an Airplane during Landing

In this chapter, the control and behavior of an airplaine during landing is discussed both in manual and automatic control cases, although only longitudinal control is discussed in the automatic control case. Behavior of an airplane as a part of the control system has been studied very carefully by many authors, and the modern autopilot and flight control techniques are largely based on these studies. Shown in the figure 6 is a manual longitudinal control system of an aircraft including the pilot and the display. The transfer functions of an aircraft can be estimated rather accurately for a given velocity by using the aerodynamic derivatives, and one finds the transfer functions largely dependent of the velocity and the angle of attack. Being a part of a closed loop, the changes in the aircraft dynamics reflect to the response of the whole system.

3.1.1 Effects of Changes in Velocity on the Equations of Motion

In appendix A, the transfer functions of the aircraft are derived in such a form that derivation of the equations due to velocity leaves only those parts left, which are implicitly dependent on the velocity. From the table 1, one can see that the change of the dynamic coefficients in the equations of motion is approximately proportional to the relative change of the velocity, and the order of derivative of the coefficient. When during landing, the speed may vary about ±10%,
<table>
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<th>Partial Derivative with respect to velocity</th>
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<tr>
<td>$P\dot{\phi}$</td>
<td>$P\dot{\phi} \cdot (-2 \frac{d\nu}{\nu})$</td>
</tr>
<tr>
<td>$P\dot{\psi}$</td>
<td>$P\dot{\psi} \cdot (-\frac{d\nu}{\nu})$</td>
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<td>$N\dot{\psi}$</td>
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<td>$N\dot{\phi}$</td>
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<td>$Y\dot{\alpha}$</td>
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<tr>
<td>$M\dot{\phi}$</td>
<td>$M\dot{\phi} \cdot (-2 \frac{d\nu}{\nu})$</td>
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<tr>
<td>$M\dot{\theta}$</td>
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<td>$M\dot{\alpha}$</td>
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</tr>
<tr>
<td>$\alpha$</td>
<td>$\alpha \cdot (-2 \frac{d\nu}{\nu})$</td>
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</table>
one may expect that the coefficients may change about the same amount.

As shown in appendices A and B, the aerodynamic coefficients also are functions of velocity and angle of attack, especially when the angle of attack is large. The author was unable to obtain any data describing the behavior of aerodynamic coefficients when the angle of attack gets large or when the airspeed gets less than 1.3 x speed of stall. Perhaps no one just now knows what is happening to the large aircrafts in that region, and an answer cannot be given for that reason. As shown in table 1, the relative change in the angle of attack is twice the relative change in the velocity and so far 10% change of velocity from its approach value. The angle of attack changes approximately 2.3°, which should already be considered. Because the exact values of the partial derivatives due to the angle of attack were not known, all the derivatives were assumed to be zero, and the results are taken more qualitatively regarding only changes due to velocity.

In figure 7 is shown what this does mean to the numerator and denominator of the \((\mathbf{PP})_{A}[\delta_{g}, \Theta]\) after closing the pitch loop with unity feedback gain. The equations are derived in appendix A, equations A.26 and A.27. One notices that when the velocity decreases, the poles and zeros move toward the origin, thus increasing the rise time and making the system slower.

The entire performance of the aircraft from pitch reference \(\Theta_{R}\) to glidepath angle \((\mathbf{PP})_{A} [\Theta_{R}, Y]\) can be described by using the
Figure 6  A Block Diagram of a Manual Landing System Longitudinal two-degree-of-freedom Guidance Loop
\[
(PF)_{\Delta}(s) = \frac{1.07(s+0.825)}{s(s+1)(s+1.01V)}
\]

**Symbols**

- \(x, \theta\) = nominal values, open-loop pole and zero
- \(\triangle\) = velocity 10% higher than the nominal value
- \(\triangledown\) = velocity 10% lower than the nominal value
- \(\Box\) = closed-loop pole

**Figure 7** Root-Locus Diagram of and how 10% deviations in the velocity affect to closed-loop values
performance of the closed-pitch loop. The layout of the roots of \( \lambda \) 
\( (P_{\alpha}^{P}\left[\Theta_{q}\right]) \) is given in figure 8, and a Root-Locus is drawn to show the 
closure of h-loop. One can see from it that:

1. Decreasing of the velocity decreases the gain of the system, and
2. Decreasing of gain makes the response of the system slower and less damped.

An advice to the pilot, based only on these two figures 7 and 8, 
would be do not do anything because everytime he increases his gain 
from zero, the system gets more oscillating and less damped. This 
kind of advice is, of course, impossible and wrong, but what the pilot 
could do to improve the stability of his system and to be able to 
make corrections to the glidepath. In order to make corrections, he 
must have some gain in pitch loop, but that is very difficult to 
generate because pitch angle is hard to detect. How hard it actually is, 
is shown in chapter VI, where one sees that from a simple picture, only 
the test pictures 1 and 2, it is nearly impossible to detect the pitch angle and so the loop cannot be closed effectively and the control is 
very poor.

The possibilities for improving the system are for the pilot:

1. Detecting pitch-rate from display.
2. Closing the \( \gamma \)-loop instead of \( \Theta \)-loop.

Results from first of these possibilities is shown in figure 9, 
where pilot has generated a proportional and rate feedback from the
The layout of the roots of the closed pitch-loop (PPF) in [\( \Theta_k, H \)] and how it affects the open-loop of closed H-loop.
display. The Root-Locus shows that the system gets more stable and
faster the higher the pilot's gain is.

3.1.2 $\gamma$-Loop Control

The second possibility, closing the $\gamma$-loop demands a new approach,
namely generating the glidepath angle to the display. This can be
done, however, by using the display system described in chapter II.
By using equations A.26 and A.27, a transfer function

$$
(\text{FRA})_{\text{E, } \gamma} = \frac{0.060 (s+4.21) (s-3.44)}{8 (s+0.955 + 36j)^2}
$$

(3.1)

Figure 10 shows the Root-Locus of equation 3.1 after the $\gamma$-loop
is closed. During the simulated approach, tests proved that the pilot
need not try to use high gain to be able to make corrections by using
$\gamma$-loop closing techniques.

3.1.3 Summary

The pilot's capability to control the longitudinal motion of an
airplane depends very radically on the display. By using the basic
pictures of the runway like test pictures No. 1 and 2 in appendix C,
he is not able to obtain much information and the response of the system
to all disturbances is poor. On the other hand, if he has a display
like test picture No. 3 in appendix C, he is able to close $\gamma$-loop and
control the glidepath angle and at the same time, his vertical velocity
very accurately.
\[
(PF)_{\|0}^{\|0} = \frac{0.2 \cdot K \cdot (s + 1.0) (s + 0.825)}{s (s^2 + 1.015 + 1.65)}
\]

Figure 9 Root-Locus of the Pitch-Loop closed with Pilot's Lead
\[ \frac{(PF)_{A}[\delta_F, \gamma]}{s} = \frac{0.060 (s + 4.20)(s + 3.40)}{s(s + 0.955 ± 0.36j)^2} \]

Figure 10 Root-Locus showing the \((PF)_{A}[\delta_F, \gamma]\) when the Pilot closes the \(\gamma\)-Loop. The Closed-Loop Poles are at Unity Feedback Position.
3.2 Differences between Human and Automatic Longitudinal Control Systems

Figure 11 presents an automatic longitudinal control system block diagram proposed for automatic landing of a Caravelle by Lear Siegler, Inc. One of the biggest advantages of an automatic control system can be noticed from this picture, namely that the designer can rather arbitrarily decide what kind of compensations and feedbacks he uses. By knowing the values of fixed parameters, like aircraft equations of motion, the designer can, by using control techniques methods, obtain very large variety of responses and to meet the specifications is in most cases question only about price and instrument components.

3.2.1 Autoland System

The classical problem which must be overcome in an automatic landing system to improve glidepath coupling is beam convergence and its detrimental effect on system stability. Since the beam width and therefore the gain (millivolts/foot) is a function of the reciprocal of distance from the glidepath transmitter, the system gain varies over a very wide range from glide engagement to minimum altitude. For example, from altitude of 1,500 feet to altitude of 50 feet, the beam gain varies over a 1-to-30 range$^3$.

To compensate this, the couple gain is scheduled to vary from 30-to-1 and the system gain would be about constant. Typically, the gain scheduling is programmed by time, or by altitude or by the
Figure 11  Block Diagram of an Automatic Landing Longitudinal Two-Degree-of-Freedom Guidance System
marker beacon signals. In the Lear Siegler system, the gain scheduling is performed so that the gain $K(L)$ is decreased from 1 to 0 when the altitude decreases from 250 feet to 50 feet, and the system is designed to tolerate 10 to 1 variations in gain. Now the system is stable and accurate enough, but still there is question about feedback instrumentation to be solved. None of the conventional, radar, inertial of barometric altimeters alone can give accurate enough altitude rate due to noise, and therefore a special computer was designed by Astronics, using barometric signal and acceleration to from vertical velocity.

Another problem is altitude feedback and the limitations caused by terrain roughness to system stability. All the problems have been solved and the system is to be installed to the Caravelle.

3. 2.2 Human System

The human pilot does the same task by using the information through the windshield during VFR approaches and landings. His advantage is adaptivity in such a large region that no automatic systems can compete with it. His limitations are accuracy and limited gain, but that can be overcome by proper design of the system.

3. 2.3 Possible Failures

In a modern aircraft, the failure rate is decreasing but still during the approach thrust may drop from one of the engines, ailerons spoilers or flaps may affect asymmetric load or gyros and other
instrument may have drift or bias causing disturbances to all the pitch, roll and yaw channels. These disturbances can be ramp or step type by nature, and to prevent the effects to be disasterous, human decisions are needed.

During the Autoland approach, the pilot stares at the instrument panel and tries to form a picture of the situation so that in case of malfunction in any part of the system, he could take over the control. The information flow through instrument panel is blamed to be too slow for this work. To study the information flow and the pilot's response through the windshield display, a series of step and ramp inputs were initiated from outside to the roll and pitch loops in the simulated landing situation. The step inputs were preferred because the results are easier to analyze.

3.2.3 Summary

In the aircraft, the effectiveness of an automatic control system was never in the past trusted more than the human control system. The pilot was the dominating one in the cockpit and the autopilot was his slave. Considerable effort was put to improve the automatic system, and the pilot was thought to be able to use his adaptivity and follow the system development. Because he was not given the means to do this, he was pushed aside and the automatic system does the work, without giving him possibility to control it due to the lack of information. The performance of the total system can be improved in the automatic
landing systems, if instead of neglecting the pilot, his adaptivity is used by allowing him to monitor a picture similar to that he sees during VFR approaches and landings. Each approach and landing is made by an automatic system and the pilot follows from his picture, that the system works as desired. In that case, the automatic landing is a relief to the pilot's workload, not an extra burden to rise his tension level.
CHAPTER IV

4.1 Test Setup

The objectives of the research program are to compare the relative effectiveness of three different windshield projection displays during the approach and landing phases in fixed and moving based simulations. Tests were made at the M.I.T. Department of Aeronautics and Astronautics in the Man-Vehicle Laboratory by using the NASA NE-2 moving base simulator and TR-48 analogue computer.

4.2 Experiment Equipment

As the test vehicle, a moving base two degree of freedom simulator was used in roll-pitch mode, figure 12. The simulator has a single cockpit and was equipped with an artifically loaded control stick and pedals. As a display unit, a cathode ray tube was used mounted in front of the pilot, figure 13. During test periods, the simulator was covered and for protection of the pilot, seat belts were used and the motion of the simulator was limited in roll to $\pm 45^\circ$ and in pitch, $\pm 15^\circ$.

4.2.1 Simulation Equipment

For simulation of a Caravelle aircraft, an analogue computer TR-48 was used, figure 16. This computer has 48 amplifiers, 5 multipliers, 2 squares and other auxiliary equipments. Fourteen of the
Figure 12  NASA NE-2 Moving-base two-degree-of-freedom Simulator in Roll Pitch Mode.

Figure 13  Inside view to the Simulator
amplifiers could be used as integrators. The operating voltage is ± 10V, whereas that of the simulator was ± 100V. That is why three attenuators were used to reduce the voltage of the control stick and pedal potentiometers to the computer level. Figure 14 shows the test equipment layout and the connections. In appendix B, "Simulation of an Aircraft," the analogue computer program is described and justified in detail. Due to the shortage of multipliers in the TR-48 analogue computer, no inertial cross-coupling terms were included in the simulation and the aerodynamic coefficients were kept constant, although they are functions of velocity and the angle of attack, as discussed in chapter III, 1.1. Simulation of a Caravelle with constant aerodynamic coefficients was also made earlier as shown in references 2 and 3.

The real time simulation of an aircraft done by the M.I.T. Electronic Systems Laboratory includes also nonlinear effects of the aerodynamic coefficients of a F-100A fighter aircraft. The computer used for that simulation was an experimental hybrid computer. Using that kind of computer was beyond the facilities in the Manned Vehicle Laboratory in the summer of 1965.

To be able to justify the behavior of the different control loops, the Root-Loci diagrams were made about lateral cases and shown in figures A.III to A.VI. One can expect that:

1. \((PF)^{\circ}_{A}[\text{\-\text{\circ}, \phi}]\) loop is stable having an oscillatory highly damped natural frequency and one integration and two decaying modes. If the roll loop is closed, the system stays stable as shown by the
Figure 1.4 Aircraft Simulation Layout and Connections
Figure 15  Test Pictures

Figure 16  Analogue Computer TR-48 connected to the Simulation
branches of the Root-Locus, and the better the feedback is, the better
the system will behave until the feedback-gain reaches unity, which
case is shown in figure A.III with closed loop poles.

2. \((FF)_{A}[\phi, \phi]\)-loop is also stable initially, as one would
expect, because the poles in all the transfer functions are the same.
If the \(\phi\)-loop is closed, meaning that the pilot would try to control
his roll by rudder, the system gets unstable as shown by branches of
the Root-Locus in figure A.IV. Although the pilot does not want to
close this loop, he uses rudder, which affects the same unstable,
oscillating mode as mentioned above.

3. \((FF)_{A}[\dot{\phi}, \phi]\)-loop is also initially stable, but closing the
loop effects unstable oscillatory mode. This mode can be detected
very easily in the experiment results. When a step disturbance was
initiated to the roll loop, the pilot was forced to counter-act with
ailerons, and that affected an unstable oscillation in the yaw loop.
Putting some lead to this system is the only way of getting the oscilla-
tion damped.

4. \((FF)_{A}[\phi, \psi]\)-loop is also initially stable, meaning that
the aircraft is stable. However, if the pilot desires to control the
heading only by the rudder, thus closing this loop, the heading will
get unstable having about equal time constants in the converging
stable and diverging unstable modes. To get the mode stable, the
pilot must have possibility to add lead to the loop. After adding
lead, the important control loop \((FF)_{A}[\phi, \psi]\) is very stable, having
roots only on the positive real axis, as one can see from pole-zero
configuration in the figure A.VI supposing only a single pole at the
origin.
4. 2.2 Display Equipment

The display to the pilot was simulated by using 3.5" diameter round CRT as shown in figure 13. It was fixed in front of the pilot. There was no other instruments in the cockpit, and the pilot was told to get all his information through this scope, representing "real world" picture. Generating the picture of the runway, horizon and flight-path marker is described and justified in detail in appendix C.

4. 2.2.1 Displayed Pictures

Three different pictures were generated and they were named as test pictures No. 1, 2 and 3.

The test picture No. 1, shown in figure 15a, contains two straight lines. This picture is a minimum "real world" picture of the runway. Instead of representing the boundary lines of the runway, the lines could be presented in different setup, for example, the center line and near end of the runway. The test picture No. 1, just as it is, is the most desired type by pilots.³

This picture contains information about altitude, and roll, pitch and yaw angles. How much of the available information the pilot can use is studied in the test set No. 1. The relative position and size of lines could be changed and so the real perspective picture of the runway could be projected to the pilot.

The test picture No. 2, figure 15b, contained the same basic picture of the runway as test picture No. 1, and an additional line
representing the horizon. Horizon of this picture was tilted 1:1 with "real world" picture in an "inside-out" manner, whereas the simulator in moving-base configuration "was tilted" only by an angle, representing the side force component. This test picture is supposed to contain more information about both the pitch and the roll angles and about the altitude and distance to the end of the runway. The additional information in the roll will be based on the horizon line. The increase in altitude information will be expected from the fact that altitude is proportional to the angle between horizon and the end of the runway. The detail description of the background of this information is given in appendix C. How much the pilot was able to obtain from the available information is studied in the test set No. 2.

The test picture No. 3, figure 15c, contained the same basic picture as the test picture No. 1 and an additional dot showing the spot where the velocity vector of the aircraft intersects the ground. This dot can also be called "flight path marker," because it shows the present flight path. The information given in this picture is similar to No. 1 as far as the runway is concerned, but the information in the flight path marker is entirely different. By using the flight path marker, the pilot is able effectively to close the $\gamma$-loop about flight path angle in his altitude control system, and as pointed out in chapter III, figure 10, the closure of $\gamma$-loop made the system very stable and easy to control.
How much the pilot is able to detect of the information in this picture is shown in chapter VI.

4.3 Recording Equipment

The test results were recorded by using paper recorders. Five channels were available for recording the test variables, and one was used to check the bias in the runway boundary line integrator in the display unit. In addition, two discrete markers were used to mark the disturbances.

The variables chosen for recording were:

<table>
<thead>
<tr>
<th>Variable</th>
<th>MAXIMUM VALUES</th>
</tr>
</thead>
<tbody>
<tr>
<td>x = distance to the end of the runway</td>
<td>18,000 feet</td>
</tr>
<tr>
<td>h = altitude</td>
<td>1,000 feet</td>
</tr>
<tr>
<td>γ = yaw angle</td>
<td>30 degrees</td>
</tr>
<tr>
<td>φ = roll angle</td>
<td>30 degrees</td>
</tr>
<tr>
<td>ϕ = roll rate</td>
<td>30 degrees/sec.</td>
</tr>
</tbody>
</table>
CHAPTER V

5.1 Tests

For finding out the pilot's possibilities and abilities in the control loop of the automatic landing aircraft in case of a malfunction, in some part of the whole system, a set of flight simulations was performed. Three pilots volunteered as test subjects. One of them was an experienced active airline test pilot, another an Air Force pilot with exceptional theoretical knowledge and the third one was a light plane flight instructor. The task given to them was to follow an automatic approach and in case of detecting a malfunction, to correct the motion and complete the landing. By eliminating the possibility of initiating a go-around command, the pilots were forced to fly the landing even when malfunctions were generated at low altitudes and so the full performance of the system could be recorded.

5.2 Reasons for Eliminating Go-around

As already mentioned before, the initiation of a go-around command is the only way the pilot can affect the automatic landings without head-up display of the runway to date. On the other hand, under the visual flight conditions, pilots show little reluctance in carrying automatic landings to touch down. This was the safety pilot's task during the more than 1,000 automatic landings made by the FAA with the blind landing experimental until (BLEU) autoland system. Under
full instrument conditions, with no external visual reference, however, the stress level on the flight crew is increased radically. With panel instruments, the pilot must now depend upon fragmented information from which he can reasonably infer his situation only for small departures in position, flight path and altitude from those for a normal on course trajectory along the beam. The panel displays are essentially null monitors and provide information for making small attitude and flight path changes when displacement from nominal positions are realized. Contract these severe restrictions with the pilot finds freedom in VFR conditions. Being dependent on panel instrument, the pilot, after find a difficulty, cannot stand the pressure from lack of more information and initiates the go-around command for any but slight departures from the normal flight path or position. For testing his real ability, the go-around was prohibited and each time a landing was completed.

5.3 Classification of Tests

The tests were carried out in three phases so that each display was regarded as one test set.

Each test set was divided into fixed-based and moving-based parts so that the ratio of display and feeling information could be determined. Lateral and longitudinal information was tested separately so that when testing the lateral information, the aircraft was flown down automatically along the flight path, and the pilot had only lateral control. When testing the longitudinal information, the pilot had full control over his plane and malfunctions were generated into either both or only one
of the channels. Malfunctions generated were step angular rate or ramp angular rate inputs to pitch or roll integrators, representing aerodynamic imbalance or false command from the autopilot.

After each set of test flights, pilots were interviewed and their opinions were recorded.

5.4 Pilot's Control

During tests, 461 test points were recorded and in addition to that, about 150 practicing and calibration approaches were flown. The main attempt was testing moving-base behavior and the fixed-base tests we performed for finding out the ratio between display and feeling information. Distribution of data points is shown in table 2.

<table>
<thead>
<tr>
<th>Test Set No. 1</th>
<th>Test Set No. 2</th>
<th>Test Set No. 3</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fixed-Base</td>
<td>Moving-Base</td>
<td>Fixed-Base</td>
</tr>
<tr>
<td>46</td>
<td>83</td>
<td>63</td>
</tr>
</tbody>
</table>

Total amount of Fixed-Base Test Points = 119
Total amount of Moving-Base Test Points = 345

TOTAL 464 Test Points
CHAPTER VI

6.1 Test Results

In this chapter, the test results are presented with interpretation to the background given in previous chapters. Presentation of any numerical numbers is rather difficult, not only because of the human pilots in the control loop, but also because of the definition for a steady state value of the variable is changing and so the exact settling time is difficult to estimate.

A definition for response time is that time taken for the system to recover from the primary effect of the disturbance. This, the roll response time is the time when the pilot has reached the full control of his plane in roll and is able to close the other loops effectively. An example of estimating the roll rate is given in figure 17. In estimating the response data, the author's pilot experience was used.

At first, general features of the test results are given and then each test set is discussed in more detail.

6.2 General Features of the Results

General features of the results presented here are based on the information in figures 17 - 2f.
6.2.1 Lateral Control

**Roll Case**

1. The roll response time to a step roll rate input increases with increasing roll rate.

2. The response time from the moving-base tests are considerably shorter than the fixed-base simulation values.

3. Fixed-base response results increase more rapidly than the moving-base values with increasing commanded roll rates.

4. Standard deviations of the moving base values are smaller than those of the fixed-base values.

5. The difference between the lateral control only and both the lateral and longitudinal controls is very little with small disturbances, but increases when disturbances get larger. The response time of the lateral control only is shorter than with both controls.

6. The altitude at which the disturbances were initiated did not have significant effect with roll response time.

**Yaw Case**

The first two statements in roll control case are the same, in yaw control case and in addition.

3. The yaw response time is more dependent on the disturbance amplitude than the roll response time.
4. The standard deviations of the yaw response time are considerably larger than those of roll response time.

5. The yaw response time of the lateral control only is shorter than in case of both controls.

6. In higher altitudes, the yaw response time was shorter than in low altitudes.

6.2.2 Longitudinal Control

1. None of the test pictures provided enough altitude information when altitude was above 300 - 400 feet.

2. In all the tests, the pilots tended to fly above the glide path.

3. Glide path control was poor and inaccurate when the test pictures No. 1 and No. 2 were used, but was considerably more stable and better when test picture No. 3 was used.

4. Altitude deviations from the desired glide path were larger with test pictures No. 1 and No. 2 than they were with test picture No. 3, both with and without disturbances.

5. The pilot's reaction time to disturbances was considerably longer for the first two test pictures than it was for the test picture No. 3.

6. All the pilots regarded the test picture No. 3 as best for longitudinal control.

6.3 Test Sets

The tests were carried out in three test sets, one for each test picture. A total of 464 test points were used in calculation of the
following figures. The moving-base values were considered more valuable and those tests were made in all the test sets. The fixed-base values were taken for evaluation of the pilot's feeling feedback and so the complete set of tests was not needed. The test picture No. 2 was regarded as the main fixed-base tested for yaw control.

6.3.1 Test Set No. 1

Test picture = two lines.

Total number of test points = 129
46 fixed-base
83 moving-base test points.

Test person = three subjects.

Figures 17 a and b show two representative approaches and landings in this test set. The pilot had both lateral and longitudinal controls.

In figure 17 altitude \( h \), yaw angle \( \gamma \), distance to the end of the runway \( k \), roll angle \( \phi \) and roll rate \( \dot{\phi} \) are plotted versus time. Around the optimum 3° glide path, there is a \( \pm 10\% \) deviation region down to 100 feet and after that a constant \( \pm 10 \) feet region. This is allowable altitude region during approach and landing. The vertical lines throughout the pictures represent the moment when the step disturbances are initiated. The dark line marked as 1 represents the length of the runway and distance \( x \), the distance to near end of the runway. When the dark broad line crosses \( x=0 \) line, then the aircraft is over the end of the runway and from altitude scale, one can see the altitude. When \( X \approx 800 \) feet, the pilot initiates a flare out
Figure 17 Two representative approaches and landings by using the test picture No. 1. In Part a is shown the behavior of the system to a step roll-rate disturbance, and in part b, to step pitch rate disturbances.
Figure 19 A representative approach and landing by using the test picture No. 2. Two roll-rate disturbances and three pitch-rate disturbances are generated during landing.
Figure 18  A representative approach and landing by using the test picture No. 2. Two roll-rate disturbances are generated.
Figure 20. Medium roll-response time to a step roll-rate disturbance input versus the magnitude of the roll-rate disturbance based on both fixed-base and moving-base tests for the test picture No. 1. The bars represent the range of 2σ distribution.
Figure 21 Medium yaw-response time to a step roll-rate disturbance input versus the magnitude of the roll-rate disturbance based on the moving-base tests for test picture No. 1. The bars represent the range of 25 distribution.
during which the speed of the plane decreases and rate of descent is decreased. In this simulation $h = $ altitude rate is proportional to glide path angle and during flare out, the altitude tends to increase instead of decreasing slowly. For this reason, an altitude about 30 feet is regarded desirable when $x=0$. The point $x=0$ in this simulation represents a point on the runway, about 800 feet from the end of the runway.

Estimating roll response time from figure 17a would be as following: After about 32 seconds of approach, a step roll disturbance is initiated and shown with an arrow. The roll rate and angle, which are the simulator angular rate and roll angle start to build up. The pilot counteracts to correct the roll angle. The roll angle decreases and one can see that he over did, and the roll rate starts to oscillate. After the pilot gets the wings level, the yaw rate goes to zero and now he closes the yaw loop to correct the yaw angle back to zero. To this point, about 7 seconds from the disturbance, the pilot is mainly controlling the roll angle and so the roll response time is 7 seconds.

Estimation of the yaw response time is similar to roll response time. In the longitudinal control case, the delay time is estimated to the point, where the flight path is again parallel to the optimum flight path, and also the altitude deviation is calculated to that point.

6. 3.1. 1  Lateral Control
6. 3.1. 1.1 Roll Control

In this test set, both fixed and moving-base tests were carried in roll control.

Fixed-base values as shown in figure 20a, show that for a moderate 6-8°/sec. roll rate disturbance, the response time is about 12-14 seconds. In figure 20, the roll response time is plotted versus disturbance roll rate on the ordinate axis. The reason for long response time is that the damping is fairly poor and overshooting is strong. The initial delay time to recognize the disturbance is about 1 - 1.5 seconds. One can also notice that if the pilot has the lateral control only, he can concentrate better to control the roll and when disturbances get large, he can do better than in case of both lateral and longitudinal controls.

The moving-base values as shown in figure 20b show that the roll response time to a moderate 6-8°/sec. roll rate disturbance is decreased to 6-7 seconds, which is about half of the corresponding fixed-base value. The reason for the faster response is the pilot's feeling information. He uses it in the feedback loop in addition to display feedback and the damping in the moving-base tests is much larger and over shooting as a result much less than in fixed-base configuration. The initial delay is also shorter and the pilot's direction decision for correction was never wrong.

6. 3.1. 1.2 Yaw Control

The yaw response time to a moderate 6-8°/sec. roll rate input is
surprisingly long, as seen from figure 21, about 12-14 seconds, even in moving-base configuration. The reason why the time tends to get so long is that the pilot closes the roll loop at first and levels the wings before trying to close the yaw loop. For getting the increased yaw angle back to its initial value, the pilot must bank his aircraft to the opposite side and determination of the amount of counter-banking can be made only by try-and-find method.

6. 3.1. 2 Longitudinal Case

Altitude and glide path control by using the test picture No. 1 is rather unstable even in cases when no disturbances existed in the pitch loop. In table 3 is presented the relative test results of the test pictures, and the No. 1 seems to give the least information to the pilot. The pilot's opinion was that the landing by using this picture was very difficult to perform because of the lack of references.

6. 3.2 Test Set No. 2

Test picture = three lines.

Total number of test points = 204
   63 fixed-base
   141 moving-base test points.

Test person = three subjects.

Figure 18 shows a representative approach and landing in this test set. The pilot has both the lateral and longitudinal controls. Even though no disturbance signal was commanded, the longitudinal
Figure 22: Medium roll response time to a step roll-rate disturbance input versus the magnitude of the roll-rate disturbance based on both fixed and moving-base tests for the test picture No. 1. The bars represent the range of 20 distribution.
control can be seen to be relatively inaccurate. The nomenclature of this picture is the same as given in figure 17.

6. 3.2. 1 Lateral Control

Test set No. 2 was considered as the most important basic test set and so in the lateral case, both the fixed and moving-base values were carried out completely.

6. 3.2. 1.1 Roll Control

In this test set both fixed and moving-base tests were carried out in roll control and in addition the moving-base part was divided into two subparts, namely lateral only and both lateral and longitudinal. The fixed-base values were carried out in the case of both controls.

The fixed-base values, shown in figure 22a, show that the roll response time is considerably shorter than in case of test picture No. 1. For a moderate 6-8°/sec. step roll input, the medium roll response is about 9-11 seconds, which shows clearly the importance of good reference in fixed-base simulation.

The moving-base values, shown in figure 22b, show that the medium response time to a moderate 6-8°/sec. roll rate disturbance input is about 6 seconds in both the lateral control case and in case of both controls. The difference between the two control cases in figure 22b shows up slightly when the disturbances get larger, because the correction needs more the pilot's attention.
6.3.2.1.1 Yaw Control

The yaw control of this simulated aircraft is rather difficult and in case of fixed-base control, it was rather often unstable. This means that the system behaves as predicted on the base of figure A.V. The medium yaw response time to a roll step input was 20 seconds, as seen from figure 23a, which compared with moving-base values is very large. There are three possible solutions to the problem.

1. The test picture allows the pilot to put very little lead to the slightly damped system, and when he tries to use high gain when closing the loop, the system gets more and more unstable as predicted in figure A.V.

2. The pilots were not used to handle this kind of problem.

3. The simulation was wrong.

The combination of all these three is probably the real solution, while the effect of the statement No. 3 is due to neglecting the inertial cross-coupling terms in yaw equation, as shown in reference 4.

The moving-base data is, however, much more promising and correct looking values and as seen from figure 23b, the medium yaw response time to a moderate $6-8^0$/sec. roll rate step input is about 10-12 seconds, and in case of lateral control only, even less. Trying to control both the lateral and longitudinal terms seems to be more difficult when the amplitude of disturbances increase and medium yaw response time gets longer than in case of lateral control only.
Figure 23 Medium yaw-response time to a step roll-rate disturbance input versus the magnitude of the roll-rate disturbance based on both fixed and moving-base tests for the test picture No. 2. The bars represent the range of 2σ distribution.
6.3.2.2 Longitudinal Control

The longitudinal control of the "aircraft" with test picture No. 2 did not differ very much from the case having the test picture No. 1, as can be seen from table 3. Probably the additional altitude information from the relative position of the horizon and the runway made the control a little easier, especially in low altitudes. The effect from this was the decreasing crash ratio and decreasing altitude deviation during longitudinal recoveries.

6.3.3 Test Set No. 3

Test picture = two lines and a dot.

Total number of test points = 129
  10 fixed-base
  119 moving-base test points.

Test person = two pilots carried through the total test set and the third, only partly.

Figure 19 shows a representative approach and landing in this test set. The pilot has both lateral and longitudinal controls and both types of disturbances were generated during the same test. The nomenclature in this figure is same as in figure 17.

6.3.3.1 Lateral Control

6.3.3.1.1 Roll Control

The roll control with this test picture is not expected to change very much from the values in the previous test sets. It can be also
seen to be true from figure 24, where the roll response values are plotted from all the test sets.

6.3.3.1.2 Yaw Control

It was expected that the yaw response time would be shorter when the test picture No. 3 was used compared with two other cases. The reason was that the pilot could actually visualize changes in his yaw angle and also in his yaw rate because the dot shows the direction of his flight path. So the pilot has possibilities to close the yaw loop with enough lead angle. The results do not show that as can be seen from figure 25 where the values from all yaw loop moving-base tests are collected. The reasons for this unexpected behavior can be:

1. Due to the shortage of the multipliers in the analogue computer, the coordinate transformation from earth to body coordinates was not made to the dot and so it gave wrong information when the aircraft was banked.

2. Both the test picture and the situation was new for the pilots, and so they were not able to use the information effectively.

The real effectiveness of this test picture could be tested only in flight tests, and the author is convinced that the yaw response time is shorter than the values from test pictures No. 1 and 2.

6.3.3.2 Longitudinal Control

The glide path and altitude control by using the test picture No. 3 is improved considerably. As seen in table 3, the glide path control at
Figure 24: Comparison of the moving-base medium roll-response times to step roll-rate disturbance inputs versus the magnitude of the roll-rate disturbance for all the test pictures. The bars represent the range of 2s distribution.
Figure 25: Comparison of the moving-base, medium yaw-response times to step roll-rate disturbance inputs versus magnitude of the roll-rate disturbance for all test pictures. The bars represent the range of 26 distribution.
low altitudes is very accurate and only one probable crash was
recorded. Also that, in real flight case had been a landing on the
runway, because the aiming point for landing is usually about 900-
1,000 feet from the end of the runway, as pointed out earlier in
6.3.1. The delay time to a moderate 2.5°/sec. pitch rate was
recorded in 20 cases giving an average of 6.5 seconds, which is less
than a half of the values in previous tests. Also, the altitude
development from the optimum glide path value during the recovery was
less than half of the previous values and is in several cases within
the allowable altitude region, and actually should not be considered
as altitude deviation. The reason for considering it was an altitude
development, is that it is needed in estimations for the minimum recovery
altitude with manual control in case of malfunctions in the automatic
system. As shown in figure 19, the recovery from a pitch down step
input at altitude of about 330 feet was completed in about 7 seconds,
and the pilot reported after the flight that he allowed the flight
path to fall lower on purpose and did not want to make the recovery
as fast as he could have made it, because he was above the glide path.
In case of any one of the other test pictures, the pitch down command
in that altitude had affected a probable crash. The flight path marker
control allows lower approach speeds, because the effect of front-side
or back-side of the lift drag curve does not have the same catastrophic
effect as in case of display with no angle of attack information. From
the flight path marker picture, the pilot sees immediately the effect
of his control movements and finds out when he must shift from stick
control to thrust control in controlling the altitude, as one must do in the case of back-side of the lift drag curve.

Although the test picture No. 3 allows good glide path control, it still lacks primary altitude information like all other test pictures.

6.4 Interactions

From the recorded test results, one cannot find very straight correlation between roll rate inputs and pitch output and pitch rate input and roll output. They both are rather separated, and the only case where some correlation was noticed was in case if the control stick was displaced very much from center position. This effect was not studied more detail in this work.

6.5 Summary

These are the test results with interpretation from a analogue computer study, where the analogue model was a Caravelle aircraft in landing condition. Several approximations were however made during the simulation, and the effects of these approximations to the real test parameters could be obtained only if a set of similar flight tests results were available.

The conclusion drawn from these results and from the discussion in previous chapters is represented in chapter VII.
TABLE 3
Relative test results of the test pictures in longitudinal control.

<table>
<thead>
<tr>
<th></th>
<th>No.1</th>
<th>%</th>
<th>No.2</th>
<th>%</th>
<th>No.3</th>
</tr>
</thead>
<tbody>
<tr>
<td>Total number of landings</td>
<td>26</td>
<td></td>
<td>41</td>
<td></td>
<td>33</td>
</tr>
<tr>
<td>Flight path at any time</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>below-10% deviated glide path more than 50 ft. (times)</td>
<td>11</td>
<td>42.4</td>
<td>20</td>
<td>48.5</td>
<td>6</td>
</tr>
<tr>
<td>Flight path at any time</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>below-10% glide path more than 100 ft. (times)</td>
<td>8</td>
<td>30.7</td>
<td>6</td>
<td>14.7</td>
<td>4</td>
</tr>
<tr>
<td>Flight path above ±10</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>deviated glide path more than 100 ft. (times)</td>
<td>20</td>
<td>77.0</td>
<td>24</td>
<td>58.2</td>
<td>12</td>
</tr>
<tr>
<td>Probable crash</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>10</td>
<td>38.5</td>
<td>9</td>
<td>25.7</td>
<td>1</td>
</tr>
<tr>
<td>Definite crash</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>7</td>
<td>27.0</td>
<td>6</td>
<td>18.3</td>
<td>0</td>
</tr>
<tr>
<td>Delay time to a moderate</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>2.5/sec. pitch rate (sec.)</td>
<td>16</td>
<td></td>
<td>15</td>
<td></td>
<td>6-7</td>
</tr>
<tr>
<td>Altitude deviation from</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>optimum glide path value</td>
<td>160</td>
<td></td>
<td>120</td>
<td></td>
<td>50</td>
</tr>
<tr>
<td>during recovery (feet)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
CHAPTER VII

Conclusions and Recommendations for Further Work

In this chapter, the conclusion and recommendations for further work based on the previous chapters are given.

7.1 Conclusion

In order to get full effect out of the automatic landing system, the pilot has to be taken effectively to the control loop. He is the most adaptive part in case of malfunction in some part of the automatic control system and must be able to save the aircraft either by initiating a go-around command or completing the landing manually. The go-around is only partial solution in case of malfunction in the automatic control system, and the pilot must be able to make the safe landing manually. He can perform a safe manual landing, although the approach speed was decreased for automatic landing purposes as shown in chapter II. If the visual display of the runway and the most valuable instrumented values to the pilot are displayed, the following conclusions can be drawn from the test results in chapter VI for the display of the runway and for the glide path control:

1. The roll information needed for fast recovery from moderate roll-rate disturbances can be obtained from the minimum picture of the runway.
2. Because the pilot closes the roll loop before the yaw loop, it is important that the roll response time is as short as possible. That is one reason why the image of the horizon is necessary in the display.

3. The yaw control loop of an aircraft is usually lightly damped and tends to become unstable without rate feedback. The roll angle is a measure of yaw rate and so the horizon is necessary to allow the pilot to have as much information about yaw rate as possible.

4. The flight path marker gives the stability and accuracy to the glide path control that is so necessary for safe approaches and landings.

5. To improve the effect and the accuracy of the flight path marker, the runway picture should be provided with a commanding dot to which the pilot should match the flight path marker during approaches.

6. The altitude information must be based on other sources than explained in this work, because the altitude information from the display picture is very poor in higher altitudes than 300 - 400 feet.

By using the proper display to the pilot, he could be able to make recovery from the cases if the malfunction in the pitch loop occurs higher than 90 - 100 feet during final approach, and in case of malfunction in the roll loop, if the altitude is above 100 - 120 feet. The lateral case is higher due to long yaw settling time.
7.2 Recommendations for Further Work

The results in chapter VI show very clearly that the pilot's feeling feedback carries very important part from the total information flow. That is why the values from fixed-base simulations may be rather inaccurate if the input values exceed the basic disturbance level and moving-base simulations are needed.

As pointed out in chapter VI, the results should be related more tightly to real flight conditions, and the relative value of the response times should be checked by flight test; especially this is the case in yaw response.
APPENDIX A

Derivation of Aircraft Equations of Motion

Aircraft equations of six degree of freedom motions can be divided into two groups, lateral and longitudinal cases.

Lateral Case

Lateral equations of the motion can be derived by combining roll, yaw and side force equations. Writing the equations of motion in the following form A.1 - A.3 will give an advantage that a minimum number of variables are explicitly dependent on velocity:

Roll

\[
\frac{I_{xx}}{qSb} \ddot{\phi} = C_{\phi} \beta + C_{\ell_{\alpha}} \dot{\alpha} + C_{\ell_{\delta}} \dot{\delta} + \\
\frac{b}{2v} C_{\ell_{\gamma}} \dot{\gamma} + \frac{b}{2v} C_{\ell_{\phi}} \dot{\phi}
\]

(A.1)

Yaw

\[
\frac{I_{yy}}{qSb} \ddot{\gamma} = C_{\gamma_{\beta}} \beta + C_{n_{\alpha}} \dot{\alpha} + C_{n_{\delta}} \dot{\delta} + \\
\frac{b}{2v} C_{n_{\gamma}} \dot{\gamma} + \frac{b}{2v} C_{n_{\phi}} \dot{\phi}
\]

(A.2)
Side Force

\[ \frac{m v}{g s} \dot{\phi} = c_y \beta + c_y \delta_z \cdot \delta_r - \frac{m v}{g s} \dot{\gamma} + \frac{m g}{g s} \dot{\phi} \]

(A.3)

The equations A.1 - A.3 can be rewritten by using new nomenclature, where coefficients in the roll-equation are named as P, in the yaw-equation, as N and in the side force-equation, as Y. Each one is particularized by subscript 6, 11.

Roll

\[ P_{\phi} \ddot{\phi} = P_{\beta} \beta + P_{\delta_z} \cdot \delta_A + P_{\delta_r} \cdot \delta_r + \]

\[ + P_{\gamma} \dot{\gamma} + P_{\dot{\phi}} \cdot \dot{\phi} \]

(A.4)

Yaw

\[ N_{\phi} \ddot{\psi} = N_{\beta} \beta + N_{\delta_z} \cdot \delta_A + N_{\delta_r} \cdot \delta_r + \]

\[ + N_{\gamma} \dot{\gamma} + N_{\dot{\phi}} \cdot \dot{\phi} \]

(A.5)
Side Force

\[ Y_{\rho} \cdot \phi = Y_{\rho} \cdot \phi + \gamma_{\rho} \cdot \delta_{\rho} - Y_{\psi} \cdot \psi + Y_{\phi} \cdot \phi \]

(A.6)

After taking Laplace transform and solving these equations simultaneously with respect to the variables \( \Phi(s) \) and \( \psi(s) \), we will get

\[ A \frac{\Phi(s)}{\delta_{\rho}(s)} = B \frac{Y(s)}{\delta_{\rho}(s)} + C \frac{\delta_{\phi}(s)}{\delta_{\rho}(s)} + D \]

(A.7)

\[ E \frac{Y(s)}{\delta_{\rho}(s)} = F \frac{\Phi(s)}{\delta_{\rho}(s)} + G \frac{\delta_{\phi}(s)}{\delta_{\rho}(s)} + H \]

(A.8)

where

\[ A = \left[ P_{\phi} s^2 - P_{\phi} s \right] \left[ Y_{\rho} s - Y_{\rho} \right] - P_{\rho} \cdot Y_{\phi} \]

\[ B = Y_{\rho} P_{\psi} s^2 - \left( Y_{\rho} P_{\psi} + P_{\rho} Y_{\psi} \right) s \]

\[ C = Y_{\rho} P_{\delta_{\phi}} s - Y_{\rho} \cdot P_{\delta_{\phi}} \]

\[ D = Y_{\rho} P_{\delta_{\rho}} s + P_{\rho} Y_{\delta_{\rho}} - Y_{\rho} \cdot P_{\delta_{\rho}} \]

\[ E = \left[ N_{\psi} s^2 - N_{\psi} s \right] \left[ Y_{\rho} s - Y_{\rho} \right] + N_{\rho} \cdot Y_{\psi} s \]

\[ F = N_{\phi} Y_{\phi} s^2 - Y_{\rho} \cdot N_{\phi} s + N_{\rho} \cdot Y_{\phi} \]

\[ G = (Y_{\rho} s - Y_{\rho}) N_{\delta_{\phi}} \]

\[ H = N_{\delta_{\rho}} (Y_{\rho} s - Y_{\rho}) + N_{\rho} \cdot Y_{\delta_{\rho}} \]
\( \Phi(s) \) and \( \Psi(s) \) can be solved from the equation A.7 and A.8 and we will get:

\[
\Phi(s) = \left( \frac{EC + BG}{AE - BF} \right) \sigma_A(s) + \left( \frac{HB + DE}{AE - BF} \right) \sigma_R(s)
\]

(A.9)

\[
\Psi(s) = \left( \frac{AG + CF}{AE - BF} \right) \sigma_A(s) + \left( \frac{AH + DF}{AE - BF} \right) \sigma_R(s)
\]

(A.10)

The equations A.9 and A.10 can be interpreted in the block diagram form as shown in the figure A.1. One notes that the denominators in each separate control branch are similar and only numerator parts vary.\(^{10}\) This means that in the transfer functions, the open-loop poles are the same for all, and only the open-loop zeros and open-loop gains vary for each particular branch.

**Denominators**

Solution to the denominator equations of A.9 and A.10 gives the open-loop poles of the transfer function and at the same time, natural frequencies of the system. The characteristic equation of the denominator is of sixth order, and the root of that equation was found by using Root-Locus techniques.

For a Caravelle aircraft in the landing condition, the denominator characteristic equation is by using the values in table A.1.
Fig. A I. Aircraft Lateral Performance Function Block Diagram.
\begin{align*}
0.4928^6 + 0.9703^5 + 0.4398^4 + 0.6393^3 + 0.07863^2 + 0.0075 &= 0 \\
\text{(A.11)}
\end{align*}

The same equation for a Boeing 707 airplane in landing condition is:

\begin{align*}
6.2836^6 + 6.6385^5 + 3.7234^4 + 3.7933^3 + 0.5582 &= 0 \\
\text{(A.12)}
\end{align*}

and for cruise condition:

\begin{align*}
0.6186^6 + 4.7085^5 + 0.7784^4 + 0.6383^3 + 0.0633^2 &= 0 \\
\text{(A.13)}
\end{align*}

The equations A.10-A.13 were calculated by using all the coefficients given in the equations A.4-A.6. An interesting feature is that for a different type of an aircraft and different speed of flight, the relative value of the coefficient terms were different. That is why neglecting the coefficients and data is rather difficult. Roots of the equations A.10-A.12 are shown in the figure A.II.

**Numerators**

The numerators of the transfer functions given in the figure A I have been solved for the Caravelle aircraft and the results are given in the equations A.14-A.17 and the respective Root-Locus diagrams in the figures A.II to A.VI.
Fig. A II. Roots of the Characteristic Equations for Caravelle in the Landing Condition and for Boeing 707-320B Both in Landing and Cruise Conditions.
\[ \frac{(PF)_A[\delta_A, \phi]}{s(s+0.133)(s+1.05)(s+0.303 \pm 0.77j)} \]

Fig. A III. Root Locus Diagram of \((PF)_A[\delta_A, \phi]\) for a Caravelle.
\[
(PF)_{A}^A_{\Delta \in \Theta} = \frac{0.190(s-2.65) (s+0.025) (s+0.80)}{s (s+1.85) (s+0.133) (s+0.005+0.77j)(1+0.005-0.77j)}
\]

Fig. A IV. Root Locus Diagram of \((PF)_{A}^A_{\Delta \in \Theta}\) for a Caravelle.
\[
\left( PF \right)_{A \left[ \delta_{A}, \psi \right]} = \frac{-0.0275 \left( s + 13.18 \right) \left( s + 0.063 \right) \left( s + 0.67 \right) \left( s - 0.519 \right)}{s^3 \left( s + 1.85 \right) \left( s + 0.133 \right) \left( s + 0.605 \pm 0.77 j \right)}
\]

Fig. A.5: Root Locus Diagram of \( \left( PF \right)_{A \left[ \delta_{A}, \psi \right]} \) for a Caravelle.
\[ (PF)_A[d_2, \psi] = \frac{-0.34(s+1.70)}{s^2(s+1.85)} \]

**Fig. A VI.** Root Locus Diagram of \((PF)_A[d_2, \psi]\) for a Caravelle.
\[
(PF)_A[\sigma_\alpha, \phi] = \frac{1.84(s+0.083)(s+0.045\pm 0.68)^2}{s(s+0.133)(s+0.003\pm 0.72)^2(s+1.85)}
\]  

(A.14)

\[
(PF)_A[\sigma_R, \phi] = \frac{0.190(s-2.65)(s+0.025)(s+0.80)}{s(s+1.85)(s+0.133)(s+0.003\pm 0.72)^2}
\]  

(A.15)

\[
(PF)_A[\sigma_\alpha, \psi] = \frac{-0.0228(s+13.10)(s+0.063)(s+0.67)(s-0.519)}{s^2(s+1.85)(s+0.133)(s+0.003\pm 0.72)^2}
\]  

(A.16)

\[
(PF)_A[\sigma_R, \psi] = \frac{-0.34(s+1.20)}{s^2(s+1.85)}
\]  

(A.17)

**Longitudinal Case**

In order to solve the longitudinal motion of an aircraft, the pitch and Z-force equations are needed.
Pitch

\[ \frac{I_{yy}}{q^*S^\circ} \cdot \ddot{\Theta} = C_{m\alpha} \cdot \dot{\alpha} + C_{m\Phi} \cdot \Theta + \frac{c}{2V} C_{m\phi} \cdot \Theta + \frac{c}{2V} C_{m\Psi} \cdot \dot{\alpha} \]

(A.18)

Z-Force

\[ \frac{mV}{q^*S} \cdot \dot{\alpha} = \frac{mV}{q^*S} \cdot \dot{\Theta} + C_{L\alpha} \cdot \dot{\alpha} + C_{L\Phi} \cdot \Phi + \frac{c}{2V} C_{L\phi} \cdot \dot{\alpha} \]

(A.19)

By using the same type of nomenclature as in the equations A.4-A.6, but now using M for the pitch and Z for the Z-Force equations, we will get:

Pitch

\[ M_{\Theta} \cdot \ddot{\Theta} = M_{\alpha} \cdot \dot{\alpha} + M_{\Phi} \cdot \dot{\Phi} + M_{\phi} \cdot \ddot{\Theta} + M_{\Psi} \cdot \dot{\Psi} \]

(A.20)

Z-Force

\[ Z_{\alpha} \cdot \dot{\alpha} = Z_{\alpha} \cdot \dot{\alpha} + Z_{\Theta} \cdot \dot{\Theta} + Z_{\Phi} \cdot \dot{\Phi} \]

(A.21)
Rearranging and transforming these equations, one is able to solve:

\[
\tan \frac{\alpha(s)}{\delta E(s)} = \frac{\frac{Z_{\dot{e}}}{s - Z_{\alpha}} [N_{\phi}s - N_{\theta}s] + \frac{N_{\phi}s - N_{\theta}s}{s - Z_{\alpha}} Z_{\dot{E}} \cdot s - \frac{N_{\theta}s + N_{\alpha}}{s - Z_{\alpha}} Z_{\dot{\phi}} \cdot s}{\frac{Z_{\dot{e}}}{s - Z_{\alpha}} [N_{\theta}s^2 - N_{\phi}s] - \frac{N_{\phi}s + N_{\alpha}}{s - Z_{\alpha}} Z_{\dot{\phi}} \cdot s}
\]

(A.22)

\[
\tan \frac{\alpha(s)}{\delta E(s)} = \frac{\frac{Z_{\dot{e}}}{s - Z_{\alpha}} [N_{\theta}s^2 - N_{\phi}s] + \frac{N_{\phi}s - N_{\theta}s}{s - Z_{\alpha}} Z_{\dot{E}} \cdot s - \frac{N_{\theta}s + N_{\alpha}}{s - Z_{\alpha}} Z_{\dot{\phi}} \cdot s}{\frac{Z_{\dot{e}}}{s - Z_{\alpha}} [N_{\theta}s^2 - N_{\phi}s] - \frac{N_{\phi}s + N_{\alpha}}{s - Z_{\alpha}} Z_{\dot{\phi}} \cdot s}
\]

(A.23)

Angle of glide slope \( \gamma \) can be written:

\[
\gamma = \Theta - \alpha
\]

(A.24)

and so

\[
\frac{\gamma(s)}{\Theta(s)} = 1 - \frac{\alpha(s)/\delta E(s)}{\Theta(s)/\delta E(s)}
\]

(A.25)

By using the values for a Caravelle given in the table AII, equations A.22 and A.25 get the form:
\[
\frac{\Theta(s)}{\delta_E(s)} = \frac{1.07 \left( s + 0.825 \right)}{s \left( s^2 + 1.815 + 1.04 \right)}
\]

\[\text{(A.26)}\]

\[
\frac{\varphi(s)}{\Theta(s)} = \frac{-0.053 \left( s^2 + 0.267 s - 14.5 \right)}{(s + 0.825)}
\]

\[\text{(A.27)}\]
TABLE A.1

Coefficients for Lateral Equations of Motion for a Caravelle and Boeing 707 in Landing Condition and a Boeing 707 in Cruise Condition.

<table>
<thead>
<tr>
<th>VARIABLE</th>
<th>CARAVELLE LANDING</th>
<th>707 LANDING</th>
<th>707 CRUISE</th>
</tr>
</thead>
<tbody>
<tr>
<td>$P_{\phi}$ = $I_{xx}/2Sb$</td>
<td>+0.0795</td>
<td>0.1848</td>
<td>0.0546</td>
</tr>
<tr>
<td>$P_{\phi}$ = $b/2\nu\cdot C_l\rho$</td>
<td>-0.1285</td>
<td>-0.1209</td>
<td>-0.0360</td>
</tr>
<tr>
<td>$P_{\gamma}$ = $b/2\nu\cdot C_{l\gamma}$</td>
<td>0.0885</td>
<td>0.1112</td>
<td>0.0129</td>
</tr>
<tr>
<td>$P_{\delta_a}$ = $C_{l\delta_a}$</td>
<td>0.015</td>
<td>0.01576</td>
<td>0.0229</td>
</tr>
<tr>
<td>$P_{\delta_A}$ = $C_{l\delta_A}$</td>
<td>0.145</td>
<td>0.2213</td>
<td>0.103</td>
</tr>
<tr>
<td>$P_{\alpha}$ = $C_{l\alpha}$</td>
<td>-0.117</td>
<td>-0.189</td>
<td>-0.170</td>
</tr>
<tr>
<td>$N_{\gamma}$ = $I_{zz}/2Sb$</td>
<td>0.184</td>
<td>0.3665</td>
<td>0.1092</td>
</tr>
<tr>
<td>$N_{\phi}$ = $b/2\nu\cdot C_{n\rho}$</td>
<td>-0.0356</td>
<td>-0.0696</td>
<td>-0.0152</td>
</tr>
<tr>
<td>$N_{\phi}$ = $b/2\nu\cdot C_{n\rho}$</td>
<td>-0.0320</td>
<td>-0.0499</td>
<td>-0.00402</td>
</tr>
<tr>
<td>$N_{\beta}$ = $C_{n\beta}$</td>
<td>0.080</td>
<td>0.1318</td>
<td>0.1135</td>
</tr>
<tr>
<td>$N_{\delta_R}$ = $C_{n\delta_R}$</td>
<td>-0.062</td>
<td>-0.086</td>
<td>-0.097</td>
</tr>
<tr>
<td>$N_{\delta_A}$ = $C_{n\delta_A}$</td>
<td>-0.005</td>
<td>+0.0039</td>
<td>+0.0039</td>
</tr>
<tr>
<td>$Y_{\beta}$ = $m\nu/\delta^2$</td>
<td>5.80</td>
<td>9.64</td>
<td>10.114</td>
</tr>
<tr>
<td>$Y_{\alpha}$ = $C_{\gamma\beta}$</td>
<td>-0.50</td>
<td>-1.035</td>
<td>-0.914</td>
</tr>
<tr>
<td>VARIABLE</td>
<td>CARAVELLE LANDING</td>
<td>707 LANDING</td>
<td>707 CRUISE</td>
</tr>
<tr>
<td>------------</td>
<td>-------------------</td>
<td>-------------</td>
<td>-----------</td>
</tr>
<tr>
<td>$Y_{\delta_e} = C_{Y_{\delta_e}}$</td>
<td>0.150</td>
<td>0.1897</td>
<td>0.226</td>
</tr>
<tr>
<td>$Y_{\psi} = \frac{m v}{\phi \delta}$</td>
<td>5.80</td>
<td>9.538</td>
<td>10.076</td>
</tr>
<tr>
<td>$Y_{\phi} = \frac{\theta m}{\phi \delta}$</td>
<td>0.954</td>
<td>1.32</td>
<td>0.416</td>
</tr>
</tbody>
</table>
**TABLE A.II**

Coefficients for Longitudinal Equations of Motion for a Caravelle in Landing Condition.

<table>
<thead>
<tr>
<th>Term</th>
<th>Expression</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>$M_\theta$</td>
<td>$\frac{I_{yy}}{q \cdot s' \cdot c}$</td>
<td>0.833</td>
</tr>
<tr>
<td>$M_\dot{\theta}$</td>
<td>$\frac{c}{2v} \cdot C_m \cdot q$</td>
<td>-0.44</td>
</tr>
<tr>
<td>$M_{\delta_e}$</td>
<td>$C_m \delta_e$</td>
<td>+0.90</td>
</tr>
<tr>
<td>$M_\alpha$</td>
<td>$\frac{c}{2v} \cdot C_m \cdot \alpha$</td>
<td>-0.139</td>
</tr>
<tr>
<td>$M_\alpha$</td>
<td>$C_m \alpha$</td>
<td>-0.49</td>
</tr>
<tr>
<td>$Z_{\delta_e}$</td>
<td>$C_{Z_{\delta_e}}$</td>
<td>+0.35</td>
</tr>
<tr>
<td>$Z_\alpha$</td>
<td>$C_{Z_{\alpha}}$</td>
<td>+4.9</td>
</tr>
<tr>
<td>$Z_\dot{\alpha}$</td>
<td>$\frac{m \cdot V}{q \cdot s'}$</td>
<td>+5.80</td>
</tr>
<tr>
<td>$Z_\theta$</td>
<td>$\frac{m \cdot V}{q \cdot s'}$</td>
<td>+5.80</td>
</tr>
</tbody>
</table>
APPENDIX B

Simulation of an Aircraft

An essential part of this study was simulation of an aircraft by means of an analogue computer and use of the resolved angles to give the pilot a picture of the runway and the motion of the airplane. With this information, he can judge if his glidepath is correct. At the M.I.T. Electronic Systems Laboratory, several thorough and valuable studies have been made to evaluate and weigh the necessary parts of the equations for simulation of an aircraft. The following equations are based on this work:

Pitch

\[ \ddot{\Theta} + \left( \frac{I_{xx} - I_{zz}}{I_{yy}} \right) \dot{\phi} \dot{\psi} + \frac{I_{xz}}{I_{yy}} \dot{\phi}^2 = \frac{g v^2 S' C_m (d, \Lambda)}{2 I_{yy}} + \]

\[ + \frac{g v^2 S' c^2}{4 I_{yy}} C_{m_y} \dot{\Theta} + \frac{g v^2 S' c^2}{4 I_{yy}} C_{m_z} (d) \dot{d} \]

\[ + \frac{g v^2 S' c}{2 I_{yy}} C_{m_d} (Ma) \sigma_E \]

(B.1)

Roll

\[ \ddot{\phi} = \frac{g v^2 S b}{2 I_{xx}} C_{l_{\rho}} (d) \cdot \beta + \frac{g v^2 S b}{2 I_{xx}} C_{l_{d_{\phi}}} \sigma_A (Ma) + \]

\[ + \frac{g v^2 S b}{2 I_{xx}} C_{l_{d_{\phi}}} \sigma_E + \frac{g v^2 S b}{2 I_{xx}} C_{l_{\rho}} \dot{\phi} + \frac{g v^2 S b}{2 I_{xx}} C_{l_{\rho}} \cdot \dot{\psi} \]

(B.2)
Yaw

\[ \ddot{\psi} + \left( \frac{I_{yy} - I_{xx}}{I_{zz}} \right) \dot{\phi} \dot{\theta} = \frac{g v^2 S b}{2 I_{zz}} C_{n_r} \beta + \frac{g v^2 S b}{2 I_{zz}} C_{n_d} \cdot \dot{\sigma}_x \]

\[ + \frac{g v S b^2}{4 I_{zz}} C_{n_r} \cdot \ddot{\psi} + \frac{g v S b^2}{4 I_{zz}} C_{n_d} \cdot \dot{\phi} \]

(B.3)

Y-Force

\[ \nabla \left[ \dot{\beta} + \dot{\psi} - \phi \cdot g \right] = g \cos \theta_w \cdot \sin \phi_w - \frac{I}{m} \cos (\alpha + \alpha_t) \sin \beta \]

\[ + \frac{g v^2 S}{2 m} C_{w_d} \]

(B.4)

Z-Force

\[ \nabla \left[ (\dot{\alpha} - \theta) + \dot{\phi} \cdot \beta \right] = g \cos \theta_w \cos \phi_w + \frac{g v^2 S}{2 m} C_z (\alpha, \alpha_t) \]

\[ + \frac{g v^2 S}{2 m} C_{z \sigma_e} \cdot \dot{\sigma}_e + \frac{g v^2 S}{2 m} C_{z \sigma_d} \cdot \dot{\sigma}_d \]

(B.5)
Most of the aerodynamic coefficients in the equations B.1-B.5 are nonlinear functions of the angle of attack and Mach number. Although the speed during landing is attempted to be kept constant, the nonlinear effect due to angle of attack, when it is large, also affects to the input functions such as aileron, elevator and rudder deflections. The aerodynamic derivatives like $C_{n \delta A}$ (yaw rate due to deflection of ailerons) and $C_{n \phi}$ (yaw rate due to the roll rate) can change signs, when the angle of attack increases or when the inputs are over certain limits. Data found for this simulation did not include information about nonlinear effects and so for simplicity, they were neglected.

B.1 Selection of Equations

Although the equations B.1-B.5 show very clearly the importance of inertial cross-coupling terms and the dependence of the aerodynamic coefficients on the angle of attack, these effects could not be included in the simulation due to the capacity of the TR-48 analogue computer. Four out of the five available multipliers in the TR-48 analogue computer were used for display and only one was left for the simulation of the aircraft equations. Therefore, the inertial cross-coupling terms were assumed to be constant. This kind of simulation of a Caravelle aircraft was already used in reference 2. The simulation equations are:

Pitch

$$I_{yy} \ddot{\theta} = g \cdot s \cdot \cos \left\{ \left[ C_{m \dot{\alpha}} \left( \frac{\dot{\alpha} \cdot \ell}{2 \cdot v} \right) \right] + C_{m \dot{\theta}} \left( \frac{\dot{\theta} \cdot \ell}{2 \cdot v} \right) +
\left. + C_{m \delta e} \cdot \delta e + C_{m \alpha} \cdot \alpha \right\}$$

(B.6)
Roll

\[ I_{xx} \dot{\phi} = g \delta b \left( C_{\ell r} \left( \frac{v \cdot b}{2v} \right) + C_{\ell \beta} \beta + C_{\ell \phi} \left( \frac{\dot{\phi} b}{2v} \right) + C_{\ell \delta a} \delta a + C_{\ell \delta e} \delta e \right) \]  

(B.7)

Yaw

\[ I_{zz} \ddot{\psi} = g \delta b \left[ C_{n \beta} \beta + C_{n \delta a} \delta a + C_{n \delta e} \delta e \right] + \frac{g \delta b^2}{2v} \left[ C_{n r} \dot{\psi} + C_{n \phi} \dot{\phi} \right] \]  

(B.8)

Y-Force

\[ m v \ddot{\psi} = g \delta' \left( C_{y r} \beta + C_{y \delta e} \delta e \right) - m v \dot{\psi} + m g \phi \]  

(B.9)

Z-Force

\[ v \ddot{z} = v \dot{\theta} - \frac{g v^2 \delta'}{2m} \left( C_{L \theta} \alpha + C_{L \delta e} \delta e \right) \]  

(B.10)
The maximum values for the problem variables are given in table B.I and the values of the aerodynamic and other coefficients used in the simulation are represented in table B.II. The scaled equations to be used in the TR-48 analogue computer are derived from the equations B.6-B.10 and from values in the tables B.I and B.II.

Pitch

\[
\begin{bmatrix}
100 \ddot{\theta}
\end{bmatrix} = -2.72 \begin{bmatrix}
40 \delta_e
\end{bmatrix} - 2.98 \begin{bmatrix}
20 \alpha
\end{bmatrix} - 0.232 \begin{bmatrix}
100 \dot{\alpha}
\end{bmatrix}
- 0.518 \begin{bmatrix}
100 \dot{\theta}
\end{bmatrix}
\]

(B.11)

Roll

\[
\begin{bmatrix}
20 \dot{\phi}
\end{bmatrix} = 1.82 \begin{bmatrix}
20 \delta_a
\end{bmatrix} + 0.093 \begin{bmatrix}
40 \delta_r
\end{bmatrix}
+ 1.11 \begin{bmatrix}
20 \dot{\psi}
\end{bmatrix} - 1.48 \begin{bmatrix}
20 \dot{\alpha}
\end{bmatrix} - 1.60 \begin{bmatrix}
20 \phi
\end{bmatrix}
\]

(B.12)

Yaw

\[
\begin{bmatrix}
20 \dot{\psi}
\end{bmatrix} = 0.435 \begin{bmatrix}
20 \alpha
\end{bmatrix} - 0.169 \begin{bmatrix}
40 \delta_e
\end{bmatrix}
+ 0.193 \begin{bmatrix}
20 \dot{\psi}
\end{bmatrix} - 0.170 \begin{bmatrix}
20 \phi
\end{bmatrix}
\]

(B.13)
Y-Force

\[
\begin{bmatrix} 20 \dot{y} \end{bmatrix} = -0.087 \begin{bmatrix} 20 \dot{y} \end{bmatrix} - \begin{bmatrix} 20 \dot{\psi} \end{bmatrix} + 0.170 \begin{bmatrix} 20 \dot{\theta} \end{bmatrix}
\]

(B.14)

Z-Force

\[
\begin{bmatrix} 0 \dot{\alpha} \end{bmatrix} = \begin{bmatrix} 100 \dot{\phi} \end{bmatrix} - 3.04 \begin{bmatrix} 20 \dot{\alpha} \end{bmatrix}
\]

(B.15)

A diagram showing the computer program is given in figure B.1.

B.2 Inputs to the Simulator

The simulator was a two degree of freedom simulator set up for roll-pitch mode. The servomotor in the roll channel allowed larger rollrates than the pitchrates. Sensitivities and limiting angles of both channels were:

<table>
<thead>
<tr>
<th>Channel</th>
<th>Sensitivity</th>
<th>Limiting angle</th>
</tr>
</thead>
<tbody>
<tr>
<td>Roll channel</td>
<td>0.33 v/degree</td>
<td>±45°</td>
</tr>
<tr>
<td>Pitch channel</td>
<td>1.60 v/degree</td>
<td>±15°</td>
</tr>
</tbody>
</table>
Figure B.1 Analogue Computer Diagram for Aircraft Simulation Display
Computing Diagram in Fig. C.III
The pitch output from the computer was 0.35° and it was to be amplified before going to the simulator input.

The roll-angle input to the simulator was not the same roll-angle as used in the display, because the pilot does not feel any sideforce in a coordinated turn. On the other hand, the roll-rates in the coordinated turn are so small that the pilot cannot distinguish them from aerodynamic turbulence forces. That is why no roll rate was commanded to the simulator during the coordinated turn. The commanded roll-angle can be derived from figure B.11.

\[ \tan \varepsilon = \left[ \frac{V(\dot{\phi} + \dot{V})}{g} - g \phi \right] \cos \phi \]

**Figure B.11** Force balance in a noncoordinated turn.

and we get after neglecting $\dot{\phi}$

\[ \varepsilon = \frac{V}{g} \dot{V} - \phi \]
Where \( \phi \) is the roll-angle commanded to the simulator, representing the angular deviation of the aircraft from the apparent vertical.

**B.3 Determination of Altitude**

The outputs from the pitch-angle-loop and the Z-force-loop are the pitch-angle and the angle of attack. Changes in altitude can be achieved from these two variables.

\[
h = V (\Theta - \alpha)
\]

\[(B.18)\]

\[
h = \int h \, dt + h_0
\]

\[(B.19)\]

**B.4 Automatic Approach and Landing**

In the simulation of an ideal automatic landing, the aircraft was brought down along 3° glidepath. The speed and rate of descent were constant and the altitude was proportional to the distance from the end of the runway. It was not the purpose of this simulation to include the flare-out.
Table B.I

Maximum Values of Simulation Variables

<table>
<thead>
<tr>
<th>PROBLEM VARIABLE</th>
<th>MAXIMUM VALUE</th>
<th>COMPUTER VARIABLE</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\ddot{\theta}$</td>
<td>0.1 rad./sec.$^2$</td>
<td>$[100 \dot{\theta}]$</td>
</tr>
<tr>
<td>$\ddot{\psi}, \phi$</td>
<td>0.5 rad./sec.$^2$</td>
<td>$[20 \dot{\psi}, [20 \dot{\phi}]$</td>
</tr>
<tr>
<td>$\dot{\theta}, \dot{\alpha}$</td>
<td>0.1 rad./sec.</td>
<td>$[100 \theta], [100 \alpha]$</td>
</tr>
<tr>
<td>$\dot{\psi}, \phi$</td>
<td>0.5 rad./sec.</td>
<td>$[20 \dot{\psi}], [20 \dot{\phi}]$</td>
</tr>
<tr>
<td>$\ddot{\rho}$</td>
<td>0.5 rad./sec.</td>
<td>$[20 \dot{\rho}]$</td>
</tr>
<tr>
<td>$\Theta, \alpha$</td>
<td>0.5 rad.</td>
<td>$[20 \Theta], [20 \alpha]$</td>
</tr>
<tr>
<td>$\phi, \psi$</td>
<td>0.5 rad.</td>
<td>$[20 \phi], [20 \psi]$</td>
</tr>
<tr>
<td>$\rho, \delta_{\Delta}$</td>
<td>0.5 rad.</td>
<td>$[20 \rho], [20 \delta_{\Delta}]$</td>
</tr>
<tr>
<td>$\delta_{R}, \delta_{E}$</td>
<td>0.2 rad.</td>
<td>$[40 \delta_{R}], [40 \delta_{E}]$</td>
</tr>
<tr>
<td>$x$</td>
<td>18,000 ft.</td>
<td>$[0.0005 x]$</td>
</tr>
<tr>
<td>$h$</td>
<td>1,000 ft.</td>
<td>$[0.001 h]$</td>
</tr>
<tr>
<td>$h$</td>
<td>10 ft./sec.</td>
<td>$[h]$ ft./sec.</td>
</tr>
<tr>
<td>$y$</td>
<td>1,000 ft.</td>
<td>$[0.001 y]$</td>
</tr>
</tbody>
</table>
### Table B.II

**Equation of Simulation Parameters**

<table>
<thead>
<tr>
<th>SYMBOL</th>
<th>DEFINITION</th>
<th>VALUE</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td>707/320B</td>
</tr>
<tr>
<td>W</td>
<td>Maximum landing gross weight</td>
<td>207,000#</td>
</tr>
<tr>
<td>g</td>
<td>gravity acceleration</td>
<td>32.2 ft./sec.$^2$</td>
</tr>
<tr>
<td>V</td>
<td>approach speed</td>
<td>134 knots</td>
</tr>
<tr>
<td>m</td>
<td>w/g</td>
<td>6440 slugs</td>
</tr>
<tr>
<td>q</td>
<td>aero dynamic pressure</td>
<td>60.7 #/ft.$^2$</td>
</tr>
<tr>
<td>S</td>
<td>wing area</td>
<td>2892 ft.$^2$</td>
</tr>
<tr>
<td>b</td>
<td>wing span</td>
<td>142.42 ft.</td>
</tr>
<tr>
<td>C</td>
<td>mean aerodynamic chord</td>
<td>22.69 ft.</td>
</tr>
<tr>
<td>$I_{xx}$</td>
<td>moment of inertia about principal</td>
<td>$3.25 \times 10^6$</td>
</tr>
<tr>
<td></td>
<td>X axis</td>
<td>slug ft.$^2$</td>
</tr>
<tr>
<td>$I_{yy}$</td>
<td>moment of inertia about principal</td>
<td>$5.15 \times 10^6$</td>
</tr>
<tr>
<td></td>
<td>Y axis</td>
<td>slug ft.$^2$</td>
</tr>
<tr>
<td>$I_{zz}$</td>
<td>moment of inertia about principal</td>
<td>$8.6 \times 10^6$</td>
</tr>
<tr>
<td></td>
<td>Z axis</td>
<td>slug ft.$^2$</td>
</tr>
<tr>
<td>$I_{xy}$</td>
<td>cross products of inertia</td>
<td>0</td>
</tr>
<tr>
<td>$I_{xz}$</td>
<td>cross products of inertia</td>
<td>0</td>
</tr>
<tr>
<td>$I_{yz}$</td>
<td>cross products of inertia</td>
<td>0</td>
</tr>
<tr>
<td>$C_{y/\alpha}$</td>
<td>change in side force coeff. with variation in $\alpha$</td>
<td>-.745 rad.$^{-1}$</td>
</tr>
<tr>
<td>$C_{y/\alpha}$</td>
<td>change in side force coeff. with variation in $\alpha$</td>
<td>.20 rad.$^{-1}$</td>
</tr>
<tr>
<td>$C_{r/\psi}$</td>
<td>change in rolling moment coeff. with variation in $\psi$</td>
<td>.337 rad.$^{-1}$</td>
</tr>
<tr>
<td>SYMBOL</td>
<td>DEFINITION</td>
<td>VALUE</td>
</tr>
<tr>
<td>---------</td>
<td>--------------------------------------</td>
<td>---------------</td>
</tr>
<tr>
<td>$C_{lp}$</td>
<td>change in rolling moment coef.</td>
<td>-.18 rad.$^{-1}$</td>
</tr>
<tr>
<td></td>
<td>with variation in $\phi$</td>
<td></td>
</tr>
<tr>
<td>$C_{l/\alpha}$</td>
<td>change in rolling moment coef.</td>
<td>-.18 rad.$^{-1}$</td>
</tr>
<tr>
<td></td>
<td>with variation in $\alpha$</td>
<td></td>
</tr>
<tr>
<td>$C_{l \delta_A}$</td>
<td>change in rolling moment coef.</td>
<td>.2213 rad.$^{-1}$</td>
</tr>
<tr>
<td></td>
<td>with variation in $\delta_A$</td>
<td></td>
</tr>
<tr>
<td>$C_{l \delta_2}$</td>
<td>change in rolling moment coef.</td>
<td>.0086 rad.$^{-1}$</td>
</tr>
<tr>
<td></td>
<td>with variation in $\delta_2$</td>
<td></td>
</tr>
<tr>
<td>$C_{n_r}$</td>
<td>change in yawing moment coef.</td>
<td>-.135 rad.$^{-1}$</td>
</tr>
<tr>
<td></td>
<td>with variation in $\nu$</td>
<td></td>
</tr>
<tr>
<td>$C_{n_{\beta}}$</td>
<td>change in yawing moment coef.</td>
<td>.124 rad.$^{-1}$</td>
</tr>
<tr>
<td></td>
<td>with variation in $\beta$</td>
<td></td>
</tr>
<tr>
<td>$C_{n_{\beta'}}$</td>
<td>change in yawing moment coef.</td>
<td>.0132 rad.$^{-1}$</td>
</tr>
<tr>
<td></td>
<td>with variation in $\beta'$</td>
<td></td>
</tr>
<tr>
<td>$C_{n_p}$</td>
<td>change in yawing moment coef.</td>
<td>-.1885 rad.$^{-1}$</td>
</tr>
<tr>
<td></td>
<td>with variation in $\phi$</td>
<td></td>
</tr>
<tr>
<td>$C_{n \Delta}$</td>
<td>change in yawing moment coef.</td>
<td>.00934 rad.$^{-1}$</td>
</tr>
<tr>
<td></td>
<td>with variation in $\Delta$</td>
<td></td>
</tr>
<tr>
<td>$C_{n \delta_2}$</td>
<td>change in yawing moment coef.</td>
<td>-.081 rad.$^{-1}$</td>
</tr>
<tr>
<td></td>
<td>with variation in $\delta_2$</td>
<td></td>
</tr>
<tr>
<td>SYMBOL</td>
<td>DEFINITION</td>
<td>707/320B</td>
</tr>
<tr>
<td>---------</td>
<td>-----------------------------------</td>
<td>----------</td>
</tr>
<tr>
<td>$C_L\alpha$</td>
<td>change in lift coef. with variation in $\alpha$</td>
<td>4.48 rad.$^{-1}$</td>
</tr>
<tr>
<td>$C_L\delta_E$</td>
<td>change in lift coef. with variation in $\delta_E$</td>
<td>-.238.$^{-1}$</td>
</tr>
<tr>
<td>$C_m\alpha$</td>
<td>change in pitching moment coef. with variation in $\alpha$</td>
<td>-5.12 rad.$^{-1}$</td>
</tr>
<tr>
<td>$C_m\delta$</td>
<td>change in pitching moment coef. with variation in $\delta$</td>
<td>-1.14 rad.$^{-1}$</td>
</tr>
<tr>
<td>$C_m\delta_E$</td>
<td>change in pitching moment coef. with variation in $\delta_E$</td>
<td>.645 rad.$^{-1}$</td>
</tr>
<tr>
<td>$C_m\alpha$</td>
<td>change in pitching moment coef. with variation in $\alpha$</td>
<td>-1.12 rad.$^{-1}$</td>
</tr>
<tr>
<td>$C_m\alpha$</td>
<td>change in pitching moment with zero angle of attack</td>
<td>-.057 (1/1)</td>
</tr>
</tbody>
</table>
APPENDIX C

Runway Display

During visual landing of an aircraft, the pilot obtains a vital part of the information needed through the windshield by visual means. From this picture, the pilot gets the following information:

1. Position of the runway relative to the aircraft.
2. Pitch, roll and yaw attitudes of the aircraft.
3. Approximate altitude of the aircraft.
4. Velocity of the aircraft relative to the ground.

Being aware of the position of the runway, the pilot can plan his future flight-path so that the approach takes place in the right direction. On the basis of this information, all other essential parts of the total picture are built. For the landing, he knows from experience that his own pitch, roll and yaw attitudes have to match with the picture of the runway. He gets his roll information by comparing the relative position of the fuselage contour with the visual horizon or by just looking at the perspective of runway lights during night VFR-approaches. The relative accuracy of the information obtained from different parts of the picture is discussed in Chapter VI. Although it is rather difficult to determine the pitch and yaw angles from the static picture because of the lack of references, the changes of these angles can be determined relatively easily from the dynamic picture seen by the pilot in actual flight.
Determination of altitude from the picture through the windshield alone is rather complicated and in usual cases inaccurate. There are three different information sources which can be used by the pilot, and the use of these is very much dependent on his experience.

1. Angle of runway boundary lights.
2. Relative position of horizon to the runway.
3. Motion of different points in field of vision.

The angle of the runway boundary lights gives a rough indication of altitude when the pilot effectively solves the equation:

$$\Omega = \tan^{-1} \frac{2w}{h}$$

where $2w =$ width of the runway

$h =$ altitude

$\Omega =$ perspective angle between the sides of the runway seen by the pilot. \hspace{1cm} (C.1)

This altitude information is rather useless in the beginning of the approach, because when seen from a long distance, the runway lights seem to be parallel and the perspective angle changes very little with a moderate change in altitude. When the distance to the runway gets shorter and the altitude near 300 ft., the angle $\Omega$ starts to change very rapidly and the information is more accurate. But even at lower altitudes, the exact altitude is very difficult to determine. Altitude corrections made during the initial part of the approach are nearly
impossible to rectify during the last 10-15 seconds of the approach.
The situation is similar in the second type of altitude information,
namely that given by the relative position of the horizon and the runway.

In many cases, the over-all information obtained from the motion of
different points in the field of view has proved to be important. For
example, in reference 1 it is shown that during landing, changes in the
picture occur tangentially outward from the point in the ground toward
which the airplane is heading. The velocity of this motion is inversely
proportional to the the distance from the point, and so nearby points
move very fast giving good altitude information at low altitudes. This
motion also gives some indication about yaw and pitch attitude and the
speed of the aircraft relative to the ground.

To determine the display, which gives the pilot the maximum amount
of information with the least amount of picture, three different simple
pictures were generated on the CRT located in front of the pilot, namely:

1. Two lines, representing runway boundary lights.

2. Three lines, two representing the runway boundary
   lights and the other, the horizon.

3. Two lines and a dot; the two lines representing
   the runway boundary lights and the dot showing
   the spot where the velocity vector of the aircraft
   intersects the ground.

The pictures were displayed on a CRT and generated by a TR-48
analogue computer. In deriving the computer diagrams, nomenclature
of figure C.1 was used. Figure C.1 shows one static situation from
which one is able to derive coordinates for the display by using geo-
metrical relations.
\[
Z = d \cdot \frac{-h}{x+1}
\]

(C.2)

\[
Y = d \cdot \frac{y-w}{x+1}
\]

(C.3)

The runway boundary lights can be created by using a square wave signal and integrator as shown in figure C.II. The square wave alternates between \( +w \) and \( -w \) where \( 2w \) is the width of the runway. An integration of this square-wave gives a triangular wave. It is adjusted so that its amplitude is proportional to the length of the runway. By this means, exact synchronisation is achieved. Unfortunately, the triangular wave is very sensitive to developing bias drift due to variations in square wave signal generation.

When the attitude of the aircraft is changed from the neutral position, the picture of the runway changes because the display is in the aircraft body axes and the runway is in the fixed earth axes.

In order to get the true picture, a coordinate transformation from earth axes to aircraft body axes must be made.

In the three-dimensional case, we have:

\[
\begin{bmatrix}
X \\
Y \\
Z
\end{bmatrix}_{\text{body}} = \begin{bmatrix}
C_{B.E}
\end{bmatrix} \begin{bmatrix}
d \\
d \cdot \frac{y+w}{x+1} \\
d \cdot \frac{-h}{x+1}
\end{bmatrix}_{\text{earth}}
\]

(C.4)
Where $C_{B.E}$ is the coordinate transformation matrix from earth-coordinates to body-coordinates. By using the Euler-angles of the airplane, the transformation matrix can be obtained by a different sequence of rotation. In this case, the sequence of rotation roll $\rightarrow$ pitch $\rightarrow$ yaw is selected and so by using direction cosines:

\[
C_{B.E} = \begin{pmatrix}
\cos \psi & \sin \psi & 0 \\
-\sin \psi & \cos \psi & 0 \\
0 & 0 & 1
\end{pmatrix}
\begin{pmatrix}
\cos \Theta & 0 & -\sin \Theta \\
0 & 1 & 0 \\
\sin \Theta & 0 & \cos \Theta
\end{pmatrix}
\begin{pmatrix}
1 & 0 & 0 \\
0 & \cos \phi & \sin \phi \\
-\sin \phi & \cos \phi & 0
\end{pmatrix}
\]

and multiplying through:

\[
= \begin{pmatrix}
\cos \psi \cos \Theta & \sin \psi \cos \phi + \sin \Theta \sin \phi \sin \psi \cos \Theta & \sin \psi \sin \phi - \sin \Theta \cos \psi \cos \phi \\
-\sin \psi \cos \Theta & \cos \psi \cos \phi - \sin \Theta \sin \phi \sin \psi \cos \Theta & \sin \psi \sin \phi + \sin \Theta \cos \psi \cos \phi \\
\sin \Theta & -\sin \phi \cos \Theta & \cos \phi \cos \Theta
\end{pmatrix}
\]

(c.5)

Because of the limited capacity of the TR-48 analogue computer (only 5 multipliers), all the terms from the coordinate transformation matrix cannot be included in the computation. At first, a small angle approximation was made and third-order sine-terms were dropped. The final transformation matrix is:

\[
C_{B.E} = \begin{pmatrix}
1 & \Theta \phi + \psi & -\Theta + \psi \phi \\
\psi & 1 & \phi \\
\Theta & -\phi & 1
\end{pmatrix}
\]

(c.6)
The matrix used in equation C.4 gives:

\[ \bar{x} = d + (\theta \phi \psi) d\left(\frac{y \times w}{x+L}\right) + (-\theta + \psi \phi) d\left(-\frac{h}{x+L}\right) \]

\[ \bar{y} = -d\left(\psi + \frac{y \times w}{x+L} + \frac{h \phi}{x+L}\right) \]

\[ \bar{z} = -d\left(-\theta + \frac{y \times w}{x+L} \phi + \frac{h}{x+L}\right) \]

(C.7)

from which only \( \bar{y} \) and \( \bar{z} \) are used in the display. Figure C.III shows the computer diagram used for the generation of display coordinates \( \bar{y} \) and \( \bar{z} \) from equation C.7.

Horizon

The same techniques as used above were used to generate the horizon line. In the Y-direction, the scope was driven from one end to the other by using a rep-op. peak signal coming from the computer. The Z-drive was:

\[ Z = d \cdot \Theta + Y \phi \]

(C.8)

Due to a shortage of multipliers, part \( Y \phi \) in equation C.8 was generated by using a repetitive operation of the computer. The computer diagram used is shown in figure C.III.

Dot

The third variation of the picture displayed to the pilot was a dot
showing the spot in the ground, toward which the velocity vector of
the aircraft was pointed. In figure 4, the geometry on which the
use of dot is based is shown:

\[ \ddot{y} = \dot{d} \gamma \]  \hspace{1cm} (c.9)

\[ \ddot{z} = d (\dot{\theta} - \alpha) = d \dot{\gamma} \]  \hspace{1cm} (c.10)
Figure C.I Geometry of the Display

Figure C.II Generation of voltages for Display of the Runway Boundary Lines
Figure C.III  Computer Diagrams for Displayed Pictures
(For Aircraft Simulation, see fig. B.I)
REFERENCES


