SPERRY

SPERRY PHOENIX COMPANY Division of Sperry Rand Corporation PHOENIX, ARIZONA

CR 73092

final report

Avionics Requirements for All Weather Landing of Advanced SST's

Volume I
Analysis of System Concepts and Operational Problems

Prepared Under
NASA Contract NAS2-4124 for
NASA - Ames Research Center
Moffett Field, California

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Prepared By S. S. Osder

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SECTION I

INTRODUCTION

This report reviews the technical and operational problems of all weather landing (AWL) with the intent of focusing on those aspects which are amenable to solution by progress in avionics concepts and equipment. The approach taken is to analyze the present status of AWL activity and then project the problem into the era of the supersonic transport (SST). With a projected view of all weather landing operations in the 1975-1985 period, we can appraise the adequacy of the avionics trends we see today and, in an idealized sense, we can define a direction for avionics development. Hopefully, this analysis can lay the groundwork for a fruitful research and development plan to ensure that avionics systems will meet the challenge of the anticipated revolutionary strides in aviation. As a minimum reward, this quest for future directions could help avert some of the chaos which many pessimists predict as the outcome of the rapid expansion of air transportation.

A view of the AWL situation today is one of considerable development and flight test activity. There is interest and a strong determination on the part of the aircraft operators, the government agencies, the equipment suppliers, and the technical community to proceed with the objectives of making AWL an operational reality; but there is uncertainty and controversy on how these objectives can be achieved. Progress is often difficult to ascertain and the participant in AWL programs usually has difficulty "distinguishing the forest from the trees". Many of the individual airline carriers have embarked on orderly programs to introduce AWL operations on an evolutionary basis; and from their vantage point, progress is readily discernible and often satisfactory. To the technical observer viewing the AWL situation from the outside, there is a considerable amount of confusion concerning the extent of progress which has been made. The technical literature and the various symposia that have been held on this subject describe the thousands of successful automatic landings performed at many airports and with many different aircraft. Indeed, the various equipment manufacturers with a vested interest in AWL do not hesitate to inform the symposia audiences about their history of successful automatic landings dating back to the 1940's and early 1950's. Every symposium seems to have at least one commentator who reminisces about his blind landing demonstrations in the 1930's. It is therefore obvious that the technical feasibility of automatic or blind landing is neither questioned or considered as the pacing factor in attaining the objectives sought by the aviation industry.

The main impediments to the introduction of all weather landing involve operational procedures rather than individual technical problems. While technical difficulties do exist, the problem is primarily one of resolving a diversity of forces and constraints imposed by considerations of economics, safety and reliability, psychological factors, and vaguely defined philosophies. Economics has always been the primary motivation for AWL as well as the major constraint on its implementation. Airline economic losses resulting from weather diversions and flight cancellations justify expenditures on equipment that could minimize the total number of these diversions. In this scientific age, tradeoff studies are therefore performed to define the reduction in dollar losses as a function of expenditure on new equipment that will permit the aircraft to penetrate to lower altitudes during low visibility situations. Hence, the inevitable meteorological statistics enter the problem. The economic dilemma is, unfortunately, not this simple because the necessary equipment investment is beyond the control of any individual aircraft operator or government agency. Attaining lower minimums often involves major modifications of airport facilities as well as investments in airborne avionics equipment. Various competing approaches have different ratios of ground facility costs to airborne equipment costs. Most operators already have sizable investments in some types of equipment that heavily bias their economic studies in favor of certain technical approaches. For example, a carrier that has its fleet equipped with radio altimeters would obviously prefer that investments in airport improvements concentrate on leveling the runway approach profile so that the effectiveness of landing procedures using these altimeters would be enhanced. He could not be expected to show enthusiasm for an airport facility investment aimed at introducing a radar derived precision approach system which would substitute a new set of airborne equipment for his radio altimeters.

Safety and reliability are the essential guidelines for AWL activities, but a clear concept of how to evaluate these factors is lacking. While on the surface these two virtues seem synonymous, they may actually be mutually incompatible. For example, safety considerations often require monitoring circuits or even complete monitoring subsystems to warn or provide some action in the event of a failure in a system component. These monitoring devices must themselves be fail-safe so that before long, system complexity tends to grow to unreasonable levels. It is not uncommon to find monitors more complex than the devices being monitored. Safety is thereby bought at the expense of reliability. Similar observations may be made concerning the use of redundant systems. This pattern toward progressing complexity is not too often viewed with concern because of the ubiquitous faith that microelectronics will somehow provide the necessary breakthroughs. Indeed, one of the objectives of the present study is to identify areas in which the future avionics, dominated no doubt by "computer on a chip" type devices, can be applied to these problems.

When we mention psychological factors, reference is made to attitudes, prejudices, and other nonquantifiable, emotional aspects of the problem. The role of the pilot in managing the aircraft is a major source of controversy in this area. How will pilots accept a role of systems manager with the "automatics" flying the aircraft? How will pilots develop and maintain flying skills if normal procedures leave the flying task to the autopilot? These fears have led to conflicting schools of thought which are often identified as the United Kingdom and United States philosophies of all weather landing. The UK approach emphasizes the complete landing operation by the automatic equipment. It attempts to ensure safety through triplicated redundancy and uses the pilot to monitor system performance. A common view of the US approach is that it emphasizes the pilot's active role in controlling the aircraft. It seems to stress those factors which would help the pilot abort the landing rather than those which would ensure the continued operation of the automatic equipment in the event of individual malfunctions.

Actually, there are no authoritative directions to the US programs which challenge or conflict with the technical approach taken by the various UK efforts to implement AWL operations. Many of the US projects seem to emphasize the sanctity of the "pilot in the loop". For example, the provision of the so-called supervisory override, force wheel steering mode as an adjunct to the autopilot is generally well received by pilots. Somehow, giving the pilot the capability of inserting corrective inputs to the autopilot seems to make the automatic control of the landing approach more palatable to the pilot. A properly functioning autopilot that requires manual corrective inputs is in itself a contradiction. If the autopilot is not functioning properly, then why insert commands through the autopilot. It would be convenient to state that the corrective inputs will bypass the malfunctioned part of the autopilot, but unfortunately the major reliability hazards of an autopilot system lie in the path of the force wheel input signals. It is often suggested that the supervisory override function is offered to the pilot as a psychological placebo.

We have dwelled on this point somewhat in order to note how the era of the SST should inherently resolve this problem. The primary flight control systems of the Concorde, the Lockheed L-2000, and the Boeing 2707 all use electronic command systems for manual control. These systems are of the triply redundant, fail-operational type with the necessary complement of failure monitors and warning displays. Thus, the philosophy which we tend to associate with the United Kingdom approach to AWL will actually be implemented in the primary flight controls of the SST. The pilot will always control through the "automatic" equipment, except during emergency situations when a degraded performance, manual backup capability will be available. (This backup mode may not provide sufficient control precision to be used in a low visibility approach.) The autopilot command signals will normally flow through the same electronic servomechanisms as the signals generated by the pilot's application of control force. In an AWL approach, the pilot in the loop will not really be able to bypass any automatic equipment except possibly to abort the landing.

His normal role appears more and more to be that of avionics system manager. There appears to be universal agreement that the pilot's performance monitoring capability and his ability to make judgment in abnormal circumstances cannot be duplicated by avionics equipment.

This rapid scan of some of all weather landing phenomena takes us back to the main premise of this report. The trend toward avionics playing an ever increasing role in aircraft operations has been demonstrated. Whether avionics will be used effectively is a continuous challenge. In order to define its role in the SST or in second-generation SST's, we must clearly perceive the trends in terms of today's activities. The AWL systems in use in 1985 will not be unique to the SST. Just as our highways are built to handle Volkswagens as well as Cadillacs, the airport's ground-based electronic facilities of 1985 will have to accommodate all types of aircraft including most of those operating today. Consequently, the avionics equipment on future SST's will have its origins in developments underway today. Section II of this report is a brief review of the operational problems that are defining the directions of AWL developments today. Sections III to V summarize various approaches to solve the problems of AWL. It is shown that many different systems associated with various equipment and aircraft manufacturers are essentially identical except for details of mechanization. The problem areas are identified and possible solutions are defined. Since this report is concerned with avionics, the various problem solutions suggested are based on avionics progress. It is noted, however, that in some instances, competitive but less elegant solutions using cement and asphalt are possible. That is, many of the problems are alleviated by wider and longer runways.

Sections VI to VIII provide an extrapolation of the AWL problem to the SST. The unique characteristics of the SST's primary flight controls are described. The handling qualities of this class of aircraft are reviewed from the standpoint of the AWL problem. The expanded role of avionics in this type of aircraft creates new problems of data handling and transmission. Various concepts of integrated avionics based on central digital computers or small autonomous but interconnected computers are examined. From this perspective, the relationship between the AWL subsystems and other subsystems of the large avionics complex is explored. This section therefore provides a prediction of the probable form of the AWL avionics that will be needed for aircraft in the 1985 era. When predictions concerning future electronic systems are made, there is a tendency to fall into the realm of fantasy. Since the purpose of this report is not to provide science fiction entertainment but to develop a realistic set of research and development (R&D) objectives, the treatment of these long-term trends emphasizes the practical steps which must be taken to achieve the desired progress.

SECTION II

AWL SYSTEM CONCEPTS AND OPERATIONAL PROBLEMS

A. THE GEOMETRY OF AN ALL WEATHER LANDING

The best description of the all weather landing problem is obtained by viewing the geometry of the aircraft's trajectory as it progresses through its final approach to touchdown on the runway. Figure 2-1 shows the vertical profile of such a trajectory and the various control and maneuvering phases associated with the sequence of events leading to touchdown. The distances and durations shown are typical for aircraft operations using present day Instrument Landing System (ILS) facilities. These dimensions are typical but the fact that they may be quite variable causes many of the problems involved in achieving automatic or manual control accuracies. The effect of variable factors are described in the subsequent section on the control accuracy problem.

A horizontal view of the aircraft's landing trajectory is shown in figure 2-2. The landing approach starts with the intercept and capture of the localizer beam. The fact that the localizer is established by a radiated pattern of a particular frequency and modulation characteristic is immaterial for this discussion. The important point is that a reference path representing the extension of the runway centerline must be used to align the aircraft for its final approach. It will be shown in the subsequent discussion of control accuracy factors that the approach and turn onto this path must be performed within certain intercept angle and distance from the runway constraints. As shown in figure 2-2, this turn on capture maneuver is performed about 16.7 kilometers (9 nautical miles) out from the runway threshold. The aircraft should stabilize on this path, preferably before it has begun its final descent. As indicated in figure 2-1, this phase proceeds for about 2 minutes usually at constant altitude. The airspeed is programmed down from about 160-180 knots to a 140-150 knot range at this time.

When the aircraft approaches an intersection with the glide slope descent path, the altitude hold function is discontinued and a glide slope capture maneuver program is initiated. At this time, final flap adjustments may be made and the throttles reduced for the final descent speed, usually about 120 knots in present day turbojet aircraft. The glide slope capture maneuver is designed to produce a flight path rotation which should ideally provide a tangential intersection with the glide slope. At the proper time during this capture maneuver, closed loop control to the reference glidepath is initiated. As rapidly as possible the aircraft is stabilized on this path so that its velocity vector is directed precisely at the point defined by the intersection of the glidepath and the runway.

	Flare (E)	Glide Slope (D)	Glide Slope Capture (C)	Localizer On Course (B)	Localizer Capture (A)
Pitch	$h + \dot{h}$ Control	G/S Stabilization and Attitude	Programmed Capture Maneuver	Altitude Hold	Altitude Hold
Lateral	Roll-out Gudiance	$y + \dot{y}$ Stabilization →			Heading Select Plus Capture Maneuver Program
Throttle	Reduce Power	Throttle Hold at 120 Knots	Throttle Ref. Reduced to 120 Knots	Auto Throttle Reference Programmed Down to 140 - 150 Knots	Auto Throttle Reference 160 - 180 Knots
Time	10 Sec.	120-130 Sec.	20-30 Sec.	120-130 Sec.	

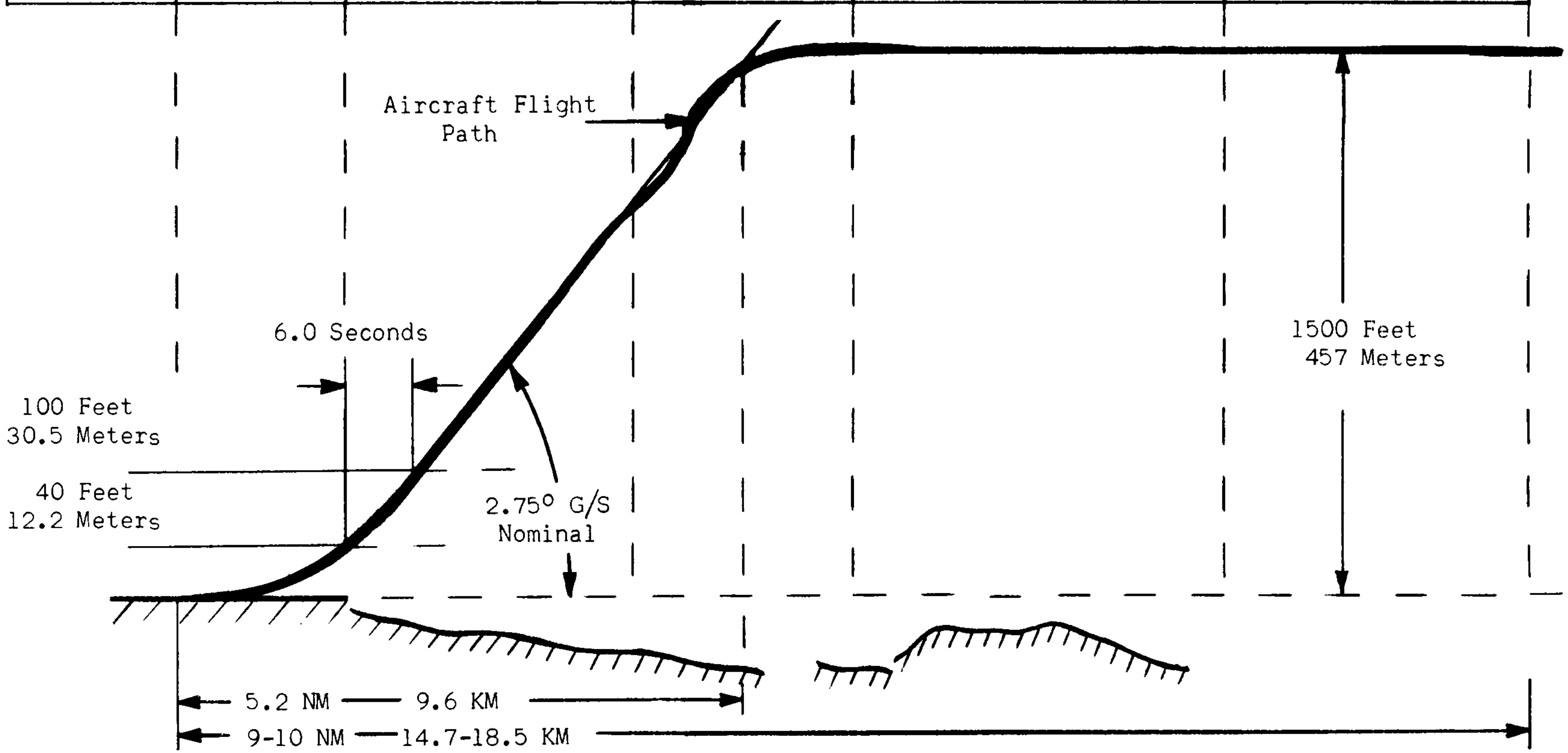


Figure 2-1
Phases of AWL

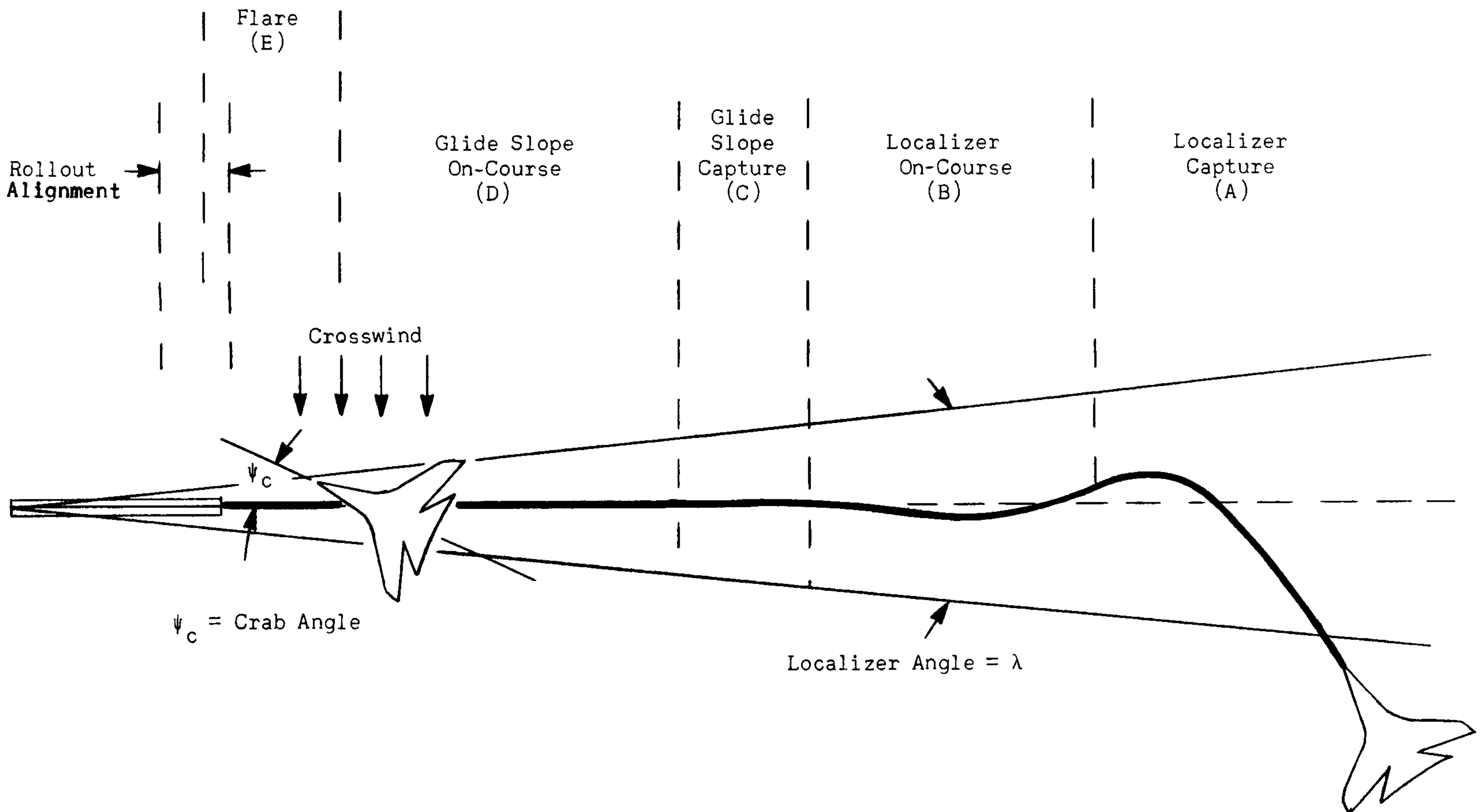


Figure 2-2
Horizontal View of Automatic Approach

When the aircraft reaches an altitude of about 15.24 meters (50 feet), the flareout begins. The rate of descent must now be reduced from about 3.05 meters (10 feet) per second to about 0.30 meter (1.0 foot) per second at the time the wheels contact the runway. During this flight path rotation the airspeed will decay because of an aircraft's usual dynamic relationship between flight path angle and speed. In addition, throttles are generally cut back to further reduce speed at touchdown. Prior to touchdown, the aerodynamic phenomenon referred to as ground effect often helps ameliorate a landing that was not executed with the ideal flare rotation. Ground effect provides increased lift and some change in drag (usually a reduction) as the ground is approached. Hence it tends to reduce the rate of descent, and indeed, some of the SST designs which have been considered could reduce the descent rate at touchdown by depending almost entirely on ground effect rather than the flare maneuver.

Near the end of the flareout maneuver the phase often referred to as rollout alignment is initiated. The type of alignment maneuver required depends upon the crosswind conditions prevailing and the technique used to adapt to this crosswind. Figure 2-2 shows a schematic representation of the aircraft compensating for the crosswind by assuming a crab angle, ψ_c, with respect to the runway heading. As illustrated in this figure, the aircraft's velocity vector is not aligned with the runway heading. The sideslip angle is zero since the velocity vector is also aligned with the relative wind but the aircraft heading is displaced from the runway heading by the amount ψ_c. For this type of approach it is desirable that the aircraft be yawed immediately prior to touchdown to avoid the side load on the landing gear. This yawing maneuver, referred to as the decrab phase, will tend to produce a sideslip angle equal to ψ_c at touchdown, but it must be accomplished rapidly to minimize translation across the runway. An alternate method of compensating for the crosswind avoids the need for a crab angle. This method approaches touchdown with the aircraft's nose aligned to the runway heading and hence it is sideslipping. To maintain this slip condition, the wing facing the crosswind is dropped and the rudder is deflected to prevent turning into the crosswind. No decrab maneuver is needed and the aircraft rolls to level naturally after ground contact is initially made with the wheel on the wing down side. Most automatic flight path control systems are configured to produce the crab angle approach although some systems have been implemented to provide the wing down, sideslip landing. The latter configurations involve more complex computations and are somewhat restricted by autopilot authority limitations. The best technique to use is often dependent upon specific aircraft characteristics.

After touchdown, continued automatic guidance is required for true zero-zero conditions. The steering information can, in general, be obtained from the same localizer signal which provided lateral guidance during approach. In aircraft that mechanically couple the nosewheel and rudder and that contain a large authority autopilot rudder servo to move the rudder pedals as well as the rudder, such a runway steering mode is easy to implement. This combination of features is unfortunately not found very often in modern aircraft.

B. ALL WEATHER LANDING CONTROL ACCURACY PROBLEMS

If we provide an automatic control system with sufficient information and authority, almost any level of performance can be obtained. The only problem remaining (assuming we had solved the weight and cost problem) would be that of unreliability, which, to our regret, seems to grow in direct proportion to complexity. The problems which have been encountered thus far in attempts to guide an aircraft to a blind landing have resulted from the information and control authority constraints. A system can be optimized to perform an automatically guided approach and landing if we fix such factors as the following:

- The program of aircraft speed along the reference trajectory
- The ILS beam geometry and signal characteristics
- The sequence of configuration changes (landing gear and flaps)
- Aircraft cg location and dynamic characteristics
- Wind velocities and velocity gradients (shear)
- The final approach path intercept azimuth and altitude
- The control system maneuvering limits

System performance will tend to change with variations in these factors. If the system received information regarding the status of these factors, it could be designed to cope with many of these variations. Consequently, two approaches have been followed to improve all weather landing system performance. They are as follows:

- Standardize on procedures and geometry of approach flight paths
- Provide additional information interfaces to improve the system's ability to cope with problem variables

The following discussions review how the above mentioned factors affect all weather landing system performance and how present trends are attempting to solve these problems.

1. Lateral Flight Path Control

1.1 Localizer Capture

As seen in figure 2-2, the first phase of the final approach involves a turn on maneuver or localizer beam capture phase. It would be desirable to approach the beam center with any intercept angle, perform a single turn and, upon rollout, end with the aircraft's velocity vector on the beam center and pointing to the runway target. As implied by figure 2-2, a nonideal response such as an overshoot of the beam center is a usual result. An overshoot, if it occurs, does not result from a lack of skill on the part of the system designers. There are initial conditions for which a successful turn on is physically impossible within the bank angle and roll rate constraints. Likewise, there are conditions where some overshoot, but with recovery capability is a physical necessity. This can be understood by referring again to the sketch of the aircraft intercepting the localizer beam during Phase A in figure 2-2. When the aircraft is outside the boundary of the beam (defined by the angle λ), it proceeds at a constant heading toward the beam center. The maneuver to rotate the aircraft begins after the lateral guidance system has sensed that the confines of the beam have been penetrated. Because of the convergent nature of the localizer beam, the distance between the outer boundary and the center is variable with distance from the transmitter. Thus at 12 nautical miles out from the transmitter, a typical localizer beam width (outer boundary to center) is about 1249.68 meters (4100 feet). However, at 6 nautical miles out, the beam width will be only one-half this distance. The turning radius of an aircraft flying at 160 knots and banking at 30 degrees is about 914.40 meters (3000 feet). Thus if the intercept angle is 90 degrees, and if we assume infinite roll rate and acceleration capability, this aircraft could not possibly acquire the localizer beam at the 6 nautical mile point. It would have to overshoot nearly the full width of the beam.

When we consider the roll rate restrictions and reasonable filtering of the sensed data, the region of permissible localizer intercepts is quite restrictive. Aircraft velocity (and winds), intercept angle, and specific beam geometry parameters are additional factors which enter into this problem. Figure 2-3 is a plot of how these parameters define a region of acceptable approaches to the localizer. These graphs were obtained from Sperry Phoenix Company simulator studies of performance attainable with various control configurations. Note that for the 160 knot, 10 nautical mile range case discussed previously in terms of the kinematic limits only, the actual capability with roll rate restrictions and realistic proportional steering laws allows a maximum beam intercept angle of only 51 degrees. The 75-millivolt overshoot criterion used in the study corresponds to an overshoot of one-half the full width of the localizer beam.

The point of this discussion is that a system which can provide excellent response when the proper procedures and restrictions are observed can begin to produce objectionable and then completely unacceptable performance when deviations from the nominal procedures occur. Unfortunately, in the case of the localizer capture maneuver, some of the procedural deviations have operational advantages. When the air traffic control problem is considered, there are obvious advantages to funnelling the aircraft into the localizer as close to the

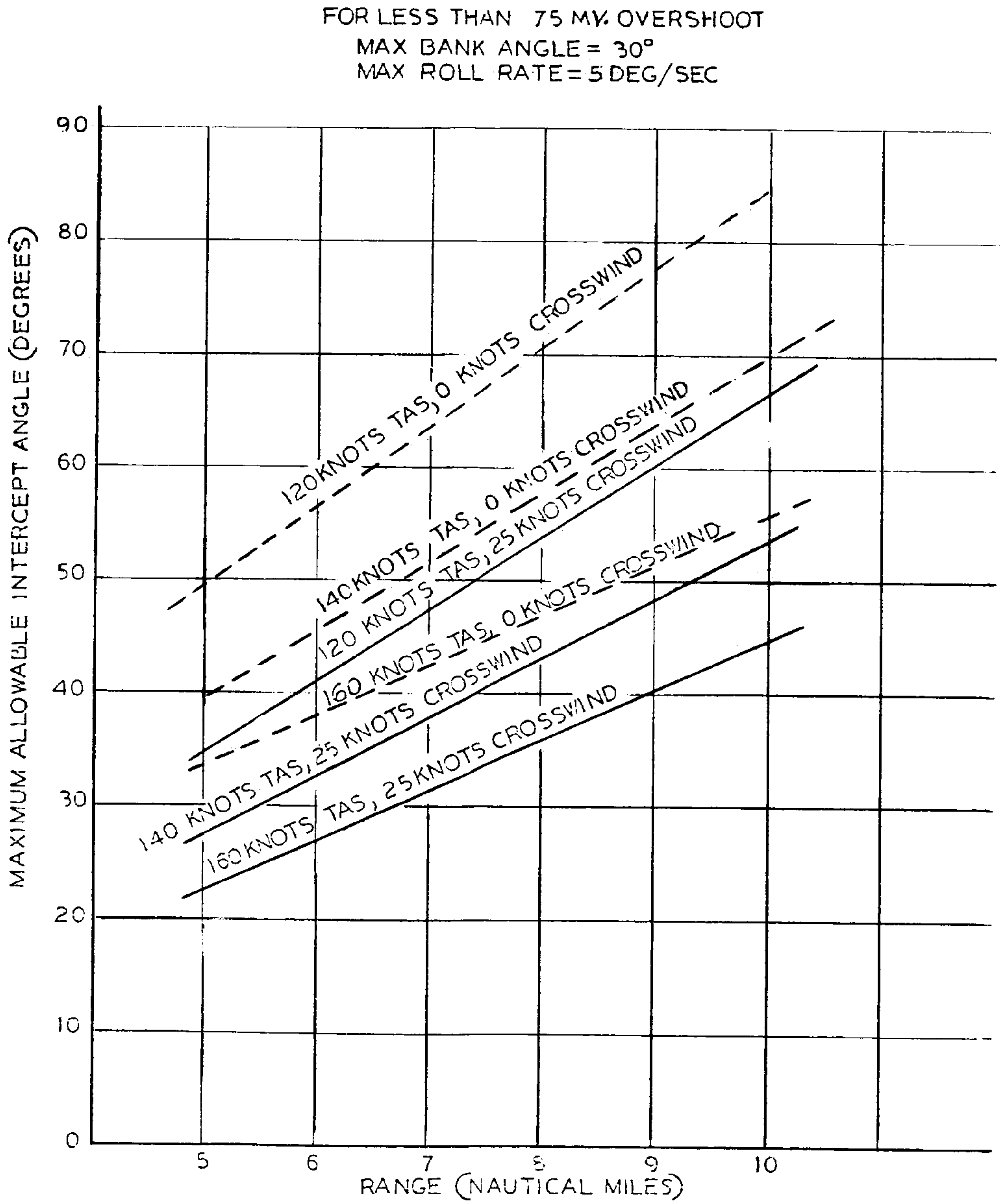


Figure 2-3
Maximum Allowable Intercept Angle Versus
Range from ILS Localizer Transmitter

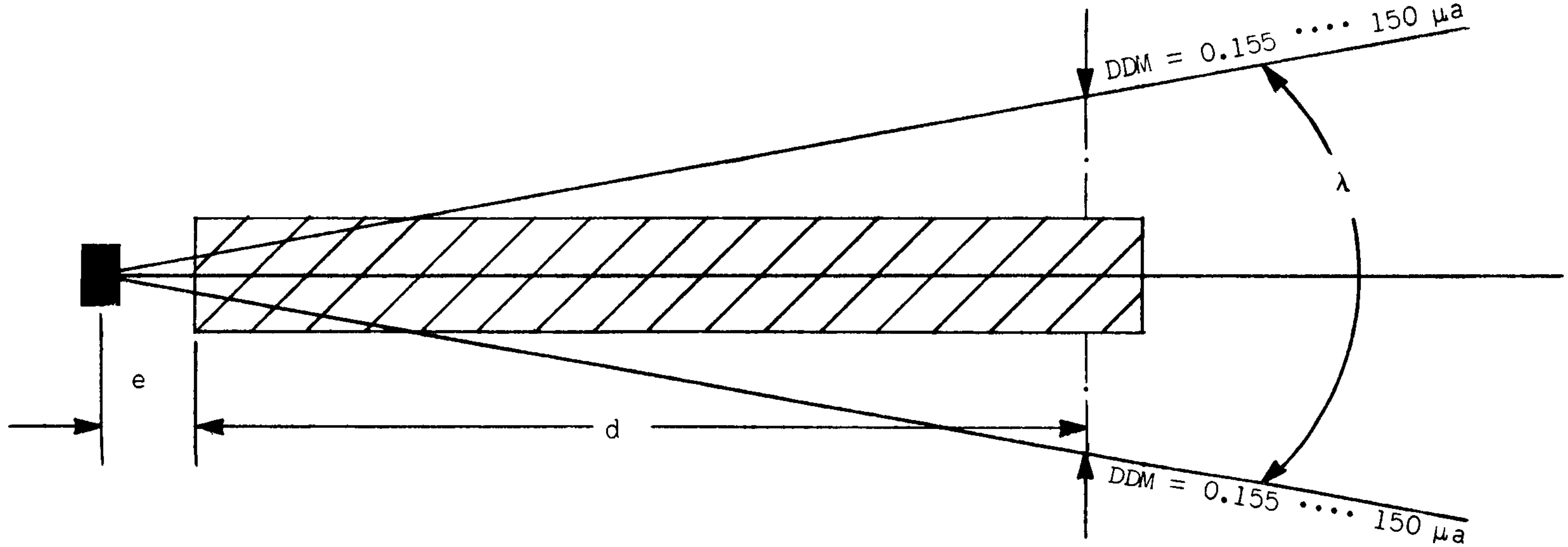
runway as possible. The automatic system can cope with a wide range of intercept angles and can give dead beat response if the aircraft intercepts the beam about 10 nautical miles out. It can also give excellent performance for intercepts that are closer in if the intercept angle is reduced proportionately. To cope with a wider range of intercept angles, the automatic system must be given more maneuvering authority. However, pilots of large turbojet aircraft do not like to see the autopilot command 40- to 60-degree bank angles at any time and especially not during the landing approach phase of flight. (Most passengers would prefer that neither the pilot or the autopilot produce large bank angles.) Autopilots, especially in the approach modes, have roll rate restrictions that the pilots are also anxious to retain. It is difficult to see how this type of problem can be resolved. To obtain good automatic approach performance and even good manually controlled ILS approaches, certain procedural restrictions regarding localizer intersections will have to be observed.

1.2 Localizer Beam Characteristics

There are several other aspects of the present day localizer beam system of lateral guidance that prevent ideal or optimum lateral control performance. We have discussed the beam capture phase because it is important that the initial turn on be accomplished with a minimum overshoot. The overshoot must be minimized but not at the expense of overdamping which would consume valuable range to go. This initial response is important because of the problems associated with the final stabilization on a path along the beam center. These problems will be discussed subsequently. First, let us examine how the localizer guidance concept inherently causes some of the control problems. In the previous discussion of the capture maneuver it was seen that the beam boundary occurs at a varying distance from the beam center. Hence, the start of the capture maneuver occurs closer to the beam center when the range is short. The short range situation is conducive to overshoots and overshoots are least tolerable close to the runway. The information regarding displacement from the beam center is provided by the localizer receivers as a signal proportional to angular displacement rather than linear displacement. That is, a given displacement from the centerline will produce a variable signal depending upon the distance from the transmitter. When the aircraft is 10 nautical miles from the transmitter, a fixed displacement from the centerline will result in a signal equal to one-half the signal magnitude occurring for that displacement at a 5 nautical mile range. The guidance system attempts to control position on the basis of such a signal. In effect, it always has a well-defined null position but it does not have a consistent measure of the actual deviation. This will obviously result in a variable response when proportional steering laws are used.

It is possible to compensate for this difficulty if range-to-go information were available. The use of this additional data (if it were available) would result in a more complex control system but one that provides more uniform response along the entire approach trajectory. However, it would be most undesirable if operational requirements necessitated this type of invariant response. For one thing, range-to-go information by itself would not be enough to define the actual beam displacement resulting from a given localizer signal. Localizer beams have been notoriously variable in their characteristics over the years. The recent imposition of new standards for lower minima operations and in anticipation of blind landing operations have improved the situation, but localizer geometry remains a variable from airport to airport. The reason for this is shown in figure 2-4. The localizer transmitter is located at a distance e behind the runway [usually about 304.80 meters (1000 feet)]. The reference beam will be specified in terms of its width at the runway threshold. (See reference 1 for a summary of ILS standards.) The threshold is defined in relation to the glide slope (which will be discussed later when the vertical guidance problems are reviewed). Consequently, the runway length will determine the localizer course sector angle ($\lambda/2$ in figure 2-4). Note that the full beam-width signal is defined as 150 μA (microamperes) from the localizer receiver when the DDM (difference in depth of modulation) is 0.155. Different runway lengths will result in course sector angles which vary as shown in figure 2-5. If we used range-to-go information and were attempting to compensate for the variable localizer sensitivity in microamperes per 0.30 meter (microamperes per foot), that compensation would be applicable to only one specific course angle. To take into account the variations in that angle, the system would have to be programmed separately for each airport. Figure 2-6 shows how the standardized localizer sensitivity varies in the vicinity of the runway threshold for various runway lengths.

Fortunately, it is possible to obtain excellent lateral control performance during automatic approaches despite these variations in localizer characteristics. (This is true providing reasonable procedural restrictions are observed.) We have dwelled on the variation in localizer sensitivity from airport to airport for an ulterior motive. It is often suggested that the Inertial Navigation System (INS) of the newer aircraft be used in conjunction with the ILS system during automatic approaches and landing. The most frequently suggested use of the INS is as a performance monitor in lieu of redundant ILS receivers and other equipment. Studies performed at Sperry Phoenix Company have verified that projected state-of-the-art INS equipment can approach the desired accuracy if we assume that a proper initializing phase is completed at the start of the landing approach. If the INS were used to monitor performance of the localizer receiver, it would have to compute the same instantaneous lateral position as defined by a properly operating receiver. However, the accuracy limitation is in the ILS rather than the INS. It is such factors as the variation in localizer beam sensitivity defined in figure 2-6 as well as beam bends and other inaccuracies which make the use of the INS as a pseudo localizer impractical.



Localizer Angle

$$\lambda = \frac{1400}{(d+e)} \ (57.3) \text{ Deg.}$$

Course Sector Angle = $\lambda/2$

* Threshold occurs at intersection of glide slope with altitude plane 50 ft. above runway.

Figure 2-4
ILS Localizer Geometry

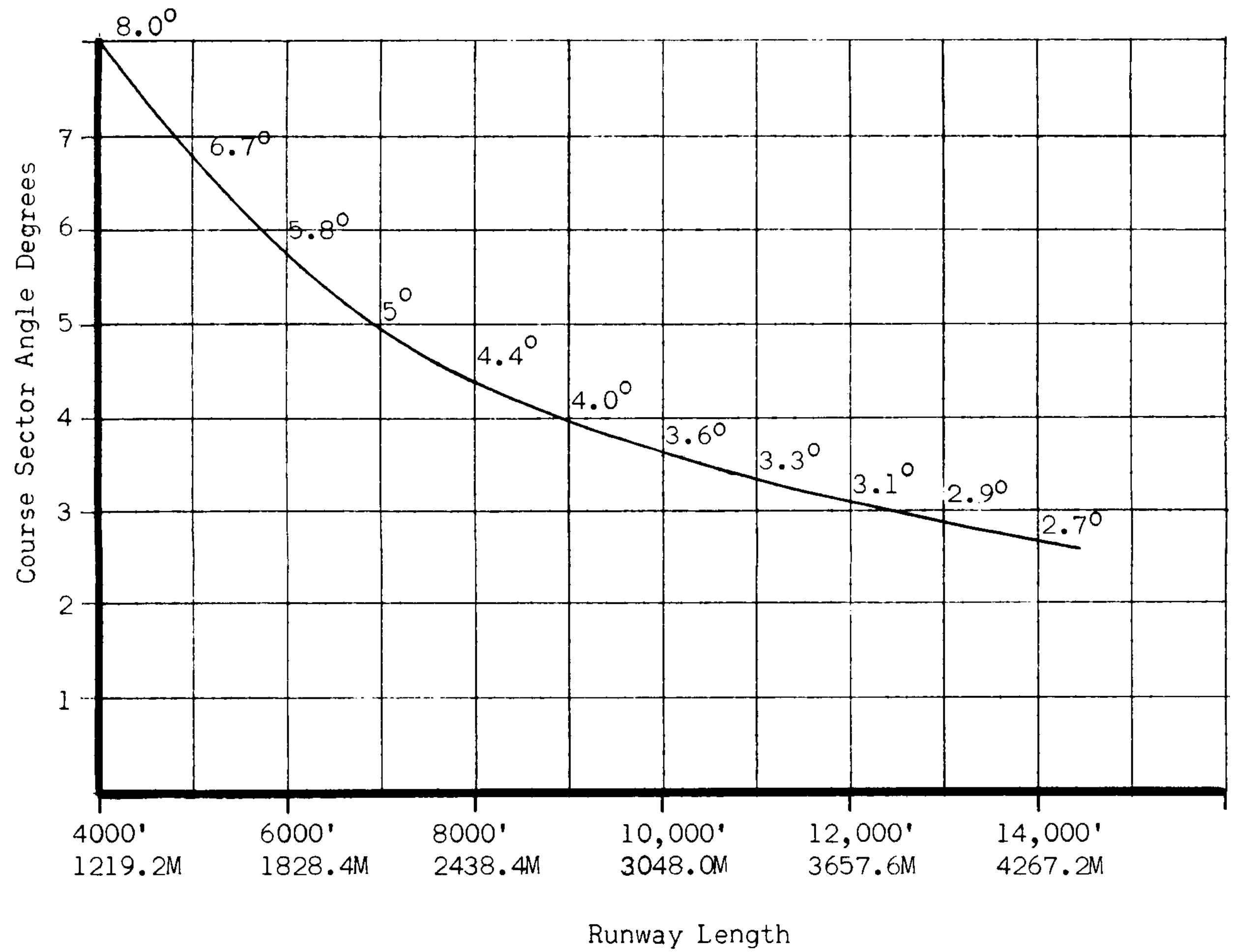


Figure 2-5
Course Sector Angle Versus Runway
Length for Constant Course Sector Width
of 700 Feet at Threshold

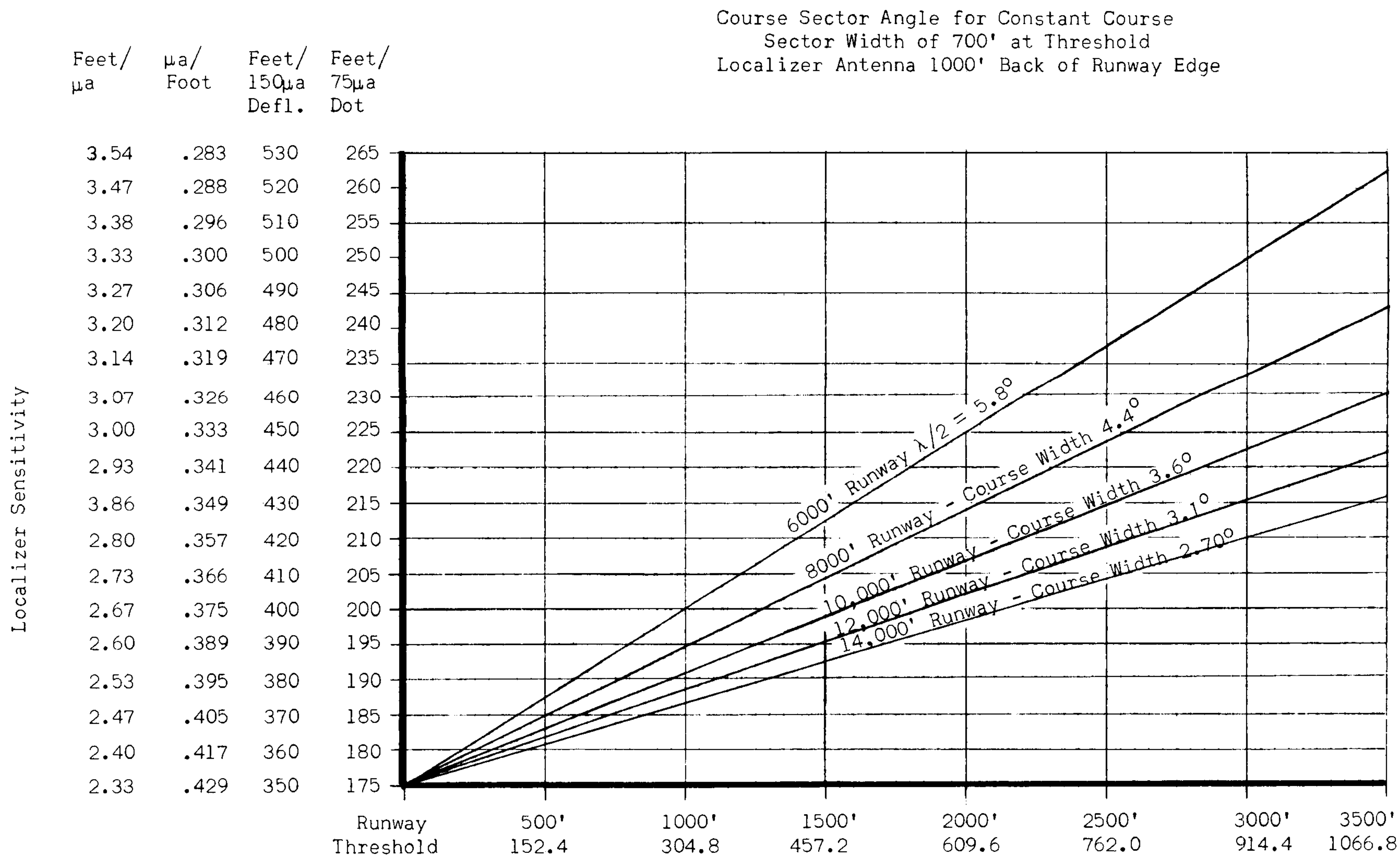


Figure 2-6
Localizer Deviation Constants for
Different Runway Lengths

1.3 Lateral Stabilization

There are many lateral flight path control configurations used with the various autopilot-aircraft combinations operating today. While these systems appear to have many differences, if we look at their implementations or even their functional block diagrams, they are all essentially identical from the standpoint of the control laws used. In general, the problem is to reduce the aircraft's deviation, y, from the reference flight path to zero. (See figure 2-7.) A steering command to correct for a path deviation must produce a bank angle. Since a bank angle will produce a proportional side acceleration, $\ddot{y}$, it is immediately apparent that a steering law which attempts to produce a lateral acceleration proportional to lateral displacement will be unstable. For stability, the bank angle steering command, ϕ_c, must be of the form

$$\phi_c = k_1[y + k_{\dot{y}}\dot{y}] \tag{2-1}$$

In practice, it is often necessary to add an integral term. The reasons for this will be discussed later. Thus the usual linearized representation of the lateral steering law is

$$\phi_c = k_1[y + k_{\dot{y}}\dot{y} + k_I\int y dt] \tag{2-2}$$

It should be emphasized that this linearized representation of the steering law is highly simplified and is being shown in this manner only to illustrate the nature of the stabilization problem. In reality, the various operations on y and its time derivative include variable limits (nonlinear elements) and fairly complex linear filters. Figure 2-7 illustrates how this steering law enters into the overall stability problem which includes the roll stabilization dynamics and the aircraft's turning kinematics. The aircraft kinematics are represented by a transfer function which assumes the small perturbation turn rate equation

$$\dot{\psi} \approx \frac{g}{V}\phi \tag{2-3}$$

and the sideslip = 0 and crosswind = 0 representation of the lateral velocity as

$$\dot{y} = V\psi \text{ and } \ddot{y} = V\dot{\psi} \tag{2-4}$$

so that

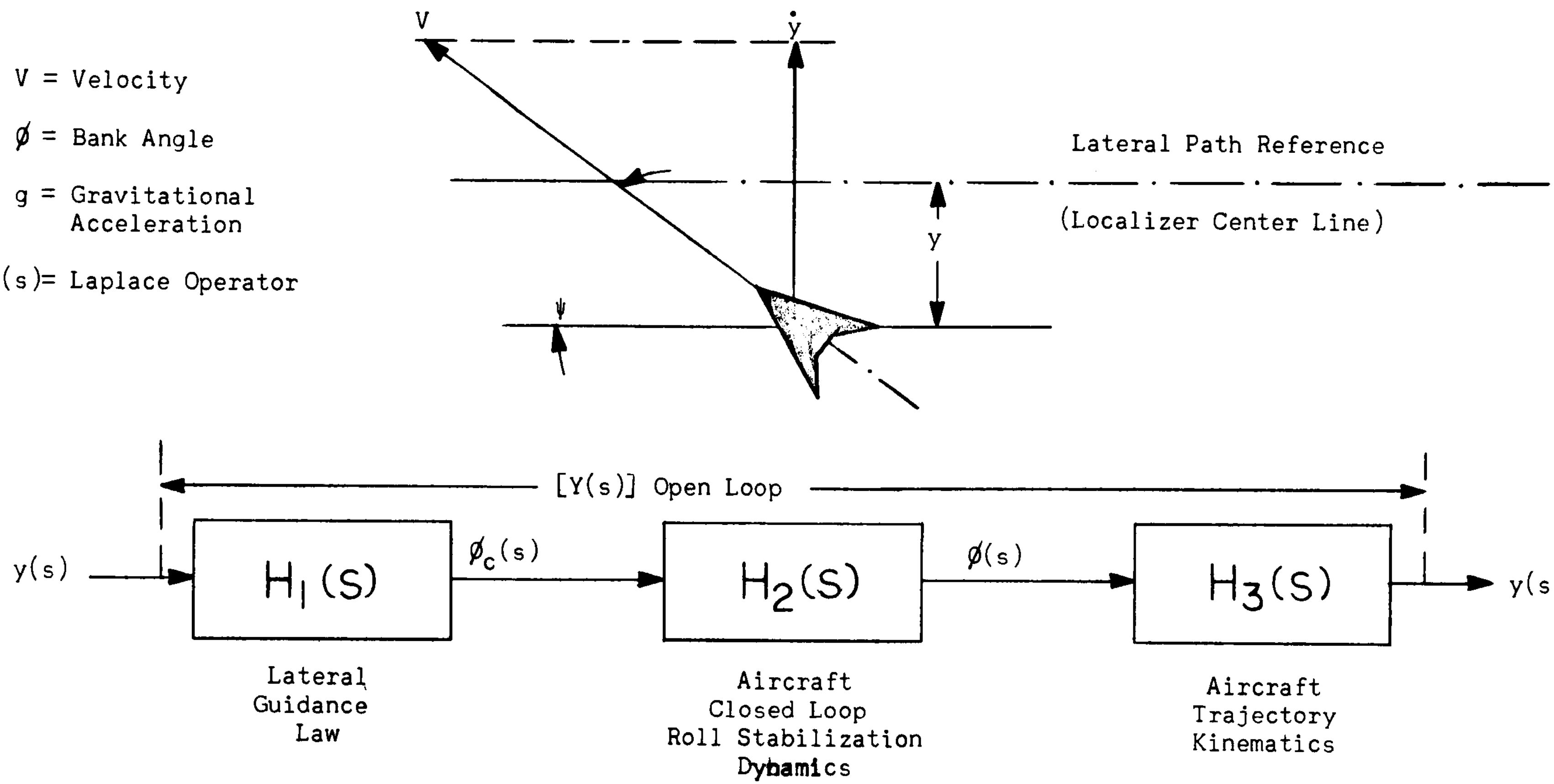
$$\phi = \frac{V\dot{\psi}}{g} = \frac{\ddot{y}}{g} \tag{2-5}$$

V = Velocity

ϕ = Bank Angle

g = Gravitational Acceleration

(s)= Laplace Operator



$$\left[\frac{\phi_c(s)}{y(s)}\right] = K_1\left(1 + K_{\dot{y}}\, s + \frac{K_I}{s}\right) \qquad \left[\frac{\phi(s)}{\phi_c(s)}\right] \qquad \left[\frac{y(s)}{\phi(s)}\right] = \frac{g}{s^2}$$

Based on Small Perturbation Approximation

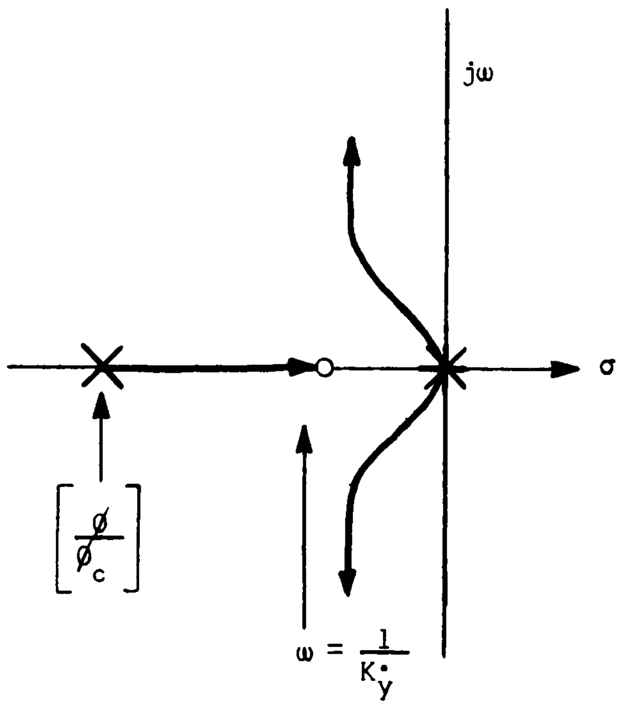
$$\phi = \left(\frac{V}{g}\right)\dot{\psi}$$

Figure 2-7
Lateral Flight Path Stability Parameters

The stability of the y control loop can therefore be analyzed by considering the sequence of transfer functions for the lateral guidance law, $H_1(S)$, the aircraft's closed loop roll stabilization dynamics, $H_2(S)$, and the trajectory kinematics, $H_3(S)$, as shown in figure 2-7. A qualitative view of the stability problem without and with the steering law integral term is shown in figures 2-8(a) and 2-8(b), respectively. In these root loci, the roll stabilization dynamics are assumed to be a first-order lag. This is also a simplification. The phase lag of the roll response is a critical factor in the flight path stability problem. Indeed, the control parameters are generally chosen to cope with the nonlinear contributions of the roll dynamics such as the roll-rate limits. Figure 2-8(a) shows that an oscillatory tendency exists even without integral control. The gain variable in the loci is the forward loop gain, K_1. Note that when the gain is near zero, the characteristic response will be that of a very low frequency, undamped oscillation. This assumes that the $\dot{y}$ term and y term are both at very low gains. A system which obtains its $\dot{y}$ by differentiating the localizer y signal can be represented by this root locus. The low gain oscillatory case will be characteristic of the response a great distance from the transmitter. As the distance to the transmitter decreases, the damping will improve and the control frequency will increase; but eventually the damping will again tend to decrease. If the aircraft is flown on the localizer along a limited distance such as that associated with a typical approach (as shown in figures 2-1 and 2-2), then the range of damping ratios and frequencies can be bounded to acceptable levels.

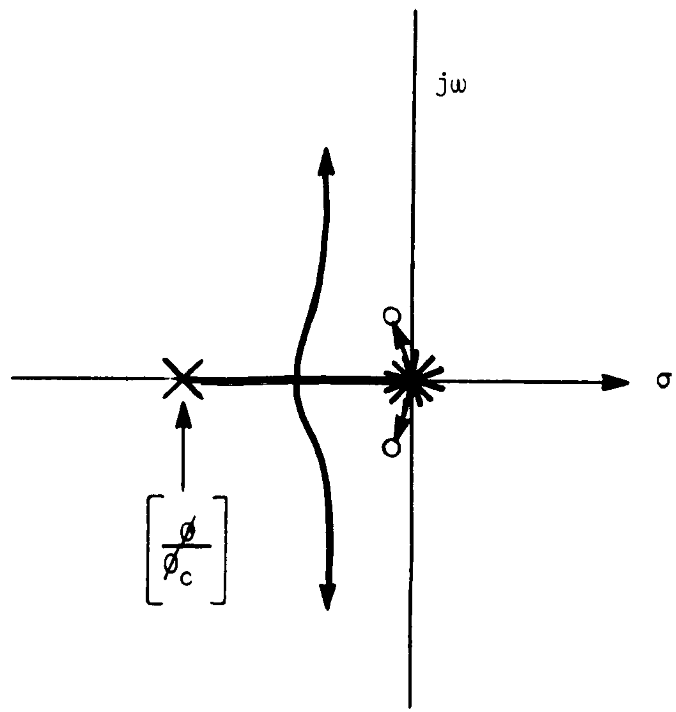
If more uniform response dynamics are desired, compensation can be provided for the localizer sensitivity variation, K_1, with distance from the transmitter. This compensation could be obtained from several sources since a precise adjustment of K_1 is not necessary. One source might be a radio altimeter which could provide information for lowering gains during the final descent phase. The fact that terrain irregularities would introduce variables is not, in general, too important because the terrain effect is not significant in relation to the crude accuracies needed. Also, marker beacons signifying that the aircraft has reached a specified distance from the runway can be used to initiate open loop gain reduction programs. The most satisfactory source of gain reduction data could be derived from precision DME (distance measuring equipment), but DME has thus far been avoided as a source of data for all weather landing systems.

Figure 2-8(b) shows the root loci of the lateral stabilization system when the integral term is used in the control law. A new low frequency oscillatory tendency is noted. The damping of this new mode will depend upon the ratio of path integral to displacement gain, K_I. A value of K_I as high as 0.05 is usually excessive from the point of view of stability. Also, it is interesting to note the frequency (or period) of this new mode. The characteristic frequency of the basic lateral path stabilization mode occurring without the



$$\left[Y(s)\right]_{\substack{\text{Open}\\\text{Loop}}} = K_1 g \, \frac{[K_{\dot{y}} s + 1]}{s^2} \left[\frac{\phi}{\phi_c}\right]$$

(a) Lateral Path Control
Root Locus Without
Integral Control



$$\left[Y(s)\right]_{\substack{\text{Open}\\\text{Loop}}} = K_1 g \, \frac{[K_I + s + K_{\dot{y}} s^2]}{s^3} \left[\frac{\phi}{\phi_c}\right]$$

(b) Lateral Path Control
Root Locus With
Integral Control

Figure 2-8
Stability Analysis of Lateral Flight
Path Control

integrator varies with the sensitivity, K_1. Typical periods range from about 30 to 70 seconds. The integral mode period may be several minutes. Quite often, the entire length of the localizer flight corresponds to one-half wavelength of the lateral integration mode. One may therefore ask why such a destabilizing term is used in the control law. The answer to this question leads to the core of the localizer tracking accuracy problem. It can readily be shown how the lateral integrator is theoretically justifiable, but its proper application is the source of many problems. It is in the technique of programming the lateral integrator that many automatic approach systems display subtle, but often important differences. Indeed, some designers and manufacturers of automatic approach systems have concluded that the problems associated with the use of an integrator outweigh its advantages. They therefore accept certain performance compromises such as susceptibility to beam noise and perhaps larger errors in windshear, but they gain in system simplicity and flexibility.

To appreciate the reasons for using the lateral integrator, refer again to figure 2-7. Note that the open loop transfer function

$$[Y(s)]_{\substack{\text{Open}\\ \text{Loop}}} = K_1(1 + K_{\dot{y}}s + \frac{K_I}{s}) \left[\frac{\phi(s)}{\phi_c(s)}\right]\frac{g}{s^2} \qquad (2\text{-}6)$$

is completely independent of aircraft velocity (except to the extent that the roll stabilization dynamics are dependent upon aircraft speed). For good performance, the value of $K_{\dot{y}}$ is in the range of 20 to 40. If a value of $K_{\dot{y}} = 40$ is used, it means that a 0.30 meter/second (1.0 foot/second) path deviation rate will command 40 times the bank angle that would be commanded by a 0.30 meter (1.0 foot) displacement. If we obtained $\dot{y}$ by differentiating the y signal, even an insignificant level of beam noise could saturate the differentiator. To use beam rate $\dot{y}$, the data must be filtered and consequently the bandwidth of the $\dot{y}$ data is compromised. The amount of filtering needed depends upon the beam noise characteristic. Filtering, which affects the bandwidth of the $\dot{y}$ signal, influences the attainable loop gain, K_1, and hence the tightness of the localizer tracking.

There are other methods of measuring $\dot{y}$. The most convenient method and one that has found the most widespread application is the use of aircraft heading deviation from the runway centerline azimuth. For small angles, as shown in figure 2-8

$$y = V\psi \qquad (2\text{-}7)$$

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$$\dot{y} = V\psi \qquad (2\text{-}7)$$

Thus, the basic control law may be written as

$$\phi_c = K_1(y + K_{\dot{y}}V\psi) = K_1(y + K\psi) \qquad (2\text{-}8)$$

where

$$K = K_{\dot{y}}V$$

Now, suppose a crosswind existed such that a 10-degree crab angle is required to keep the velocity vector aligned with the localizer centerline. If the aircraft velocity were 160 knots (270 FPS), the absolute minimum standoff error obtainable can be calculated as follows:

$$K_{\dot{y}} = 40$$

$$V = 270 \text{ FPS}$$

$$\psi = 10/57.3 = 0.1745 \text{ radian}$$

$$y = K_{\dot{y}}V\psi = (40)(270)\left(\frac{10}{57.3}\right) = 575.07 \text{ meters (1890 feet)}$$

Reference to the localizer geometry [for a 3048 meters (10,000 feet) runway] described in figures 2-4 through 2-6, shows that a deviation of 575.07 meters (1890 feet) will correspond to the outer boundary of the localizer beam at a distance from the transmitter of 9174.48 meters (30,100 feet; 4.94 nautical miles). Thus, this crab angle requirement will cause the aircraft to depart completely from the localizer boundary when the aircraft is beyond 4.94 nautical miles from the transmitter. All guidance will be lost beyond that point.

In recognition of this problem, the designers of automatic approach systems who conclude that there is sufficient advantage to the use of heading as a lateral damping signal also include some means of compensating for the crab angle problem. The usual method for autopilot couplers is to use the lateral-path integrator. For manual approach systems using a flight director, an integration in the forward path is never acceptable, but an equivalent compensation is obtained by adding a high-pass filter (washout) with a time constant of about 30 seconds to the heading error signal. The use of a washout in this manner produces a low frequency destabilizing effect that is analogous to that of the lateral integrator.

The lateral integration function is used to provide the steady-state signal that will bias the crab angle heading error. In effect, the history of the lateral displacement error provides the corrective signal to yield a zero bank command when the aircraft's velocity vector is perfectly aligned with the center of the localizer beam. If the crosswind were variable (windshear), the steady-state solution will require a changing crab angle. This rate of change of heading is provided by a proportional bank angle. In an idealized situation, the lateral integrator can also provide the proper bank angle command to produce the yaw rate corresponding to perfect windshear compensation. Idealized refers to the conditions under which the steady-state solution prevails. Unfortunately, in the case of lateral approach path guidance, the steady state can never be attained because of the extremely long periods involved in that particular dynamic process. The flight path response is dictated largely by disturbance dynamics and initial conditions. The lateral path integrator may be viewed as working toward a desired steady-state solution, but never quite reaching its objective.

There are many things which must be done to help the integrator achieve its accuracy objectives without introducing destabilizing long period transients. First, it should not integrate the path error during the beam capture sequence. The integration should occur only when the aircraft has nearly acquired its final path. This implies that the steering law must adapt to the aircraft's instantaneous state. Most automatic approach systems in use today continuously monitor the status of y and $\dot{y}$ signals and adjust the steering law accordingly. The more sophisticated this adjustment, the greater is the possibility of accurate path control under a variety of initial conditions and disturbances. The most recent systems will use many sources of data as inputs to the steering law. The gains of the different input contributions are continuously programmed as a function of the error parameters and as a function of additional data regarding proximity to the terminal conditions. A typical steering law in a modern system designed for lower minima is of the form

$$\phi_c = K_1 y G_1(s) + K_2 y G_2(s) + K_3 G_3(s) + K_4 \phi G_4(s) + K_5 \psi \qquad (2\text{-}9)$$

where

$G_1(s)$ is a lag filter

$G_2(s)$ is a derivative function with high frequency lag filters

$G_3(s)$ is an integrator

$G_4(s)$ is a dynamic integrator (large lag)

Figure 2-9 illustrates how this information is processed into variable weighting functions that are controlled by a steering law programmer. This type of system continuously synthesizes an optimum $\dot{y}$ signal from the rate of the beam displacement, runway heading deviation, and the dynamic integration of bank angle. (Note that bank angle, ϕ, is proportional to $\ddot{y}$ when the aircraft is aligned with the beam.) The various gains and dynamic shaping functions are adjusted to minimize the influence of the crosswind susceptible heading signal at the terminal phase of flight, and to optimize rapid capture of the desired

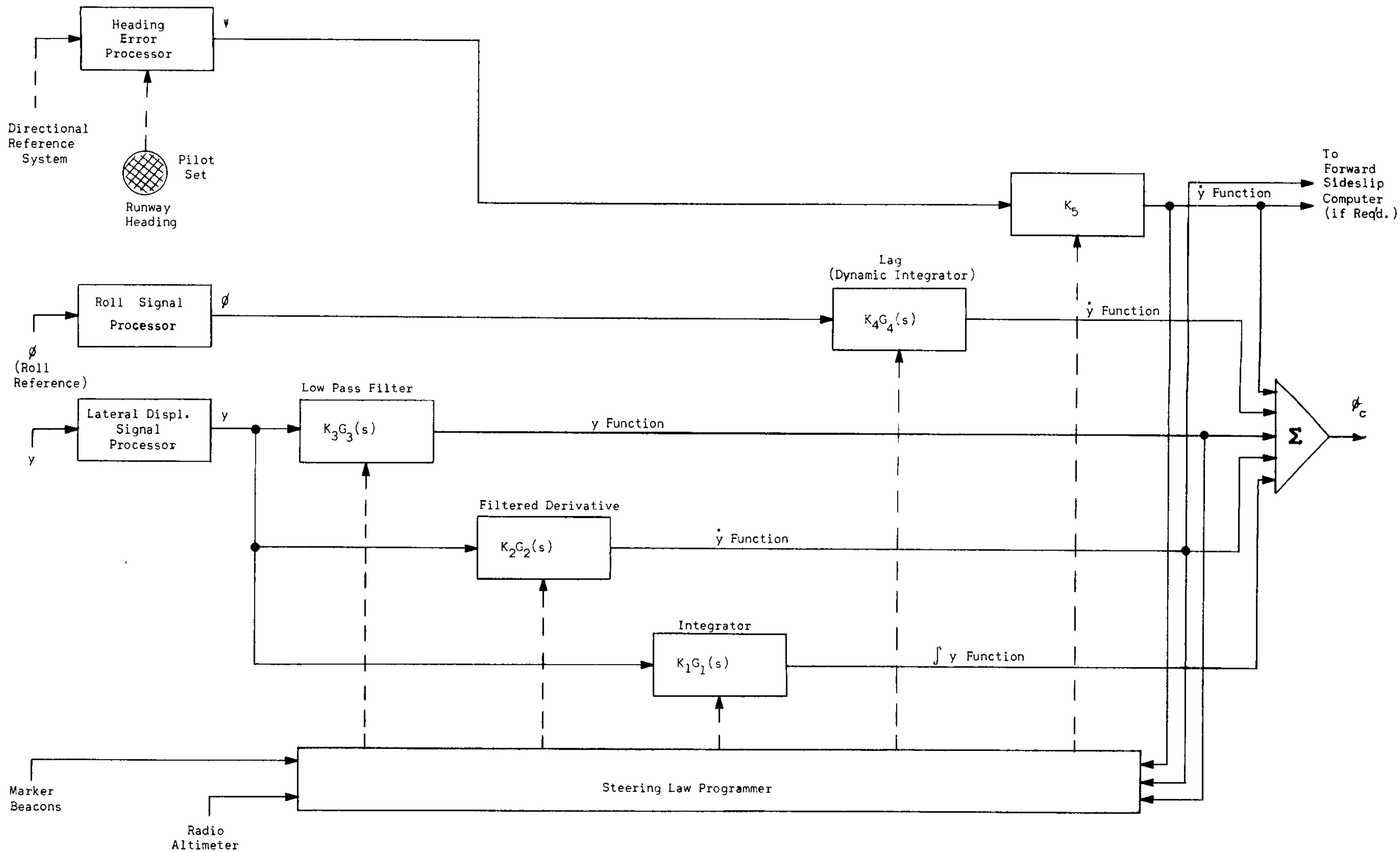


Figure 2-9
Modern Automatic Approach System Lateral
Flight Path Computer

flight path during the initial phases of the approach. Examples of the operation of such a system are illustrated in figures 2-10 and 2-11. These figures are complete histories of the significant parameters associated with actual automatic approaches and landing performed during the FAA Category II Certification Tests of the DC-8 and the SP-30A autopilot. The two figures shown are reproduced from reference 2 which documents test results at five airports (Oakland, Stockton, San Francisco, Long Beach, and Ontario) including automatic and flight director demonstrations under varying weight; cg, wind, engine out, hardover failure simulations, and control system tolerance situations. Of interest in these figures are the localizer position errors, the crosswind and crab angle conditions as the aircraft approaches the runway. The localizer trace includes a shaded region which corresponds to the accuracy window required for certification. Figure 2-10 shows an approach starting with a 90-degree intercept while figure 2-11 shows an approach starting about 3 nautical miles closer in but with a 45-degree intercept. The steering laws used were similar to those shown on figure 2-9, except that the roll function $K_4G_4(s)$ was not used. The various gains were programmed downward during the final descent with the radio altimeter used as a source of program data. The steering law parameters during beam capture were programmed as a function of the internally sensed position and rate. The significant point made by these recordings is that the slight performance differences were determined by the initial conditions and the disturbances. It is extremely difficult to generalize on the relative merits of even gross changes in the steering laws. If we could create a realistic analytical model for this problem, we would probably find a very broad optimum. The problem, of course, is how to define a quantitative criterion of "optimum".

1.4 The Rollout Alignment Maneuver

In the above discussions, the problem has been to align the aircraft's ground track with the runway centerline. In the presence of a crosswind, the aircraft assumes a crabbed attitude to prevent lateral drift. This is not the only method of cancelling crosswind drifts. An alternate approach implied by figure 2-9 uses the so-called forward-slip maneuver. The runway heading deviation signal and the path displacement are gradually phased into a rudder control channel to cause the aircraft to align its heading with the localizer centerline. This results in a steady-state solution requiring the dropping of the upwind wing and the holding of a steady sideslip angle by means of opposite rudder deflection. With this system, a decrab maneuver would not be needed prior to touchdown. The aircraft would contact the runway with the upwind wheel. The initial contact would produce the corrective rolling moment that restores the wing-level condition.

The decrab maneuver is needed to reduce sideloading on the landing gear if the approach is made with a crab angle. Some aircraft have been designed

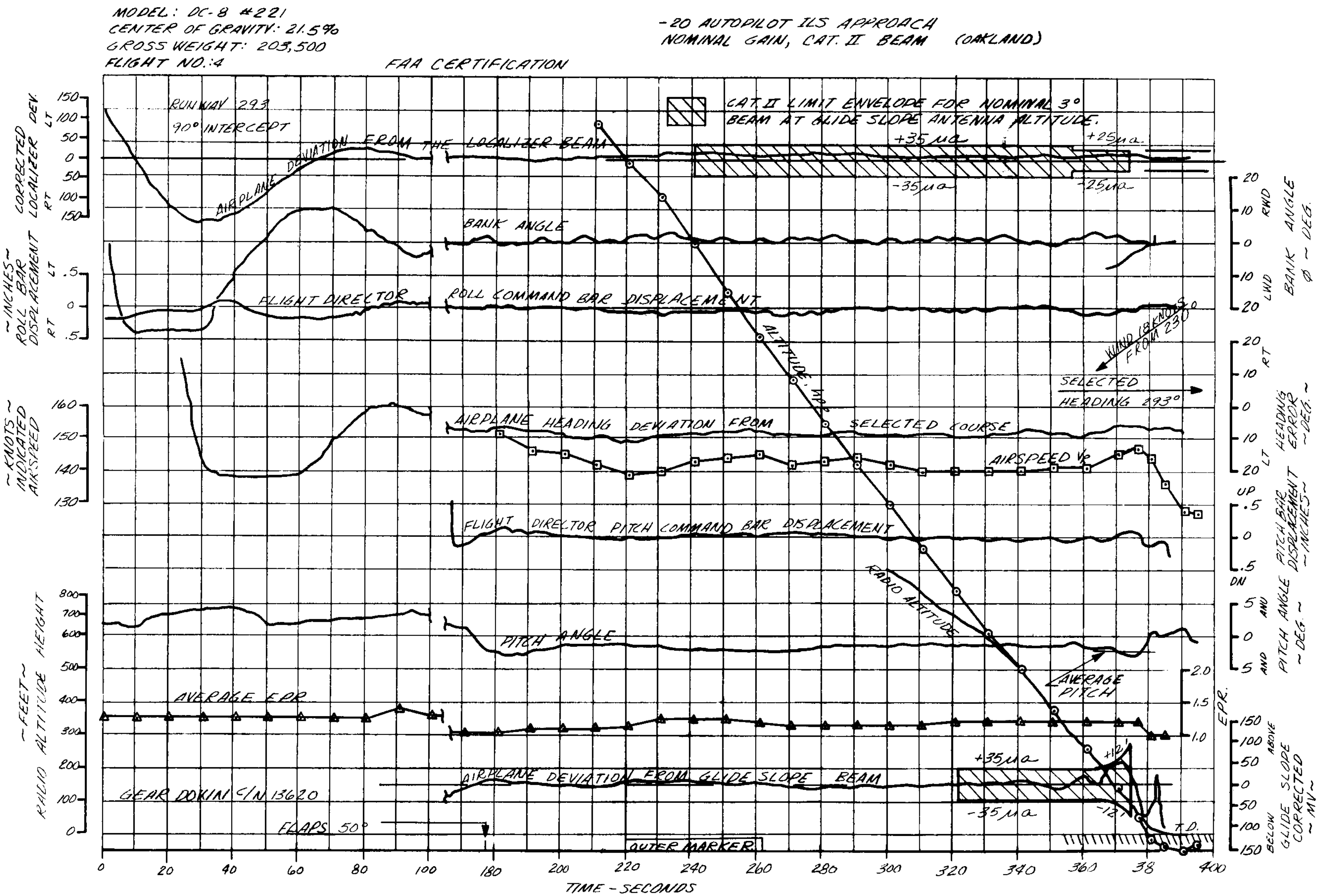


Figure 2-10
Flight Recording - DC-8 Automatic Approach and Landing (90° Localizer Intercept)

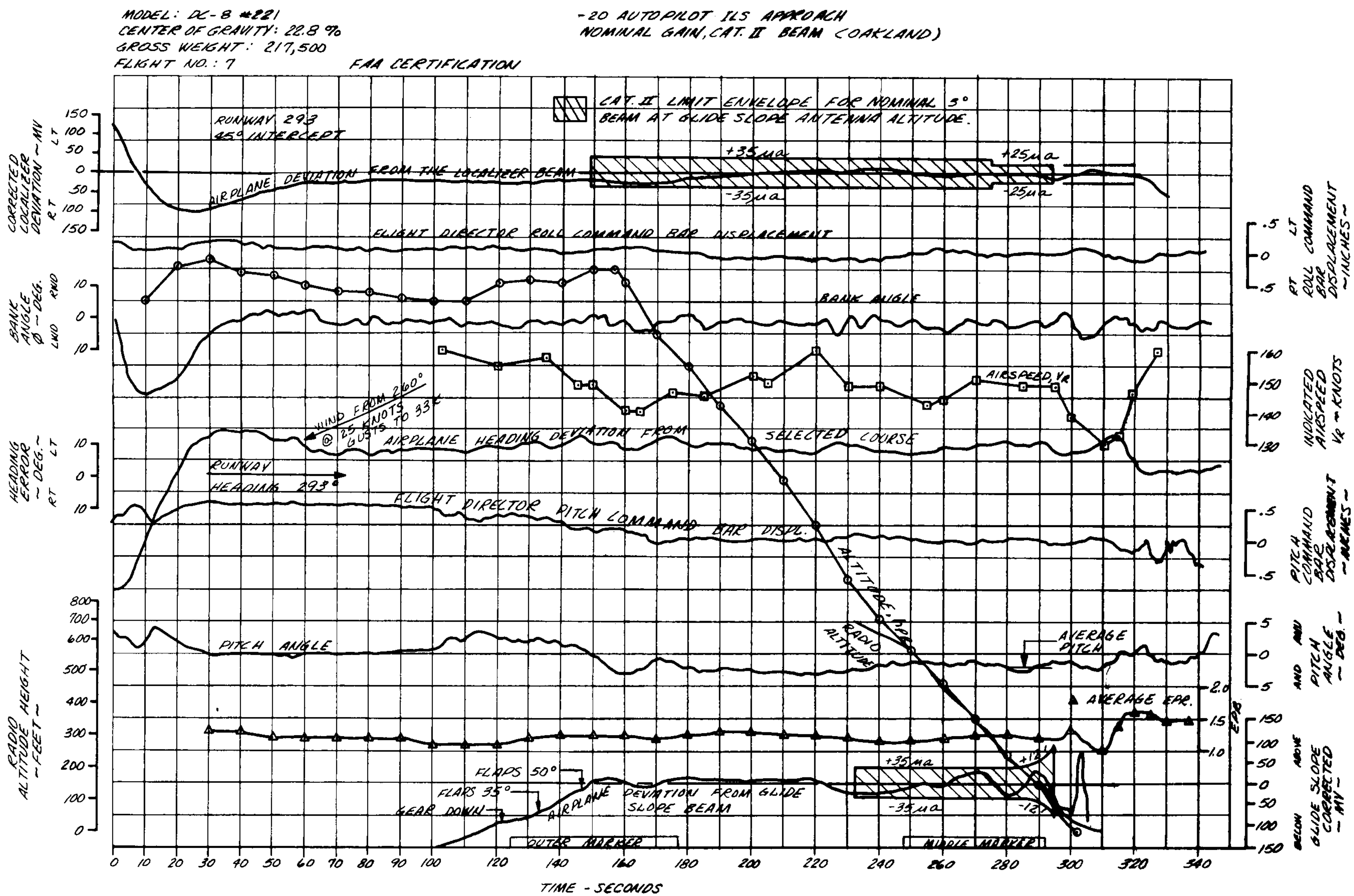


Figure 2-11
Flight Recording - DC-8 Automatic Approach and Landing (45° Localizer Intercept)

with steerable landing gear that can be positioned to align with a runway centerline despite the crab attitude approach. Most aircraft landing gear is designed to withstand fairly high sideloads which would occur if the aircraft were not decrabbed in a moderate crosswind. Moreover, the relationship between the aircraft cg and the landing gear configuration is such that azimuth alignment is automatically produced by the side force on the gear at the time of runway contact. The double track landing gear configurations of the new larger class of aircraft including the SST will probably not provide this type of favorable azimuth alignment when the runway is contacted. Decrab maneuvers, however, are quite easily implemented in automatic systems, but performance is very sensitive to slight variations in the vertical trajectory. An automatic system will typically command a decrab yaw rotation when the landing gear altitude is about 2.44 meters (8 feet) above the runway. The result is almost entirely dependent upon the time which will elapse between the initial rudder kick and touchdown. If the flareout is on the high side with a relatively long floating flare, the decrab can result in a high lateral drift velocity and excessive displacement from the runway centerline. A short flareout may not provide adequate time to rotate the aircraft heading, but this error will not cause any lateral touchdown dispersion problems.

The critical problem for the automatic or manual decrab is therefore the timing of the maneuver. An automatic system must receive its decrab command from a radio altimeter. Altitude errors of 0.61 meter (2 feet) at this time can cause significant variations in the touchdown parameters. This is one of the reasons why a large segment of the participants in the AWL development programs tends to favor a manual rather than an automatic decrab maneuver. Another reason for uncertainty in the desired procedure for rollout relates to a prevailing preference for the forward slip approach in crosswinds. This preference is largely based on manual operating procedures. For example, the following paragraph is extracted from Pan American Airways Operating Technique Manual for the Boeing 707 300/300B. (The same paragraph appears in PAA's Operating Techniques for the DC-8.)

> Cross-wind
>
> Proceed with landing and before starting flareout, remove the crab angle and dip the wing slightly into the wind to eliminate drift. Contact is made on one wheel; align the nosewheel down the center of the runway with the rudder and control drift with the aileron. Kicking drift out too rapidly at the last second will induce Dutch Roll.
>
> Caution - Do not scrape pod.

Another reason often suggested for preferring the forward slip approach relates to its possible advantage when heads-up displays are used. Aligning the aircraft heading with the runway provides a more desirable alignment of the windscreen projections with the real world view of the approaching runway.

There are no clear superiorities of either alignment concept when all factors are considered. The forward slip approach is often advocated because it might reduce the rudder authority needed. If such an advantage exists, it is very slight since it requires about as much rudder to fly the cross-controlled sideslipping approach as it does to add the sideslip angle immediately prior to touchdown. The forward slip is often criticized because of excessive roll angle requirements and the danger of scraping the pod on landing. The following simplified analysis demonstrates that roll angle requirements are actually not too severe:

(All symbols used are per the nomenclature of "Dynamics of the Airframe", Bureau of Avionics Report AE-61-4II)(Reference 3)

For side force balance (no cross-course acceleration)

$$C_{y_\beta}\beta - C_{y_{\delta_R}}\delta_R = C_L \emptyset \tag{2-10}$$

For zero yawing acceleration,

$$C_{n_\beta}\beta = C_{n_{\delta_R}}\delta_R \tag{2-11}$$

Thus

$$\beta\left[C_{y_\beta} - \frac{C_{n_\beta}}{C_{n_{\delta_R}}} C_{y_{\delta_R}}\right] = C_L \emptyset \approx \frac{\emptyset}{q(W/S)} \tag{2-12}$$

or

$$\emptyset = \frac{\beta q}{(W/S)}\left[C_{y_\beta} - \frac{C_{n_\beta}}{C_{n_{\delta_R}}} C_{y_{\delta_R}}\right] \tag{2-13}$$

$$= \frac{V\binom{\text{Cross-}}{\text{wind}} q}{V(W/S)}\left[C_{y_\beta} - \frac{C_{n_\beta}}{C_{n_{\delta_R}}} C_{y_{\delta_R}}\right] \tag{2-13}$$

This bank angle, Ø, is required to compensate for the sideslipping force. It is plotted in figure 2-12 as a function of crosswind for various aircraft at a final speed of 120 knots. If bank angles greater than 5 degrees are considered objectionable, only the Boeing 747 at crosswinds in excess of about 25 knots appears to indicate a problem in this regard.

Most automatic landing systems that are being implemented today use the decrab rollout alignment, primarily because of its simplicity advantages. (The forward slip method is being implemented in the DC-9.) The following is a summary of advantages and disadvantages of both methods:

<u>DECRAB MANEUVER ROLLOUT ALIGNMENT</u>

<u>Advantages</u>

- Simplest system to implement
- Avoids wing-down hazard at touchdown
- Permits zero sideslip approach - hence, minimum airspeed required

<u>Disadvantages</u>

- Susceptible to large lateral position and velocity discrepancies for variable flareout flight paths
- Compromises heads-up display computer complexity and utility of windscreen presentation of data
- Large authority rudder servo required
- Requires extremely accurate radio altimeter programmer

<u>FORWARD-SLIP MANEUVER ROLLOUT ALIGNMENT</u>

<u>Advantages</u>

- No yawing maneuver needed prior to touchdown - roll to level is automatic on contact
- Runway view is good - nose of aircraft is aligned with flight path

<u>Disadvantages</u>

- Implementation more complex than decrab technique
- If slip maneuver is implemented during the final phase of the approach, a transient disturbance is created that introduces lateral errors at a critical part of the flight

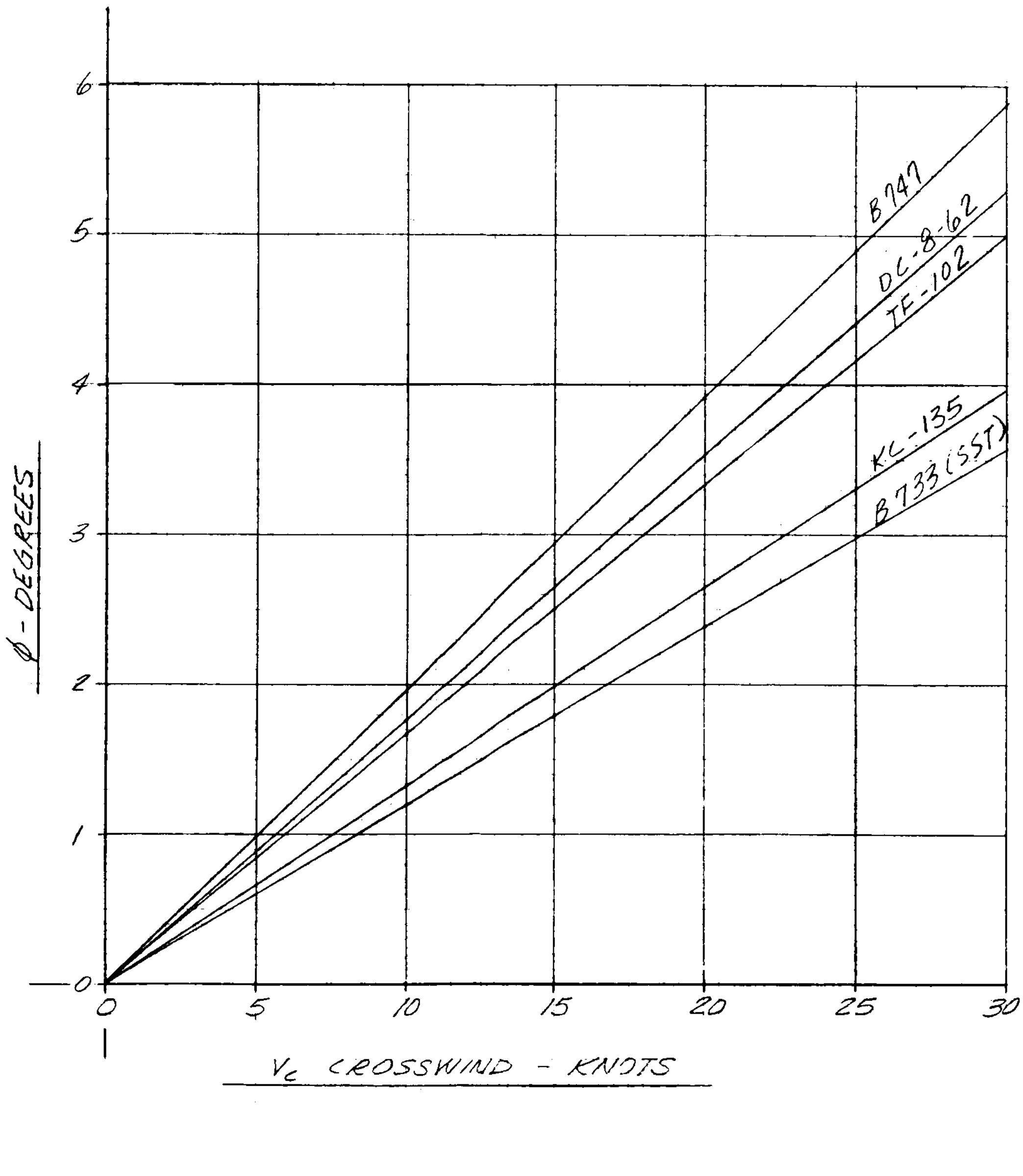


Figure 2-12
Bank Angles for Cross-Controlled Sideslip Approach, Five Turbojet Aircraft at 120 Knots Final Speed

- Enhances utility of heads-up display

- Higher approach speeds required to fly a sideslipping configuration

- Excessive bank angles may be required for some aircraft at high crosswinds

2. Vertical Flight Path Control

2.1 Glide Slope Characteristics

The use of the ILS glide slope as a means of directing the aircraft to a precise position on the landing runway is illustrated in figure 2-1. As in the case of the localizer beam and lateral guidance, there is nothing fundamentally unique about the ILS radio technique. Any equivalent method that can define a precise flight path reference could be used and indeed there are certain deficiencies in the present ILS glide slope systems that should be eliminated if we were to configure an alternate system. Major problems associated with the use of ILS glidepath systems relate to variations in the geometric relationships to the runway, noise problems (beam bends) and the converging nature of the beam. As in the case of the localizer, the convergent beam results in a variable sensitivity which poses control stability problems. However, the glide slope problem is much more severe in this regard because the sensitivity actually approaches infinity in a region where flight path guidance information is still needed. For this reason, gain programming of the glide slope signals is far more critical than for the localizer lateral steering case. Also, a transition to some other reference is required by the time flareout altitudes are reached. The radio altimeter is the usual final reference when the ILS glide slope is used, but there are other guidance systems which can provide a continuous flight path reference through the flareout phase.

Some of the ILS glide slope geometry variables are being eliminated as more stringent standards are imposed on airport facilities for lower minima operations. In addition to improvements in beam accuracy, the trend is toward standardizing locations of the glidepath intersection with the runway and reducing the range of angles permissible for glide slopes (reference 2). However, even the more restrictive range of glide angles (2.5 to 3.0 degrees) corresponds to significant variations when very precise and repeatable performance is an objective. Figures 2-13 and 2-14 illustrate the geometric properties of the 2.5-and 3.0-degree beam respectively. Note the extreme sensitivity in the 15.24 meter (50 feet) to 30.48 meters (100 feet) altitude region. On the 3.0-degree

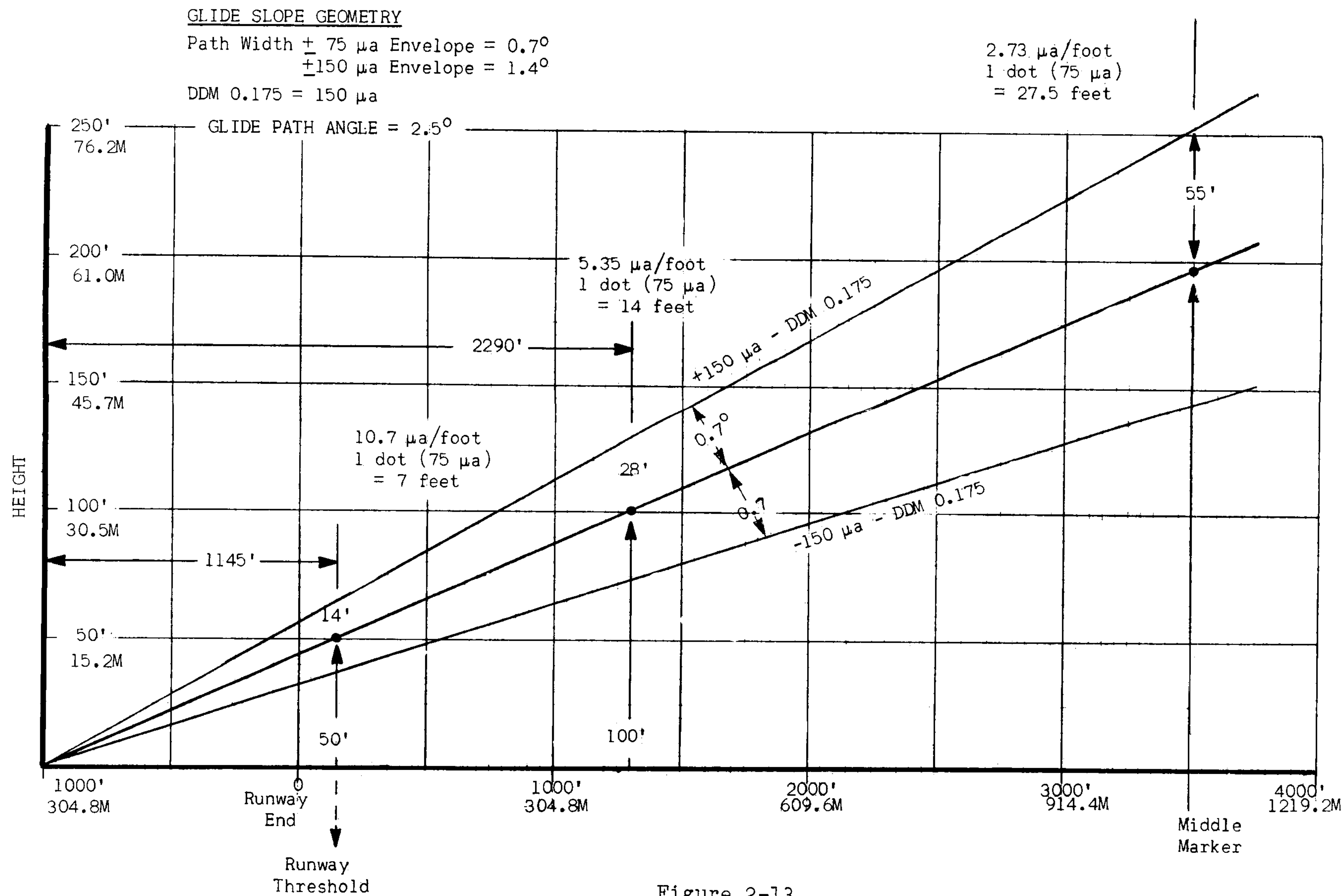


Figure 2-13
2.5 Degree Glide Slope Geometry

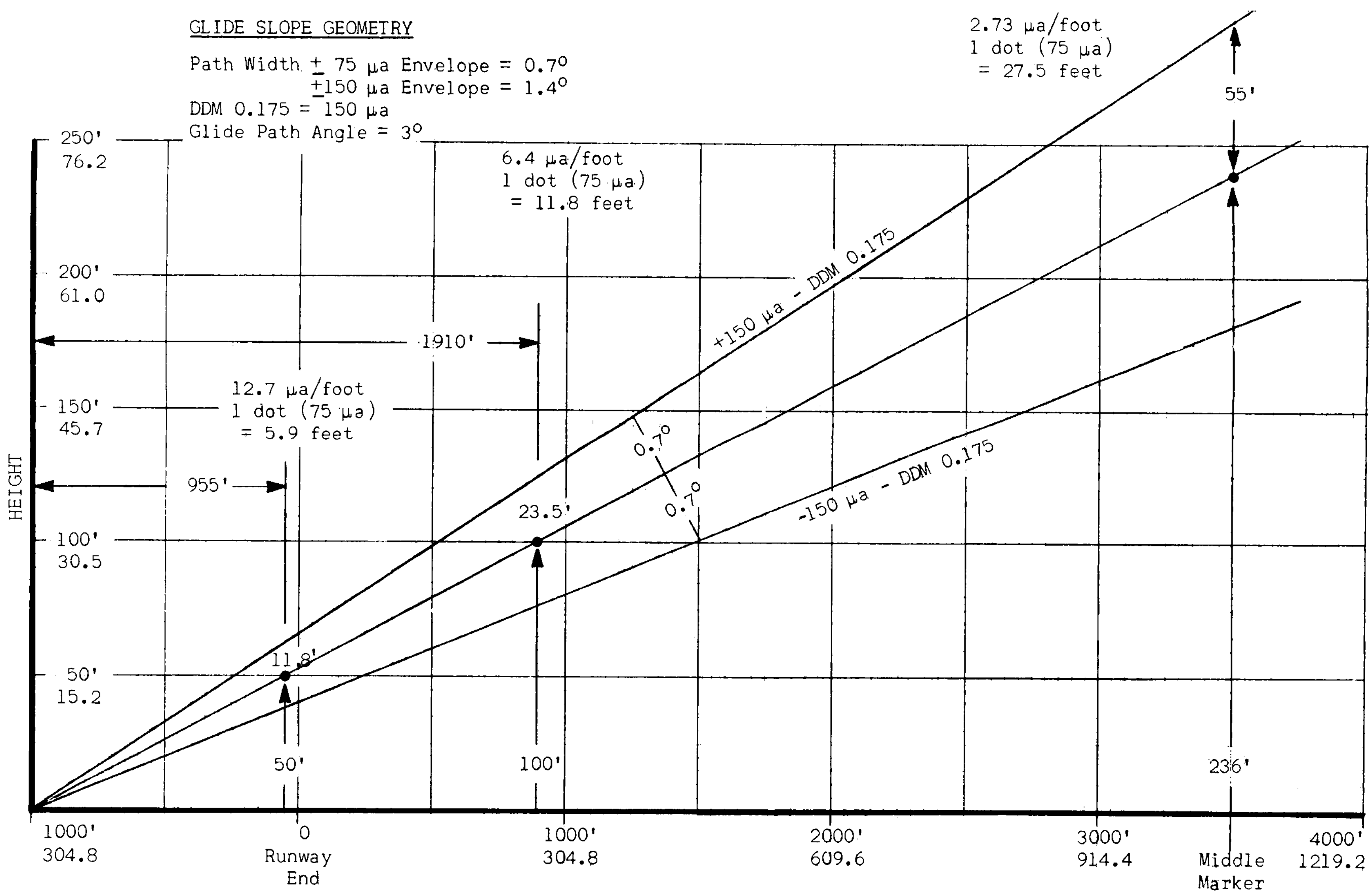


Figure 2-14
3 Degree Glide Slope Geometry

beam, a 3.66-meter (12 feet) error at an altitude of 15.24 meters (50 feet) would take an aircraft outside of the beam confines and a hardover signal would result. While the two glidepaths illustrated in these figures would both meet the most recent ICAO standards, note how different information would be provided by each of these beams. The standard location of the middle marker beacon is 1097.28 meters (3600 feet) from the runway end. If a landing system used this beacon signal to program control functions, the possible altitudes for an aircraft at the center of the glidepath would be 59.74 or 71.93 meters (196 or 236 feet). This corresponds to a 20-percent variation in altitude even for a fixed middle marker beacon location.

Of greater significance, however, is the variation in descent speeds that would occur on these two glidepaths. Table 2-1 is a summary of typical aircraft approach speeds and the corresponding descent rates along 2.5, 2.75, and 3.0 degree glidepaths. A 20-percent spread of descent speed for a given aircraft because of glide angle differences will have a significant effect on flareout precision and repeatability. Fortunately, it appears that this is not a very severe problem because a fairly broad range of touchdown parameters should be acceptable. However, if certain more stringent specifications on touchdown dispersions now under consideration by the FAA were applied, this could become a serious system problem. A more detailed discussion of these touchdown dispersion specifications will be given later in the section on flareout systems.

2.2 Vertical Flight Path Stabilization

The capture of the glidepath and the exercise of a tight closed loop control to minimize the deviations from that path is an accepted accomplishment of the automatic flight control state of the art. As in the case of most flight control problems, the difficulties which occasionally do occur are not caused by negligence in applying the analytical disciplines of control theory, but rather by subtle limitations of practical hardware. The theoretical description of the problem can be demonstrated by simplified third- and fourth-order models of the dynamic process involved. System designs start by providing excellent performance in terms of the linearized model that will now be reviewed. The systems are subsequently refined to account for the omnipresent nonlinear phenomena and the practical difficulties associated with measurement and signal processing.

TABLE 2-1

AIRCRAFT APPROACH SPEEDS

Aircraft	Typical Approach Speeds Knots			Typical Rates of Descent						Time From 30.48 Meters (100 Feet) to 12.19 Meters (40 Feet) From 2.75° Glide Slope Typical - Seconds
				2.5° Glide Slope		2.75° Glide Slope		3.0° Glide Slope		
	Low	Typical	High	FPS	MPS	FPS	MPS	FPS	MPS	
B727	105	115	120	8.47	2.58	9.32	2.84	10.17	3.10	6.43
B707-120B	125	130	135	9.58	2.92	10.53	3.21	11.49	3.50	5.70
B720B	121	125	132	9.21	2.81	10.13	3.09	11.05	3.37	5.92
B737		120		8.84	2.69	9.72	2.96	10.61	3.23	6.17
Electra	114	120	123	8.84	2.69	9.72	2.96	10.61	3.23	6.17
DC-9-10		123		9.06	2.76	9.97	3.04	10.87	3.31	6.01
DC-9-30		115		8.47	2.58	9.32	2.84	10.17	3.10	6.43
BAC-111		120		8.84	2.69	9.72	2.96	10.61	3.23	6.17
Convair 990		145		10.68	3.26	11.75	3.58	12.82	3.91	5.11
SST		135		9.95	3.03	10.94	3.33	11.93	3.64	5.48
DC-8-50	120	130	140	9.58	2.92	10.53	3.21	11.49	3.50	5.70
DC-8-62	125	135	150	9.95	3.03	10.94	3.33	11.93	3.64	5.48
DC-3	90	100	110	7.37	2.25	8.10	2.47	8.84	2.69	7.40

Figure 2-15 is a block diagram that defines the flight path control loop and the various parameters affecting performance. In this representation, an attitude control inner loop autopilot is assumed. The closed loop dynamics of this inner loop are identified as $H_2(s)$; and for the purposes of this qualitative analysis, we can assume that $H_2(s)$ is a first-order lag. The assumption of an attitude inner loop is consistent with established aircraft flight control practices. That is, the autopilot contains a basic pitch attitude stabilization loop and the steering commands are directed into this inner loop as pitch attitude commands. There are other types of inner and outer loop guidance and control relationships such as an attitude rate inner loop that receives normal acceleration commands. However, such systems involve considerable control law complexity to achieve a precise flight path control of the type desired for an automatic approach system. When this type of system is finally synthesized so that it provides the required flight path stabilization, its block diagram can usually be redrawn so that it is identical to figure 2-15. This is hardly surprising since we are attempting to control flight path by pitching the aircraft; hence, the basic control mode should be pitch attitude. While it is theoretically possible to control flight path by varying engine thrust and simultaneously maintain airspeed by pitching, this technique does not yield the rapid response and precision capability needed for an automatic approach.

It should be noted that figure 2-15 describes the flight path control stability parameters for either an automatic approach or a manual approach using flight director displays. In the manual flight director case, the closed loop pitch dynamics, $H_2(s)$, are provided by the pilot. If the proper quickening compensations based on pitch attitude and pitch rate feedbacks are added to the flight director computations, a first-order response with a time constant, τ_θ, can be approximated. There will be higher-order dynamics in $H_2(s)$ for both the manual and automatic cases, but the basic stability problem can be described first by assuming the simplified dynamics and then extrapolating to the effects of the higher-order terms.

The aircraft dynamic and kinematic terms are included as $H_3(s)$ and $H_4(s)$. The flight path angle response to a pitch attitude change is

$$H_3(s) = \left[\frac{\gamma(s)}{\theta(s)}\right] \approx \frac{1}{\tau_\gamma s + 1} \qquad (2\text{-}14)$$

where

$$\tau_\gamma = \frac{mV}{C_{l_\alpha} qS} \qquad (2\text{-}15)$$

Equation 2-14 is an approximation in that the long-term effects of the speed transient on the lift equation are neglected. The approximation is justified,

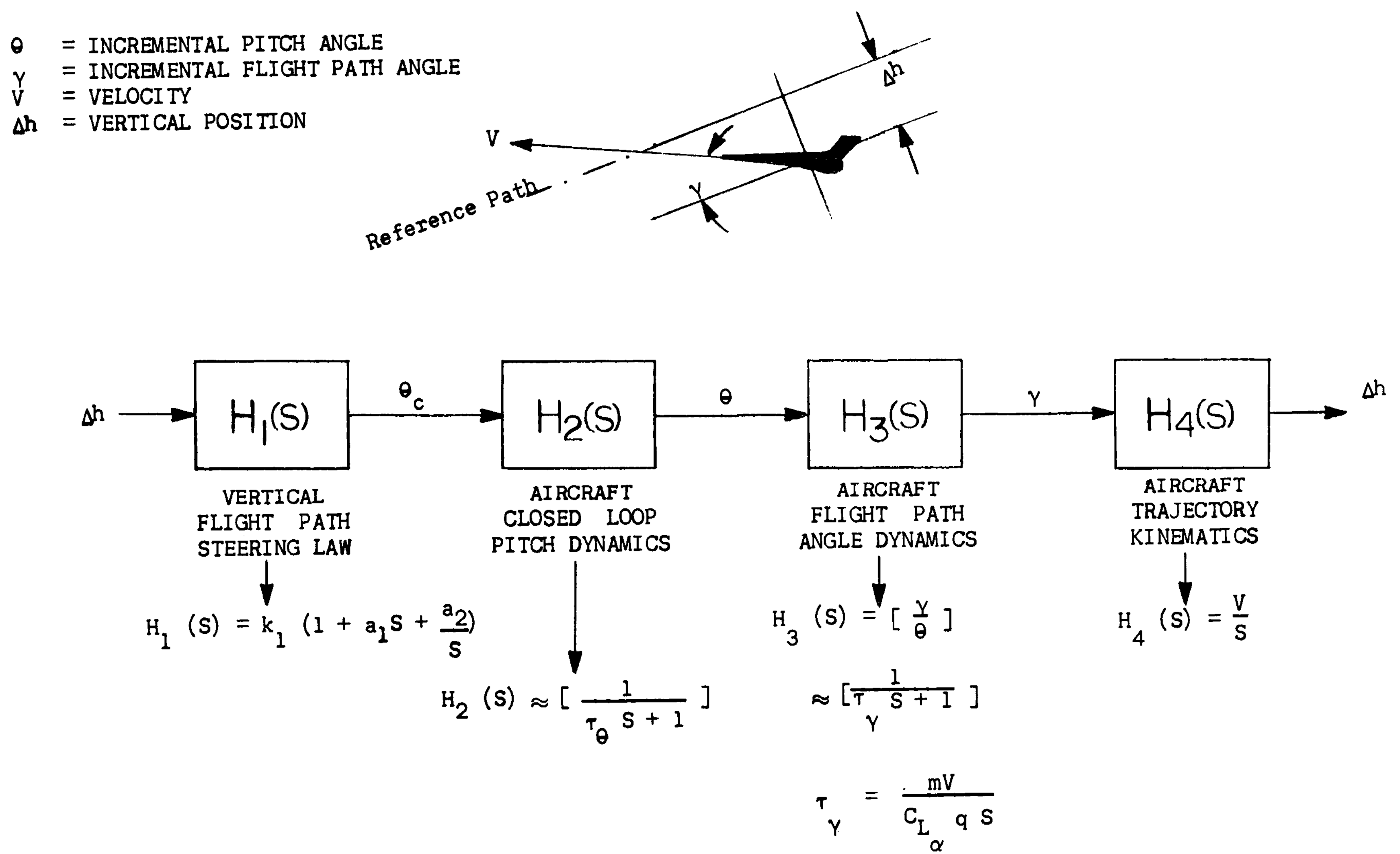
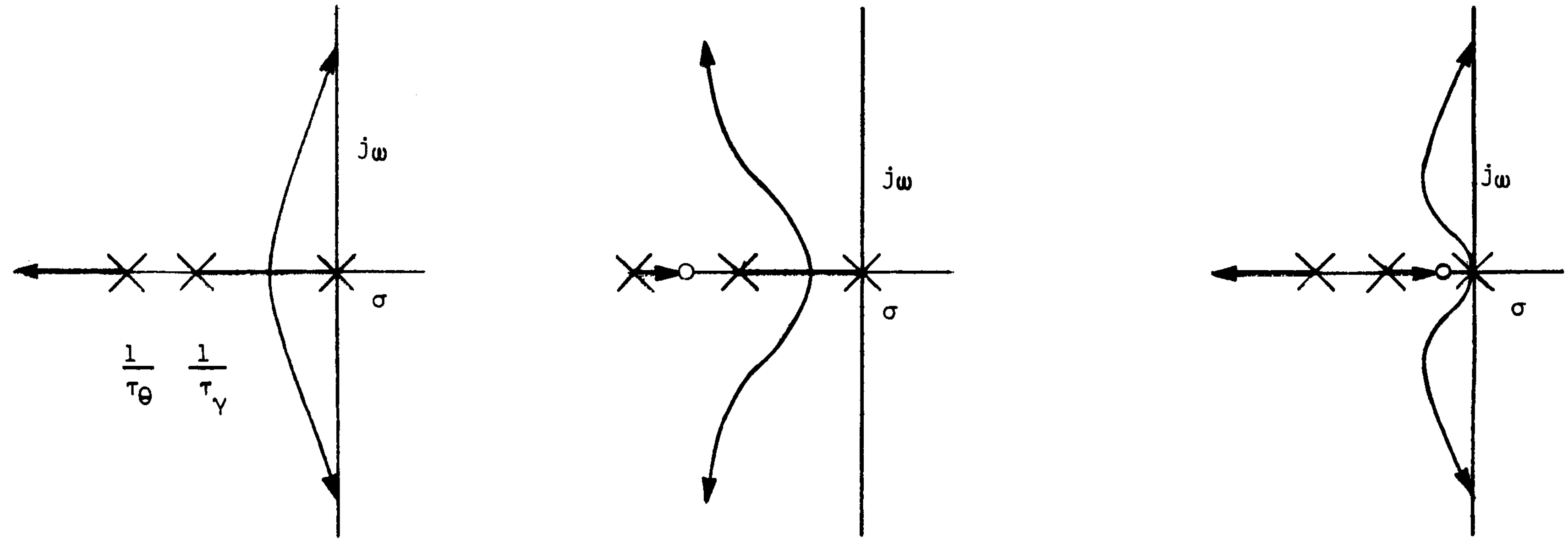


Figure 2-15
Vertical Flight Path Control Stability Parameters

however, if a throttle control loop (automatic or manual) is used to maintain airspeed.

The control law, $H_1(s)$, develops a pitch command as a function of the path displacement, rate, and integral. These three terms are shown in order to keep the problem general, but they are not always used or necessary. The rate term, for example, will usually be obtained by filtering and integrating accelerometer signals. It is often difficult to justify its use from the standpoint of performance improvement; nevertheless, it does find wide application. The integral term, on the other hand, does make an important operational contribution despite the fact that like most integrators, it has a destabilizing influence. Figure 2-16 shows qualitative root loci of how the various control law terms influence stability. In figure 2-16, the displacement term only is used. This root locus therefore corresponds to the variation in closed loop poles as the aircraft moves closer to the glide slope transmitter. The gain, k_1, in effect, corresponds to the changing sensitivity of the glide slope beam. As shown in this figure, a divergent instability will eventually occur even for the simplified approximation in the closed loop pitch dynamics. Figure 2-16(b) indicates that this divergence can be avoided if a rate term is added. The root locus in this case, however, is deceptive because a more complete representation of $H_2(s)$ would have shown that the addition of the rate term can excite the short period pitch mode. This pitch mode does not occur when we use a first-order lag to represent the pitch command response.

Figure 2-16(c) shows the influence of a small amount of the integral term in the control law. (Typical values of a_2 are 0.05 to 0.10 maximum.) The stability problem occurs when the loop gain is low. Thus, the initial response to a glidepath capture will be oscillatory at a low frequency if the beam is acquired at a great distance from the transmitter. Consequently, the higher the altitude of glide slope beam penetration the more troublesome the integral term will be. Nevertheless, for a fully automated system which must capture the glide slope from above or below and then accurately hold the reference descent path, the integral function is necessary. It is noted that in flight director-manual approach systems, the integrator in the form shown here is not allowable, but an alternate method of obtaining integral control is used. Integration is effectively accomplished by washing out the pitch feedback [within $H_2(s)$] at a time constant of about 20 seconds. This is dynamically similar to the integral term (a_2/s) in its destabilizing effect. The integral is needed to provide the steady-state pitch command corresponding to the change in aircraft attitude associated with the flight path change occurring after the glide slope is acquired. There are open loop techniques which can be used to minimize the dependence upon the integrator for the steady-state pitch command, but the integrator is the only closed loop means of eliminating beam standoff error caused by equilibrium attitude change requirements. It is sometimes suggested that the



(a) $H_1(S) = k_1$

DISPLACEMENT CONTROL ONLY

(b) $H_1(S) = k_1(1 + a_1 S)$

DISPLACEMENT PLUS RATE.

(c) $H_1(S) = k_1\left(1 + \frac{a_2}{S}\right)$

DISPLACEMENT PLUS INTEGRAL

Figure 2-16
Vertical Flight Path Control Stability Analysis

integrator is also useful in minimizing errors resulting from windshear. Unfortunately, the stability problem caused by the integrator restricts its utility in the case of windshear because sufficient integral gain to be effective is not attainable. This can be demonstrated by the following simplified analysis:

$$\text{Windshear} = \left(\frac{dV}{dh}\right)$$

$$\dot{V} = \left(\frac{dV}{dt}\right) = \left(\frac{dV}{dh}\right)\left(\frac{dh}{dt}\right) = \left(\frac{dV}{dh}\right) V_\gamma \qquad (2\text{-}16)$$

For nonaccelerating flight in the vertical plane

$$L \cos\gamma = W \approx L = C_{L_o} qS \qquad (2\text{-}17)$$

$$C_L = C_{L_o} + C_{L_\alpha} \Delta\alpha \qquad (2\text{-}18)$$

where C_{L_o} is the initial equilibrium lift coefficient

$$\Delta L = C_{L_o} (\Delta q)S + C_{L_\alpha} \Delta\alpha \; qS \qquad (2\text{-}19)$$

For $\Delta L = 0$ (to maintain nonaccelerating flight)

$$C_{L_o} \Delta \; qS = -C_{L_\alpha} \Delta\alpha \; qS \qquad (2\text{-}20)$$

or

$$C_{L_o} \frac{dq}{dt} = -C_{L_\alpha} q \frac{d\alpha}{dt} \qquad (2\text{-}21)$$

$$\left(\frac{d\alpha}{dt}\right) = -\left(\frac{C_{L_o}}{C_{L_\alpha} q}\right)\left(\frac{dq}{dt}\right) = -\left(\frac{W}{q^2 SC_{L_\alpha}}\right)\frac{dq}{dt} \qquad (2\text{-}22)$$

$$q = \frac{1}{2}\rho V^2 \qquad (2\text{-}23)$$

$$\frac{dq}{dt} = -\rho V \left(\frac{dv}{dt}\right) = -\rho V^2 \left(\frac{dv}{dh}\right) \dot{\gamma} \qquad (2\text{-}24$$

$$\frac{d\alpha}{dt} = \left(\frac{W}{q^2 SC_{L_\alpha}}\right)\left[\rho V^2 \left(\frac{dv}{dh}\right) \gamma\right] \qquad (2\text{-}25)$$

or

$$\left(\frac{d\alpha}{dt}\right) = - \left(\frac{W_\gamma}{2qSC_{L_\alpha}}\right) \left(\frac{dV}{dh}\right) \tag{2-26}$$

For constant γ,

$$\left(\frac{d\alpha}{dt}\right) = \left(\frac{d\theta}{dt}\right) \tag{2-27}$$

Thus, the pitch rate, $\dot{\theta}$, required for a windshear (dV/dh) is given by equation (2-26). To demonstrate its significance on flight path accuracy consider the effect of a windshear on the Boeing 747 airplane during a glide slope approach:

$$(W/S) = 102; \; C_{L_\alpha} = 4.81 \text{ per radian}$$

Assume Airspeed = 160 knots ($q = 88.1$ lbs/ft^2)

Then from (2-26)

$$\dot{\theta} = 0.291 \text{ degree/second}$$

For a typical glidepath control gain, a 0.05 degree pitch command is generated per 0.30 meter (foot) of glidepath deviation [6.10 meters (20 feet) per degree]. To maintain nonaccelerating flight in the vertical plane during the specified windshear condition, the 0.291 degree per second pitch rate must be commanded by a 1.77 meter/second (5.82 feet/second) rate of departure from the glidepath. This rate of departure will continue as long as the windshear persists and the throttles have not compensated for the speed change. Now the integral term in the control law could, in principle, halt the departure from the reference path by providing the necessary pitch rate command. Such a pitch rate command would be provided by a fixed beam standoff. Because of the very low allowable integral gains (typically 0.05 times the displacement gain), a standoff of 1.77 x 20 = 35.40 meters (5.82 x 20 = 116.4 feet) would be required. This obviously could never be the determining factor in a windshear transient since an unreasonable speed change would have to occur before a beam departure of over 30.48 meters (100 feet) could develop. Consequently, the only effective methods of coping with fore-aft windshear are high displacement gains and tight throttle controls.

There are three main factors that influence an automatic approach system's ability to steer the aircraft through the narrow vertical position window which aligns the aircraft for the flareout. They are

- Speed Disturbances - Last minute windshear and changes of aircraft configuration (flaps and gear) can produce flight path transients which penetrate the acceptance window boundary. The effects of these disturbances are countered by tight throttle controls and high gain flight path control loops.

- Flight Path Stability - The convergent nature of the glide slope beam moves the system gain toward infinity as the runway is approached. As the gain increases, the damping of the flight path mode [figures 2-16(a) or (c)] decreases and moves toward instability. To use the glide slope signal adequately at low altitudes as well as the higher altitudes, the gain, k_1, of figure 2-15 must be programmed downward. Many techniques have been used or have been proposed for accomplishing this reduction. They include completely open loop, time gain reductions initiated at glide slope intercept. This technique will work well when the aircraft always intercepts the glide slope at a fixed altitude and flies at one airspeed. A semi-open loop timed gain reduction program initiated by the middle marker offers improvement over the first system in that the system gain is effectively updated at the middle marker [nominal altitude about 67.06 meters (220 feet)]. A closed loop corrective capability is added to this latter arrangement by controlling the gain reduction program as a function of radio altitude after the middle marker signal is received. None of these methods is ideal for all operating conditions, but satisfactory results are readily obtained with the latter two or other approaches involving even more complexity. Table 2-2 is a summary of various glide slope control gain reduction programming techniques and their relative advantages and disadvantages.

- Glide Slope Beam Bends - Glide slope noise in the form of beam bends can produce flight path transients which can force the aircraft to the boundary of the flareout acceptance windows. This is a very difficult problem to document adequately. While improved ILS facility standards for lower minima operations would appear to alleviate this problem, questions persist regarding the existence of beam anomalies. Many flight recordings of ILS approaches testify to the apparent existence of beam bends, but it is usually difficult to differentiate between the effects of beam anomalies, wind disturbances, and flight path response dynamics.

TABLE 2-2

ILS GAIN REDUCTION PROGRAMMING TECHNIQUES

Method	Advantages	Disadvantages
1. Open Loop Timed Gain Reduction Program - Initiated at glide slope intercept	• Simplicity	• Performance is variable with altitude of intercept and aircraft velocity. Unstable low altitude performance can occur for short approaches.
2. Open Loop Timed Gain Reduction Program - Initiated at middle marker	• Simplicity	• Middle marker location for various ILS facilities can produce timing errors. Unstable performance possible for shallow beam with middle marker close. • Performance compromised for far-out glide slope intercepts.
3. Gain Reduction Program Initiated at Middle Marker - Program is function of radio altitude	• Good performance capability for all ILS facilities and aircraft speeds	• Performance variations possible for irregular ground profiles. • Performance compromise for far-out glide slope intercepts.
4. Gain Reduction Program initiated by radio altimeter and controlled as function of radio altitude	• Good performance capability is independent of ILS facility or intercept procedure and aircraft speed.	• Possible problems for irregular ground profiles.
5. Gain Reduction Program as function of distance from ILS transmitter (DME)	• Ideal performance possible.	• DME equipment not available.

2.3 Automatic Throttle Controls

An essential part of the vertical path stabilization problem involves control of the aircraft's speed. In the simplified description of the flight path control process as given in figure 2-15, the effects of aircraft speed changes were neglected because the most important stability problems were at frequencies where the speed transients made only minor contributions to the process dynamics. However, from the standpoint of long-term control accuracy, the speed effects are dominant. In figure 2-15, the simplification was in $H_3(s)$, the dynamic relationship between flight path angle and pitch angle. When the long-term effects are considered, this transfer function actually has the following form:

$$\left[\frac{\gamma}{\theta}\right] = \left(\frac{k\left(\frac{s}{b_1} \pm 1\right)}{\left(\tau_\gamma s + 1\right)\left(\frac{s}{b_2} + 1\right)}\right) \qquad (2\text{-}28)$$

The polarity of the zero (b_1) is related to the back side of the power curve operation. The usual view of the back side of the power curve operation is in terms of speed stability. That is, a decrease in speed will require an increase in thrust. This phenomenon is reflected in equation (2-28) by moving the zero b_1 into the right-half plane. Thus, in terms of flight path control, a nose-up pitch command will rotate the flight path angle upward initially, but eventually the flight path angle change will reverse polarity. There will be an ultimate error which will be larger than the initial value. The more rapid the speed divergence, the farther the zero b_1 moves into the right-half plane. This will correspond to a more rapid reversal of the flight path angle polarity resulting from a pitch change.

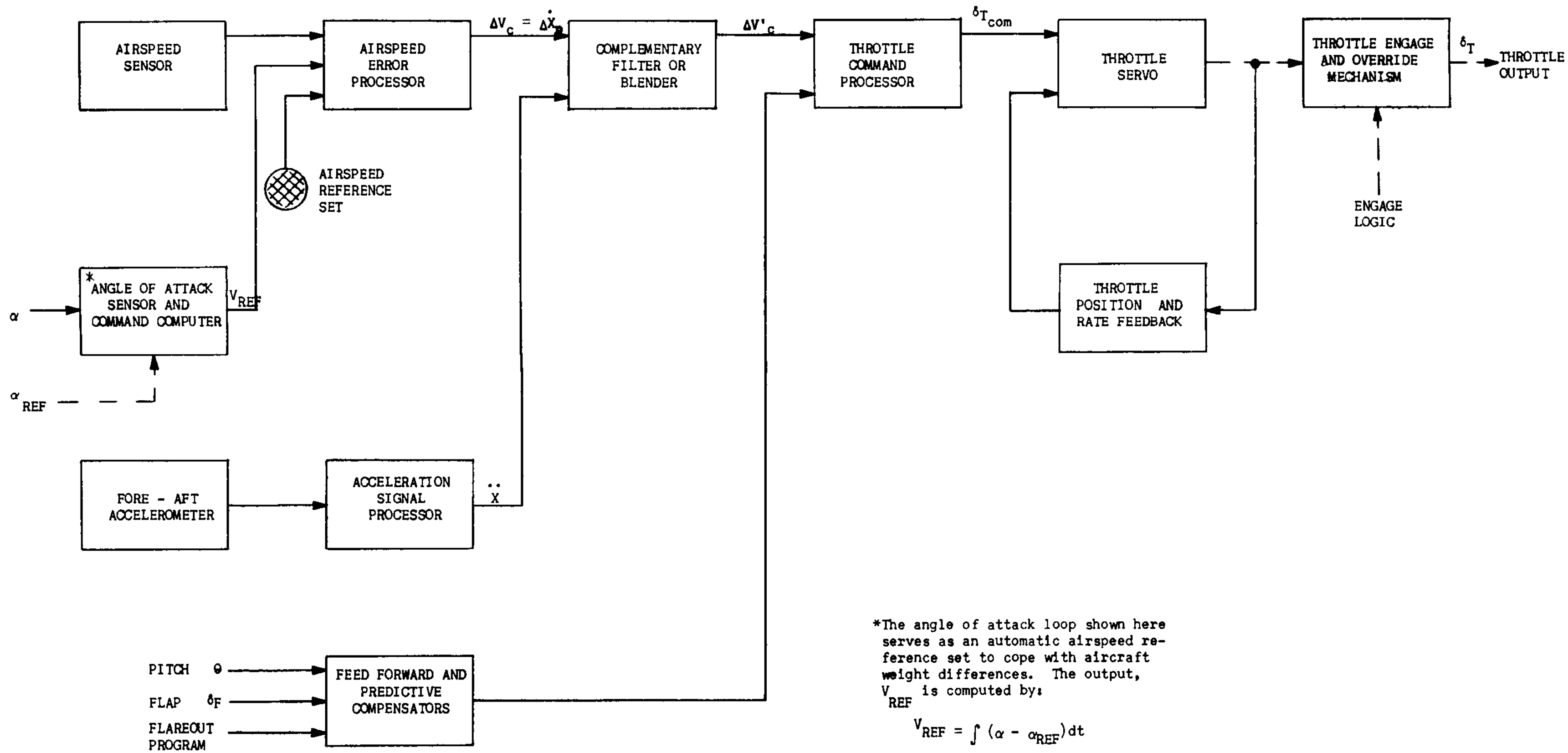
Most commercial jet aircraft operate in a speed-stable region, but where the value of k in equation (2-28) is near zero. That is, the steady-state flight path angle is not influenced too significantly by a pitch change. The double delta configuration of the Lockheed L2000 SST design operated on the back side of the power curve during landing approaches. This was considered to be a liability in some quarters; but from the point of view of flight path control precision, it makes little difference whether the aircraft is slightly divergent or convergent in regard to speed stability. The important fact is that the pitch loop illustrated in figure 2-15 cannot maintain an accurate flight path if significant speed transients are allowed to occur. These speed transients will occur for both the stable and the unstable power curve operations. In early automatic approach systems, when the pressures for lower weather minimum landings were not too great and system performance was not too carefully scrutinized, the adjustment of power was a function exclusively reserved for the pilot. If he coordinated his throttle adjustments with the control activity produced by the autopilot and its approach coupler, performance was satisfactory. When this was not done, the glidepath control accuracy deteriorated rapidly. In recent years, this intimate relationship between throttles and the autopilot's pitching maneuvers has been recognized to the point that almost all automatic landing systems include automatic throttle controls.

Controlling the aircraft's airspeed by means of a throttle adjustment is a very simple task from the theoretical servomechanism point of view. We wish to control the quantity, V, and our controller produces an output proportional to $\dot{V}$; hence, we have the classical first-order servo loop. Practical throttle control systems are more difficult. Stability is not the problem. The important requirement is to smooth or minimize the control activity and yet maintain accuracy. This is somewhat of a contradiction for a closed loop process since the best way to solve the regulation problem is to make the controller a high gain, wide bandwidth, responsive device. In the airspeed control problem, we wish to slow down the movements of the throttle even for large errors. Moreover, we want rapid elimination of airspeed errors, but not necessarily all airspeed errors. If we kept the airspeed at exactly the reference value, then compensation for wind gusts will be achieved by accelerating the aircraft continuously in response to these gusts. It is necessary that the aircraft maintain its velocity with respect to the air mass, but not to the higher frequency velocity perturbations of that air mass.

With the many constraints imposed on an automatic throttle control system, the desired results must be obtained with a fairly sophisticated combination of closed loop feedbacks and a posteriori predictive or open loop compensations. Figure 2-17 is a general block diagram that illustrates some of the functions provided by typical throttle control systems in use today. The upper group of blocks represents the basic airspeed error control path. In its simplest implementation, the pilot sets an airspeed reference corresponding to the aircraft weight. The calibrated airspeed error, ΔV_c, will command a proportional throttle increment. The setting of the airspeed reference can be made automatic by slaving the airspeed reference to an angle-of-attack loop. The optimum angle of attack for the various approach phases is programmed within an angle-of-attack command computer. The airspeed reference is driven to the value corresponding to the required angle of attack for any aircraft weight. The control law for setting the airspeed reference in this manner is of the form

$$V_{REF} = \int (\alpha - \alpha_{REF})\, dt \qquad (2\text{-}29)$$

Figure 2-17 shows the airspeed error being blended with a signal derived from a fore-aft accelerometer. As shown in this figure, a complementary filter can be used to synthesize a wide bandwidth airspeed error signal. This is accomplished while simultaneously providing considerable filtering on the



*The angle of attack loop shown here serves as an automatic airspeed reference set to cope with aircraft weight differences. The output, V_{REF} is computed by:

$$V_{REF} = \int (\alpha - \alpha_{REF})\,dt$$

Figure 2-17
Automatic Throttle Control System
Functional Block Diagram

component of that signal which is susceptible to gust and turbulence noise. If the complementary filter is a lag network of time constant τ and the inputs are forward acceleration, $\ddot{X}$, and airspeed change derived from a pitot source, $\Delta\dot{x}_\rho$, then the output, $\Delta V_c'$, is

$$\Delta V_c' = \frac{a_1\dot{x}_\rho + a_2\ddot{x}}{\tau s + 1} \qquad (2\text{-}30)$$

or, for the no wind case,

$$\Delta V_c' = a_1\dot{x}\,\frac{\left(\frac{a_2}{a_1}s + 1\right)}{\tau s + 1} \qquad (2\text{-}31)$$

and, if

$$a_2/a_1 = \tau,\ \text{then}\ \Delta V_c = a_1\dot{x}$$

regardless of how large τ is made. However, in turbulent conditions, the signal derived from the pitot source is heavily filtered so that the throttles are not required to respond to gusts. The advantages of this technique have been described in reference 4, and it is being used in the automatic throttle controls for the Boeing 727, 737, and Douglas DC-8 series and DC-9.

Figure 2-17 shows that all throttle command signals are transmitted through a Throttle Command Processor. This function provides the throttle rate and position limits that are so essential to acceptable operation. The rate limit, for example, acts as a heavy low pass filter for large errors. Since the system is effectively saturated and hence operating at low gain for large errors, it is important that such errors be avoided. A method of holding system accuracy despite the low effective gains is to apply predictive correction commands in anticipation of the speed transient which will ensue. These commands may be interpreted as feedforward compensators in some cases. (For example, if the α_{REF} is changed with flap deflection, then a feedforward throttle command proportional to flap extension, δ_F, may be used to minimize the speed transient.) A commonly used predictive compensation signal is derived from pitch angle. In this case, a nose-up pitch attitude change will command a throttle advance to compensate for the deceleration which would normally accompany the pitch change.

As seen in figure 2-17, therefore, a throttle control system for automatic landing operations can be fairly sophisticated despite the rather simple nature of the control task. The control parameters used in such systems are intimately related to the specific aircraft characteristics. In the area of throttle controls, there are not considered to be any technological problems for which solutions are not now known. These solutions usually involve additional system complexity. Economic considerations and uncertainties regarding the degree of automaticity required have kept most operational systems relatively simple.

2.4 Flareout

2.4.1 Flareout Control Laws

Flareout control laws must be compatible with the flight path aiming phase which occurs prior to the flare. The glide slope control, described previously, is accomplished with a pitch command steering loop. For compatibility, the flareout phase should be an extension of the pitch command steering. This pitch steering loop must bring the aircraft to the runway and meet four main terminal condition (touchdown) constraints. They are as follows:

- Terminal Vertical Speed, $\dot{h}_{TD} \lessgtr \dot{h}_{MAX}$
- Touchdown Position, $x_{TD} \leq x_{MAX}$
- Forward Speed, $V_{MIN} < V_{TD} > V_{MAX}$
- Pitch Attitude, $\theta_{TD} \leq \theta_{MAX}$

Before describing flareout control laws, a discussion of the general problem of flight path controllability would be appropriate. Any closed loop dynamic process, including those which attempt to control the aircraft's vertical position, can be characterized by such frequency domain parameters as closed loop frequency and damping ratio. The s plane representation of the simplified vertical flight path dynamics was shown in figure 2-16. Even nonlinear closed loop processes have their piecewise linear equivalents or instantaneous representation on the s plane. The importance of this s plane or frequency domain representation is that it can be related to response times. Thus, a mode having critically damped characteristics with a time constant τ can be thought of as requiring 4τ seconds to settle from a disturbance. Likewise, a 0.5 damped, second-order characteristic mode can be considered to have a one-cycle settling time. In order to interpret these basic dynamic properties of a closed loop system in terms of the flight path control problem, a transformation from time to distance should be made. With such a transformation, a closed loop frequency can be interpreted as a wavelength. In terms of a frequency, ω (or its period T), the wavelength, λ, will be

$$\lambda = VT = \frac{2\pi V}{\omega} \qquad (2\text{-}32)$$

The closed loop frequency of a flight path control system is dependent upon the factors identified in figure 2-15. Larger aircraft will tend to have larger lags in $H_2(s)$ and $H_3(s)$ of that figure, and hence lower closed loop natural frequencies. For jet transports, a closed loop path control frequency, ω, of 1.0 radian per second is representative of an extremely high gain system. A period of 10 seconds (ω = 0.628 radian/second) is more representative of a realizable high gain system (providing wide bandwidth position and rate data are available). For a final approach speed of 120 knots (203 feet/second; 61.87 meters/second), the values of λ for a 10-second period system will be

$$\lambda = 618.74 \text{ meters (2030 feet)}$$

A reasonably damped system will require one wavelength to settle to its final value; but one wavelength is about the entire downrange distance of the flareout.

Approach and landing guidance concepts that use only a curved path extension of the glide slope for flareout have been postulated. Such a system would always be on closed loop control to the glide slope reference except that the reference would curve near the ground. If there is no other recognition that the reference path is changing other than the sensing of glidepath error signals, then such a scheme is doomed to failure. The closed loop process requires about one wavelength to eliminate errors and the total path is about one wavelength. Path deviations sensed near the ground could never be fully corrected. It is the recognition of this basic weakness of the closed loop error sensing systems which motivated study and development of terminal control techniques for flareout. Terminal control systems as well as the applicability of other explicit guidance concepts will be reviewed later. It is noted, however, that terminal controllers are also limited when viewed in terms of the wavelength principle.

The point of this discussion is that closed loop control of a flight path is by itself inadequate for a flareout system. The solution lies in applying a predictive command in the same manner as a pilot when he lands an aircraft manually. The pilot does not wait to see flight path errors develop before he pulls back the control column. At the proper time, he initiates a rotation command and then adjusts his control in a partial closed loop response to the resulting aircraft motion. Most automatic flareout systems now operating or under development use a similar approach. Figure 2-18 is a functional block diagram that is sufficiently general to be representative of most of these systems. This figure shows the flow of information leading to the generation of the pitch steering command that is applied to the autopilot. Note that the glide slope control signal, θ_{c_1}, is summed at the same point as the flare command signals. As the ground is approached and the glide slope signal

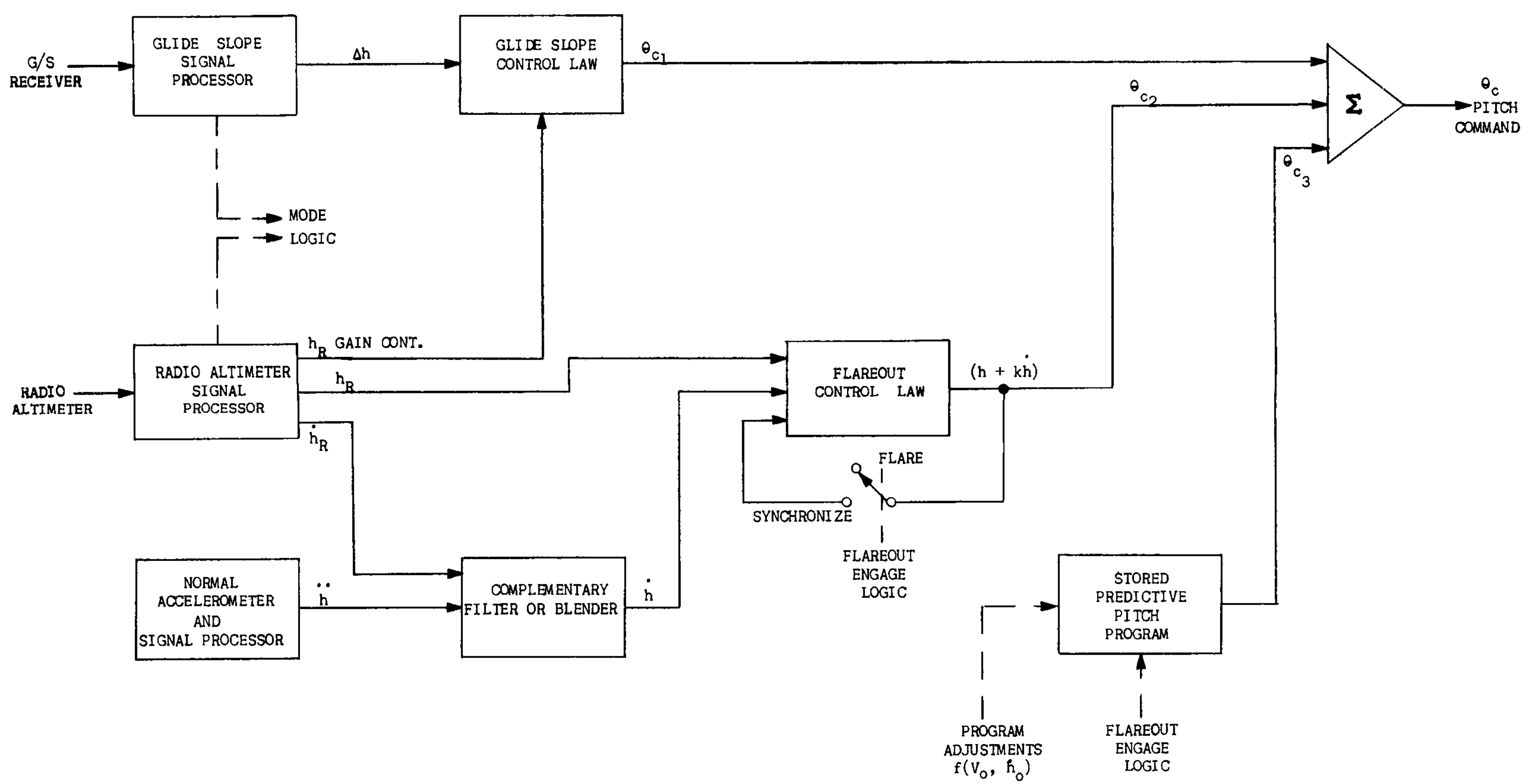


Figure 2-18
Flareout System, Functional Block Diagram

moves toward infinite sensitivity as well as doubtful accuracy, the gain is reduced toward zero by data obtained from the radio altimeter. By the time an altitude of 15.24 meters (50 feet) above the runway is reached, the glide slope loop is effectively driven to zero gain. The only information remaining in θ_{c_1} will be steady-state pitch command derived from the path control integrator. Prior to flareout initiation, the aircraft may be flying a straight attitude hold function. This can exist for the duration between the zero gain glide slope control and flareout initiation. Some systems will exercise closed loop control over the existing vertical speed at the time the glide slope gain goes to zero. If this interval is only 1 or 2 seconds, pitch attitude hold appears to be adequate.

The flareout control law, initiated at a critical altitude that is sensed by the radio altimeter is of the form

$$\theta_c = K_1(1 + \frac{a_1}{s})[h + a_2(\dot{h} - \dot{h}_F)] + \theta_p(t) \tag{2-33}$$

This control law defines a reference flight path having an exponential decay of altitude with respect to time. This path reference, h_{REF}, is

$$h_{REF} = (h_o - a_2\dot{h}_F)\ e^{-t/a_2} + a_2\dot{h}_F \tag{2-34}$$

The term $\dot{h}_F$ represents a vertical speed bias that calls for a finite sink rate at touchdown. It may also be viewed as an altitude bias that shifts the exponential altitude path to a steady-state value below the ground level. For a value of $a_2 = 5.0$ and a vertical rate bias of-0.46 meter/second (-1.5 feet/second), and if the flareout starts at 12.19 meters (40 feet), then equation (2-32) would yield the following typical path reference:

$$h_{REF} = 47.5\ e^{-0.2t} -7.5$$

The steering law which attempts to exercise closed loop control to this reference path is the loop gain, k_1, and a displacement plus integral (a_1) term. However, the term which in many systems is the most essential contribution to a successful flareout is the feedforward function or predictive pitch program, $\theta_\rho(t)$. In its ideal application, this open loop pitch command will cause the aircraft to follow a path such that

$$h + a_2(\dot{h} - \dot{h}_F) = 0 \tag{2-35}$$

If this were to occur, the control signal, θ_{c_2}, of figure 2-18 would be zero throughout the flareout. The operation of the system can then be viewed from the standpoint of the conditional feedback concepts (reference 5). That is, a corrective feedback control signal (θ_{c_2} of figure 2-18) will occur only when the actual response departs from the desired response. The $h + \dot{h}$ control loop may therefore be interpreted as a vernier control on the basic feedforward (or predictive) command, $\theta_p(t)$. The optimum predictive command is dependent upon vehicle aerodynamics, initial velocity, initial sink rate, and flare initiating altitude. When any of these items vary, the predictive command should be adjusted to minimize dependence upon the $h + \dot{h}$ closed loop vernier. Such an adjustment capability is implied by the inputs shown to the stored predictive pitch program block on figure 2-18.

The significant point of this discussion of flareout control laws is that performance is largely dependent upon open loop rather than closed loop controls. Accuracy depends upon maintaining an adequate bound on initial conditions. If these initial conditions are variable, good performance can still be achieved if we compensate with the open loop control commands.

2.4.2 Ground Effect

An aerodynamic phenomenon that often provides a similar effect as the predictive pitch program of equation (2-32) is the so-called ground effect. The ground effect will produce a net lift increase to cause an inherent flareout even when the pitch attitude is held constant. Foss, Magruder, and others in reference 6 point out that the Lockheed double delta SST design obtains a significant ground cushion effect which almost eliminates the need for a flareout maneuver. Figure 2-19, derived from reference 6, shows how a delta wing fighter aircraft which was modified to reflect the Lockheed SST platform provided a significant ground effect improvement over the current jet transport. The jet transport in this illustration is somewhat representative of the DC-8. Other transports now operating have a more significant ground effect cushion. For example, automatic landing studies for the Boeing 727 (reference 7) indicated that the ground effect contribution was sufficiently effective to eliminate the need for the predictive pitch command in the flareout control law. This is illustrated by figure 2-20 which shows simulation results of the flareout maneuver with and without ground effects included in the aerodynamic simulation. It is noted that these results were obtained with early estimates of the 727 aerodynamic characteristics. Subsequent flight measurements to derive the Boeing 727 ground effect coefficients have indicated that this preliminary data gave a somewhat optimistic result. The primary difference was in the effect on the drag coefficient. However, the basic characteristic indicated by figure 2-20 is still valid. Note that without ground effect, the closed loop $h + \dot{h}$ control law reduces the sink rate from 2.59 meters/second (8.5 feet/second) to only 1.83 meters/second (6.0 feet/second). With ground effect, the touchdown sink rate is 0.46 meter/second (1.5 feet/second); a very comfortable landing.

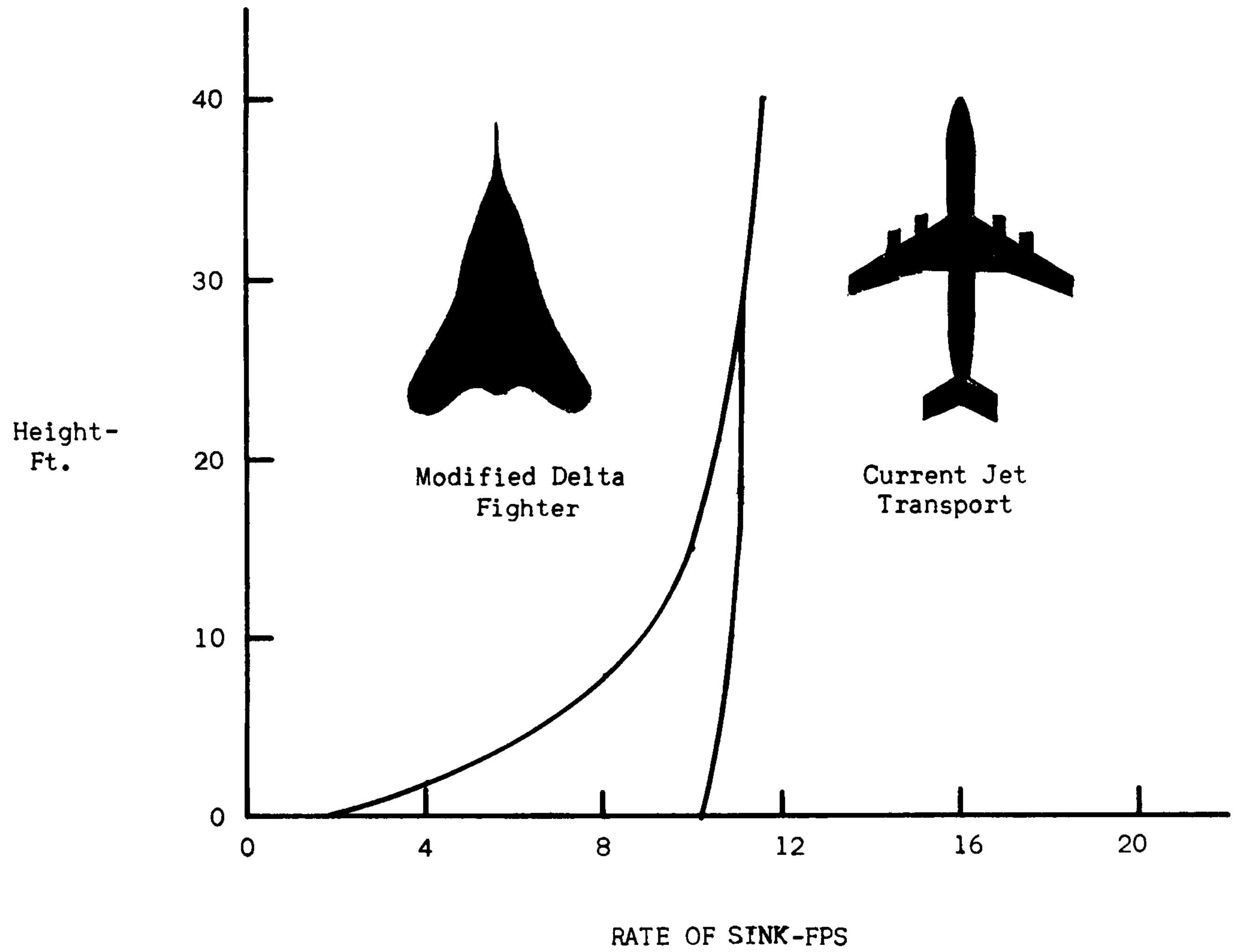


Figure 2-19
Landing Sink Rate in Ground Effect
Constant Pitch Attitude

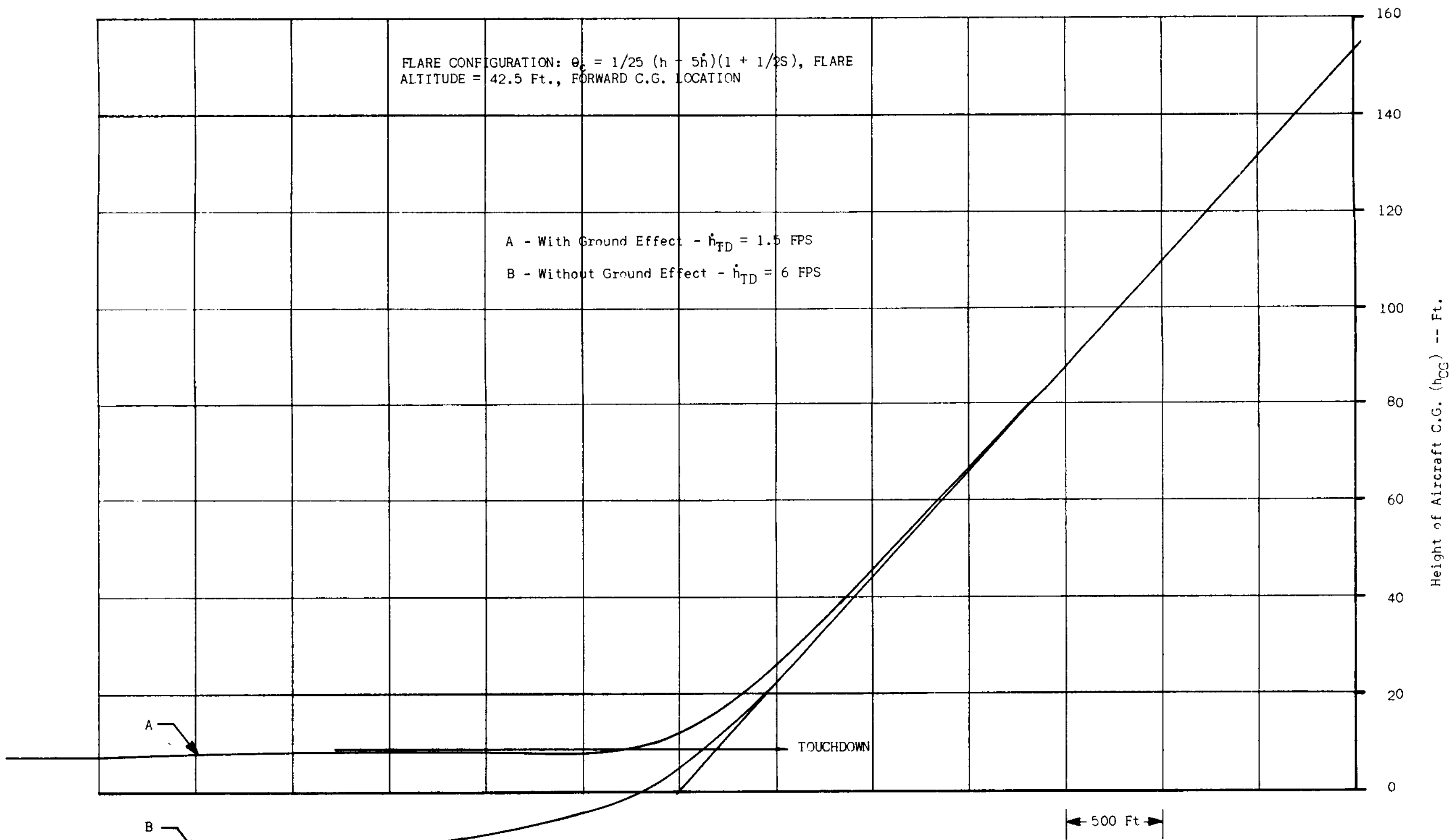


Figure 2-20
Effect of "Ground Effect" on Flare Maneuver

Figure 2-21 is a detailed breakdown of how the ground effect contributions to the pitching moment, drag, and lift coefficients influence the touchdown characteristic on the Boeing 727. If attitude is held constant, the flareout characteristic would be similar to curve A in figure 2-21. That is, the touchdown velocity will be about 1.07 meters/second (3.5 feet/second). This certainly makes the problem look quite easy. Indeed, even if there were no ground effect and the aircraft were crash landed from its glidepath trajectory, the landing would be hard but not catastrophic. The point emphasized here is that maintaining the initial conditions within satisfactory bounds is the key to a successful landing. The difficult part of the flight path control problem had been solved when the aircraft was brought to the precise flareout point at the correct velocity. Other problems relating to the effect of windshear and other variables on touchdown dispersion will be discussed later.

2.4.3 Measurement Problems

2.4.3.1 Terrain Profiles

The proper operation of the flareout system illustrated in figure 2-18 depends upon the accurate measurement of altitude and altitude rate. The radio altimeter is identified in figure 2-18 as the source of this information. It would have been desirable to use a data source that was not as dependent upon the terrain profile as is the radio altimeter. A precision radar system such as Flarescan or AILS (references 8 and 9) may provide this information without the terrain profile problem but the direction of landing system development for commercial transports has avoided any commitment to ground radar guidance concepts. The radio altimeter is becoming the standard position reference. However, altitude above the terrain can produce some erroneous information. Figure 2-22 is a representation of terrain profiles for several US airports (reference 10). It is seen that the Pittsburgh and Dallas glide slope approaches are particularly unsuited to radio altimeter guidance. It is important to note the possibility that vertical rate information derived from the radio altimeter can cause unsatisfactory results even when the profile is perfectly flat in the flareout region.

Consider the Pittsburgh profile shown in figure 2-22, for example. If flareout were to be initiated at about 12.19 meters (40 feet), we can easily demonstrate that a system such as the one illustrated in figure 2-18 can get us into serious troubles. The problem can be clarified by filling in some of the details of the flare control law generation as illustrated in figure 2-23. From this figure it is seen that the $h + \dot{h}$ error signal is synchronized prior to flare initiation. Ideally, $h = -a_2\dot{h}$ at the flare initiate altitude so that the error signal is zero at that time. This corresponds to a reference landing trajectory that is tangent to the aircraft flight path at flare initiation. If the aircraft velocity and hence the sink rate differs from the nominal value, the synchronizing loop will automatically provide the signal ϵ_o which restores

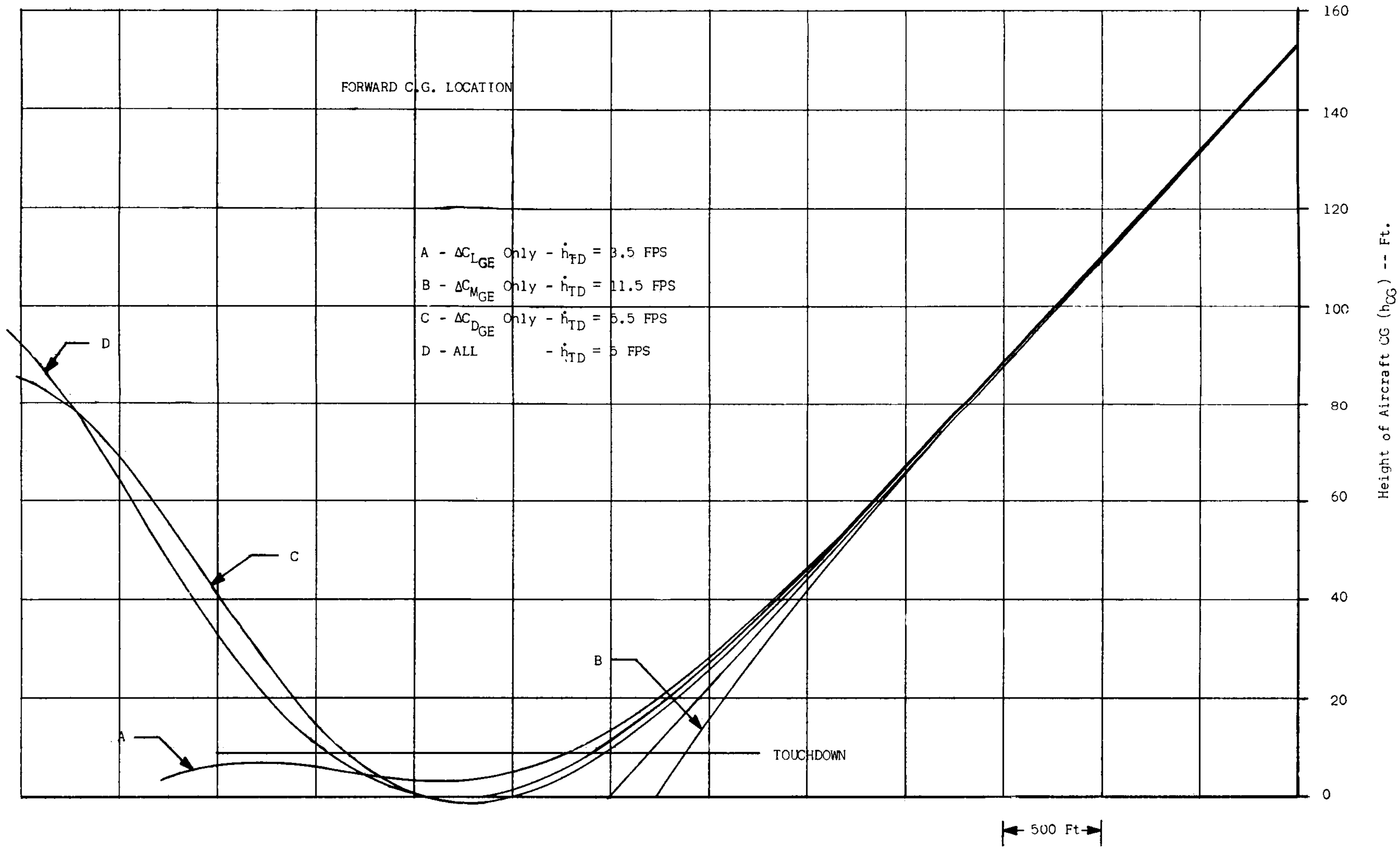


Figure 2-21
Effect of Lift, Drag, and Pitching Moment
Produced by Ground Effect on Free Airframe

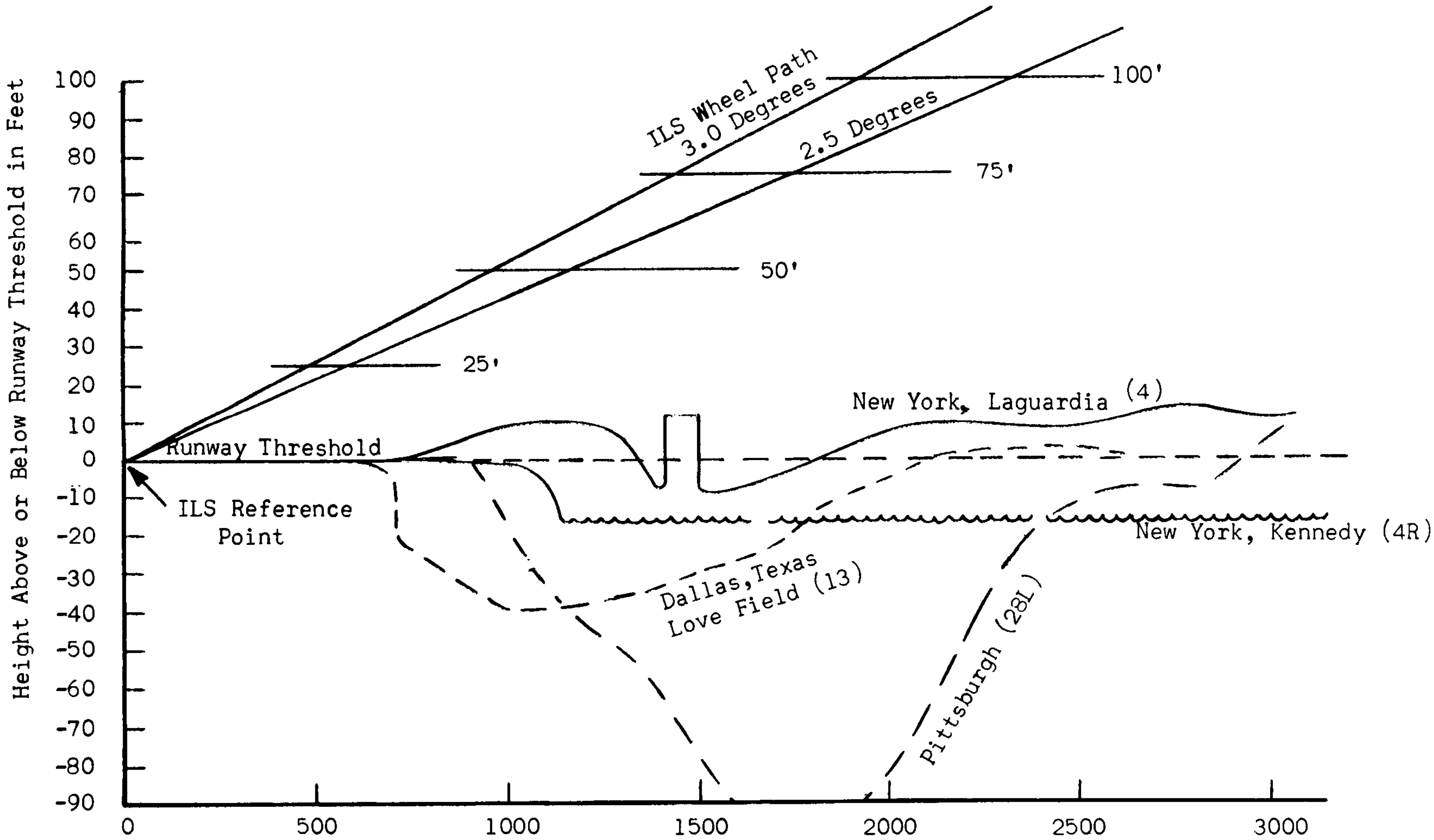
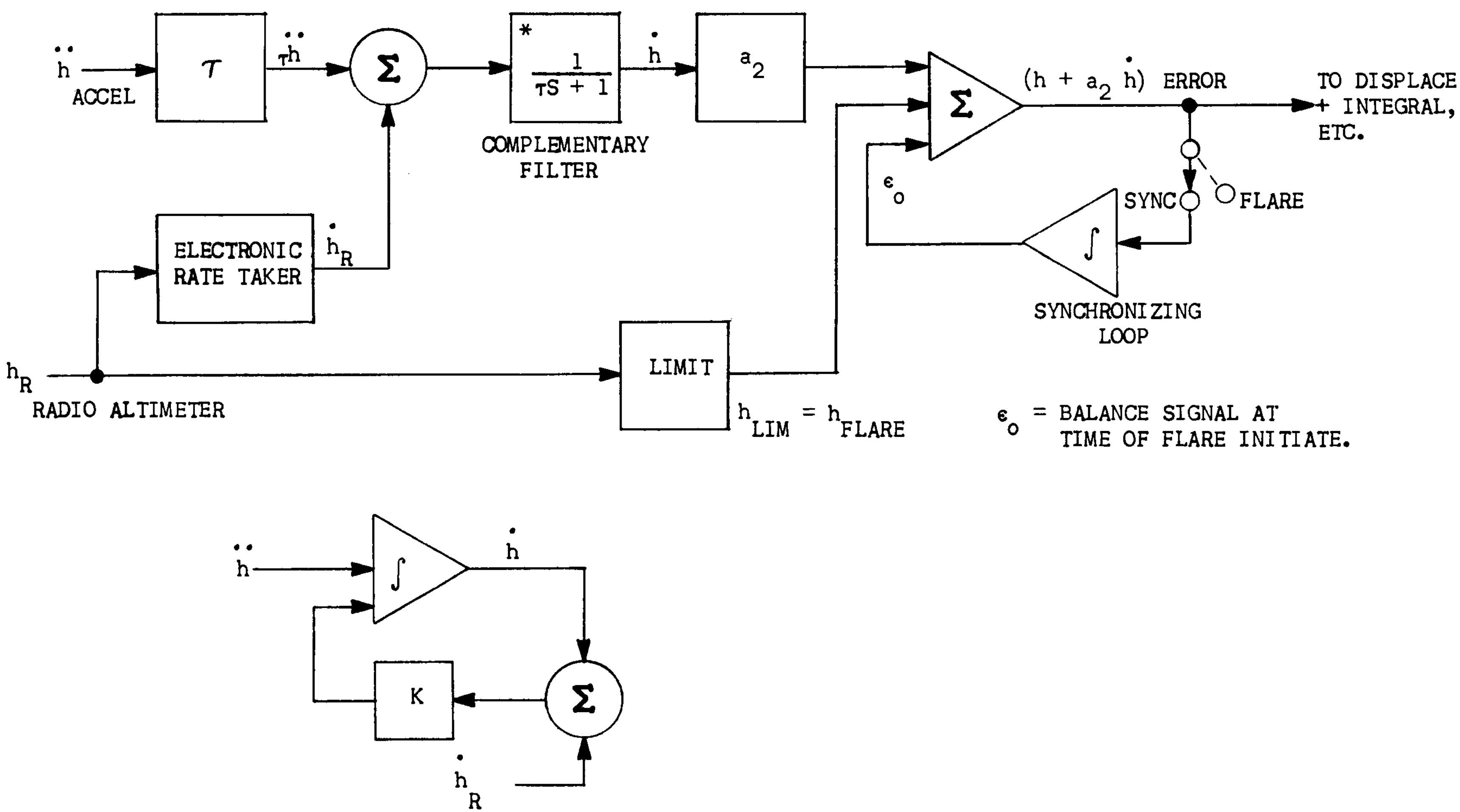


Figure 2-22
Terrain Profiles (Various Airplanes)

FOR IRREGULAR TERRAIN:

$\epsilon_o = a_2 \dot{h}_T$ where

$\dot{h}_T$ = erroneous terrain $\dot{h}$



*ALTERNATE EQUIVALENT OF COMPLEMENTARY FILTER.

Figure 2-23
Flare Control Law and the Effect of Erroneous Altitude Rate Caused by Irregular Terrain

the balance or trajectory tangency relationship when flare is initiated. Now let us consider the Pittsburgh terrain profile. For several seconds prior to the flare initiate point an erroneous radar altitude rate is experienced. If this erroneous rate is $\dot{h}_T$, the synchronizing signal, ϵ_o, will be $a_2\dot{h}_T$ for the ideal sink rate case. The signal ϵ_o will be of a nose-down command polarity since it is balancing what appears to be an excessive sink rate which called for a nose-up command. Soon after the flareout program begins, the terrain becomes level and the measured $\dot{h}$ signal begins to assume the correct value. The control law error signal will now begin to command a nose-down maneuver. Since all properly designed flareout computers include an asymmetric pitch command limiter that blocks nose-down commands, a catastrophic landing might be avoided, but nothing resembling the desired flareout path will be obtained.

Suppose we did not employ the synchronizing function and accepted a flareout system that could produce transient commands at the time of flare initiation. Would that eliminate the problem? Unfortunately, the problem would still exist except now the spurious command and polarity would be nose-up. The erroneously high sink rate existed immediately prior to the start of flareout. During the flare phase the aircraft is over level terrain; but in an inertially augmented $\dot{h}$ measurement, the long-term measurement of vertical speed is the heavily filtered radio altitude derived signal. The control signal is wide bandwidth $\dot{h}$ synthesized as shown on figure 2-23 in accordance with the following relationship.

$$\dot{h} = \frac{\tau \dot{h}_{accel}}{\tau s + 1} + \frac{\dot{h}_R}{\tau s + 1} \qquad (2\text{-}35)$$

where $\ddot{h}_{accel}$ is an inertially derived vertical acceleration signal and $\dot{h}_R$ is the derivative of the radio altitude signal. The value of τ is usually greater than 1.0 second. Thus the abrupt change in $\dot{h}_R$ that occurs when the Pittsburgh terrain flattens does not eliminate the erroneous signal caused by the previous hill. The error will be held in the lag network and will decay with a τ-second time constant. The effect of this error will be to produce a proportional nose-up command as long as it persists in the lag memory.

It is apparent therefore that pronounced terrain irregularities such as large hills and valleys will be remembered by inertially augmented vertical speed computers. Short or abrupt irregularities will be filtered. For example, the sharp rectangular protuberance in the New York La Guardia, Runway 4 terrain (figure 2-22) and the abrupt rise of the runway threshold from the water level at New York Kennedy, Runway 4R, will not be a source of difficulty because these irregularities are easily filtered. Because of these terrain problems, many systems have been designed to use barometric altitude rate rather than radio altitude data. Guidance problems associated with these systems will be discussed later. Another alternative is to attempt to extract wider bandwidth $\dot{h}$ data from the radio altimeter. This involves shortening the time constant of equation (2-35) to about 0.2 second. The ability to achieve good performance with this type of filtering depends upon radio altimeter quality and terrain roughness. Even if barometric altitude rate were used in the control law, the programming of the transition from glide slope to barometric control and the initiation of flareout must be based on an absolute position measurement. Because precision DME has not been available and ground-based precision radar landing system facilities have not been deployed for commercial transport operations, the radio altimeter has been the only device used for this absolute position measurement. Reference to figure 2-22 shows that none of the airports shown would give good data if we wished to initiate flareout at 15.24 meters (50 feet). This problem has been cause for much concern in recent years and the ICAO 7th COMM and the 7th AGA Divisions have recommended new landing facility criteria that are intended to alleviate the difficulty. Specifically, the 7th AGA Division made the following recommendations several years ago.

"In order to accommodate aircraft that will be using all weather landing facilities that incorporate a radio altimeter for final height and flareout guidance at the approach end of precision approach runways, it is desirable that slope change be avoided or kept to a minimum from a point where an aircraft, when on the nominal ILS glidepath, would be at an altitude of 22.86 meters (75 feet), until the point of touchdown. This is desirable because at an altitude of 22.86 meters (75 feet) the radio altimeter would begin to provide information to the automatic pilot for auto-flare."

Litchford (reference 10) has documented the vast extent of the glide slope-approach terrain-runway threshold variations and inadequacies. Several years have elapsed since this analysis pointed out the operational problems associated with the use of existing airport approach configurations and while some progress has been made toward improvement and standardization, most of the inadequacies described in reference 10 are still existent. As noted in reference 10, the cost of runways was estimated at $3200 per meter ($1000 per foot) and much of the required changes necessitated additional runway length.

2.4.3.2 Barometric Data for Glide Slope Extension and Flareout

While nobody suggests the use of absolute barometric altitude measurements as the source of altitude information for a flareout, barometric data can be used for a flareout control law. One often encounters statements which dismiss barometric measurements as inadequate for precision flight path control. While there are problems associated with such techniques, they cannot be excluded from consideration on the basis of arbitrary judgments regarding accuracy. Indeed, there are several successful implementations of flareout systems based on barometric altitude rate (references 11 and 12). In such systems the basic control law is rate of descent. At the flareout point, the rate of descent reference may be changed from the value maintained on the glide slope to a final value desired for touchdown. The predictive pitch program may still be used as described previously but now the vernier closed loop control is on vertical speed. Wide bandwidth vertical speed information can be obtained by using the same accelerometer blending technique used in computing vertical speed with the radio altimeter. That is, barometric rate would be applied to the complementary filter in the same manner as radio altimeter rate in figure 2-23. That the barometric pressure measurement is a potential good source of vertical speed is apparent by noting the quality of the pressure altitude flight recordings for the landings illustrated in figures 2-10 and 2-11.

Another way to use barometric rate is for the $\dot{h}$ part of the control law in the exponentail $h + \dot{h}$ flareout system discussed previously. The barometric $\dot{h}$ could avoid the terrain problem which tends to deteriorate the radio altitude rate data. However, there is an omnipresent problem associated with barometric signals in closed loop path control configurations. It is often referred to as the K_α problem. This notation refers to a coupling of angle-of-attack to the static pressure source. The units of K_α are equivalent meters (feet) of altitude error per degree of angle of attack. The polarity of K_α may be positive or negative depending upon location of the static source in each aircraft. Ideally the static source should be located at a point on the aircraft that is not susceptible to this error. Static source calibration checks are difficult and time consuming so that compromise locations are usually used. Figure 2-24 shows simulated flareouts (reference 7) that demonstrate the stability problem brought on by K_α effects. The aircraft is the Boeing 727 and the control law is similar to the one used in figure 2-21 except that a 3/4-second lag filter has been added to the pitch command,and barometric rate data is used. Because of the 3/4-second lag, the basic system performance without K_α errors is somewhat oscillatory. Note however that when the oscillations are viewed in terms of the wavelength interpretation, the fact that the flareout is essentially an open loop process is again apparent. As mentioned previously, a wavelength covers more than 609.60 meters (2000 feet) which is greater than the normal flareout downrange distance. The oscillatory tendency becomes much worse when the K_α effects are included. While none of the touchdowns were very hard, note that landing B missed the ground on the first pass and then touched down again about 609.60 meters (2000 feet) downrange. The point made by this recording is not that barometric rate causes insurmountable problems but that the system performance becomes very sensitive to another variable which is difficult to bound. Not only are the static source erros with angle of attack considered difficult to define but the possible variations of measured static pressure as a result of ground effects may also be subject to uncertainties.

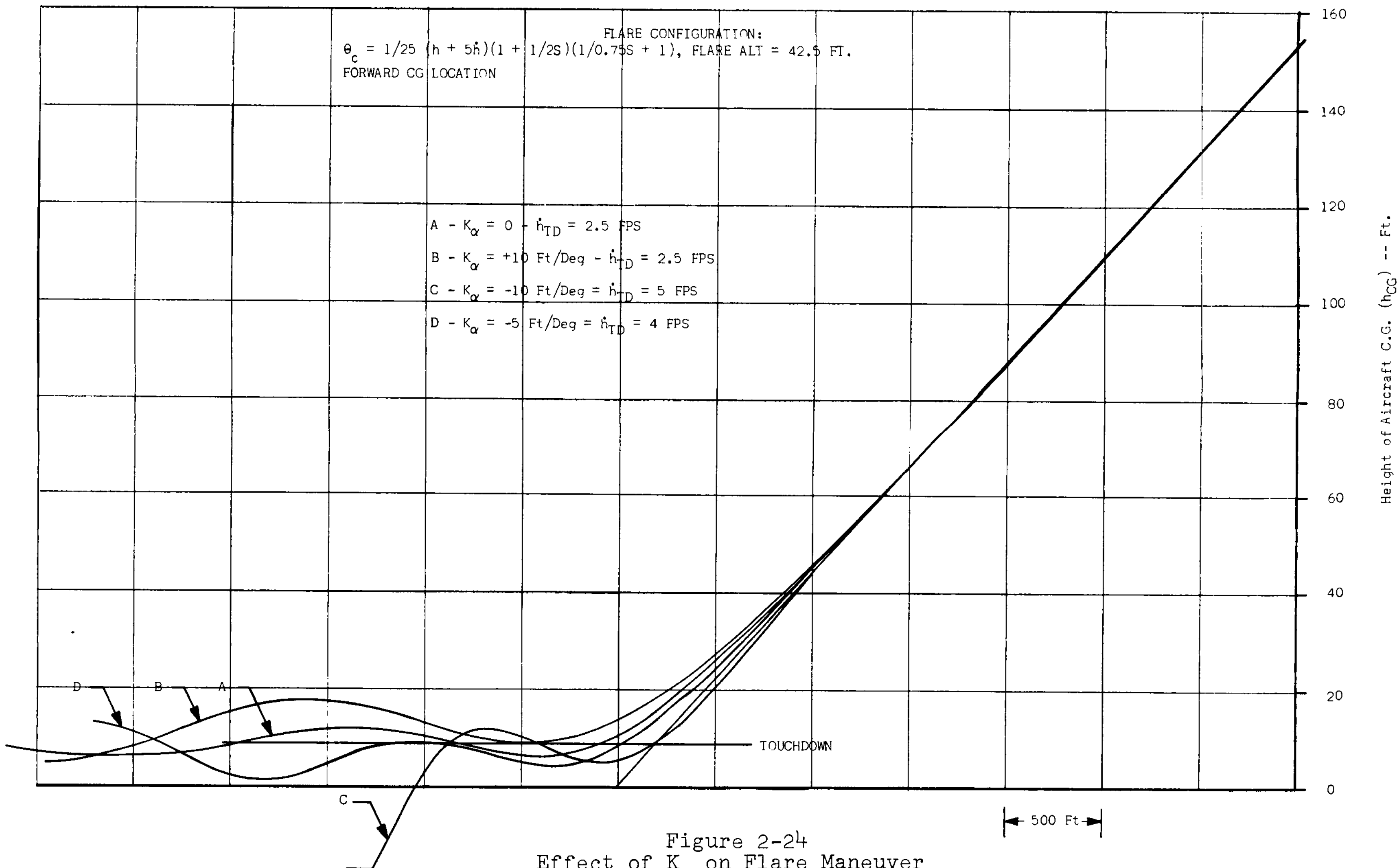


Figure 2-24
Effect of K_α on Flare Maneuver

2.4.4 Touchdown Dispersion

One of the most difficult problems associated with automatic landings or lower minima manual landings is the fact that they involve a significant difference in procedure than used in the normal manual landings. This difference is that the lower minima flareout trajectories start tangent to the glide slope and always remain above the glide slope. Landings from higher altitudes may start tangent to the glide slope, but they depart by going under the glide slope. Thus, the touchdown regions for both types of landings will be considerably different. Runway lighting standards have been designed for the forward area touchdown that would result from ducking under the glide slope prior to executing the flareout. Pilots, concerned with stopping distance, prefer a forward area touchdown. In automatic approaches where the pilot takes over the aircraft at about 60.96 meters (200 feet), the duck under maneuver is still used to recover valuable runway distance. Figure 2-25 illustrates a normal landing procedure performed by KLM airlines in a DC-8 (reference 13). The departure from the glide slope occurs at about 42.67 meters (140 feet) and the immediate response is an increase in the sink rate. If this technique were attempted at a glide slope separation altitude of 30.48 meters (100 feet) or lower, we would be setting the stage for an accident situation; for now, small errors in power setting or human response delays could easily result in a failure to reduce the aircraft's sink rate.

It is therefore fairly well accepted that lower minima touchdowns will have to cover a considerable region downrange of the glide slope intersection with the runway. This causes consternation on the part of pilots who would like to see some additional stopping distance in front of them. Also, the touchdown lighting arrangement does not adequately cope with landings which flare from above the glide slope. In attempts to standardize or establish criteria for autoland touchdown dispersion characteristics, various committees representing the Air Transport Association, Airline Pilots Association, and Aerospace Industries Association have been working with the FAA for some time. An indication that the problem discussed here is difficult to appreciate can be surmised from the ATA's proposed Advisory Circular to the FAA on Automatic Landing System Standards, dated 12/14/66. Aircraft touchdown specifications in this document are:

> Longitudinal Dispersion: -91.44 meters and +304.8 meters (-300 feet and +1000 feet) (95 percent of occasions) from a line on the runway which is the intersection of the linear extension of the glide slope with the runway.

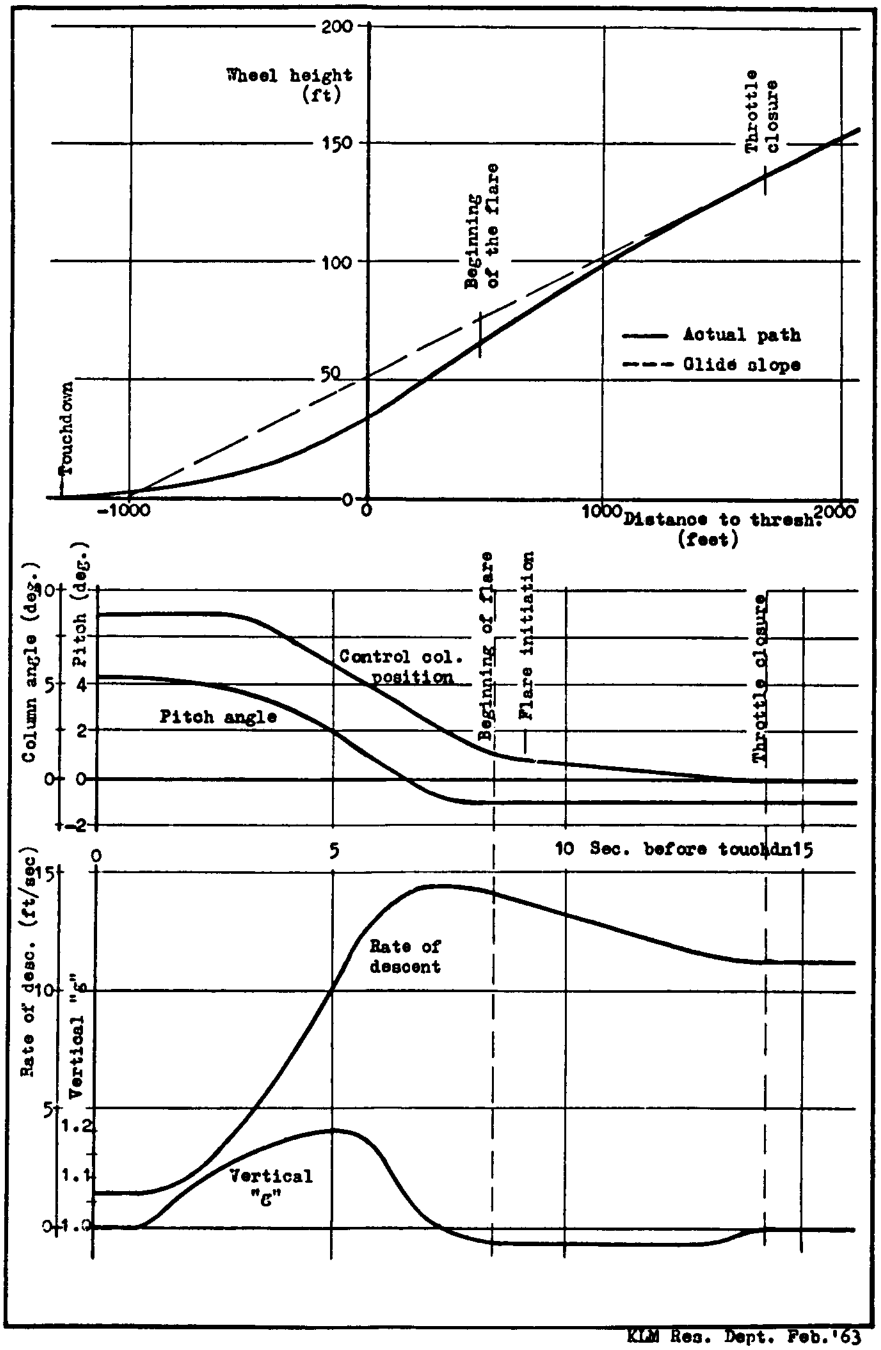


Figure 2-25
The Basic Landing Procedure for a DC-8 Airplane
(Curves Based on Flight Recording)

Under reasonable combinations of the following:

a. Headwind - 25 knots or tailwind of 10 knots maximum

b. Crosswind - 15 knots maximum

c. Moderate Turbulence

d. Windshear - 4 knots/100 feet down to 100 feet (not exceeded 95 percent of occasions)

8 knots/100 feet from 100 feet to ground (not exceeded on 95 percent of occasions).

This touchdown specification is indciated as the shaded region adjacent to a 2.5-degree glide slope on figure 2-26. Also included in this figure is a path that would be followed by an aircraft beginning its flare at about 15.24 meters (50 feet) and gently shallowing its flight path for a 0.30 meter/second (1.0 foot/second) touchdown sink rate. It is apparent that such a landing trajectory not only exceeds the above longitudinal dispersion specification, but actually lands outside of the touchdown zone light region. To minimize runway length consumed during flareout, the flare maneuver must be started at a lower altitude, and higher values of normal acceleration must be used. All of this can be accomplished, but with obvious disadvantages. First, a shorter flareout path means that the maneuver becomes more of an open loop process with less opportunity for the closed loop vernier to effect corrections. Second, and equally important, is the psychological problem of the pilot waiting for the automatic system to initiate the flareout. Pilots do not enjoy waiting for the automatic system to commit itself. They prefer automatic flareouts at the maximum possible altitudes. High altitude flares will consume runway length, as shown in figure 2-26.

The proposed dispersion specification is somewhat unreasonable from the disturbance conditions that are imposed. For example, consider the Boeing 727 simulated landing illustrated in figure 2-20. The nominal touchdown occurred at about 60.96 meters (200 feet) from the glide slope point, well within all requirements. We must recall that these flareout systems are primarily open loop processes and nonpredictable disturbances can cause significant errors. Since an open loop compensation for windshear is not possible, we can expect that type of disturbance to cause significant deviations from the nominal trajectory. This is illustrated in figure 2-27 for a 4 knot/30.48 meters (4 knot/100 feet) windshear, only one-half the value called out in the proposed dispersion specification. Note that when the windshear is in the tailwind direction, the touchdown occurs over 457.20 meters (1500 feet) from the glide slope point. That touchdown was quite soft [0.15 meter/second (0.5 foot/second)] and should be considered satisfactory; yet, the runway consumed is considered excessive. Corrections for this type of problem would most certainly bias the nominal touchdown conditions in the direction of higher sink rates. This is a dilemma

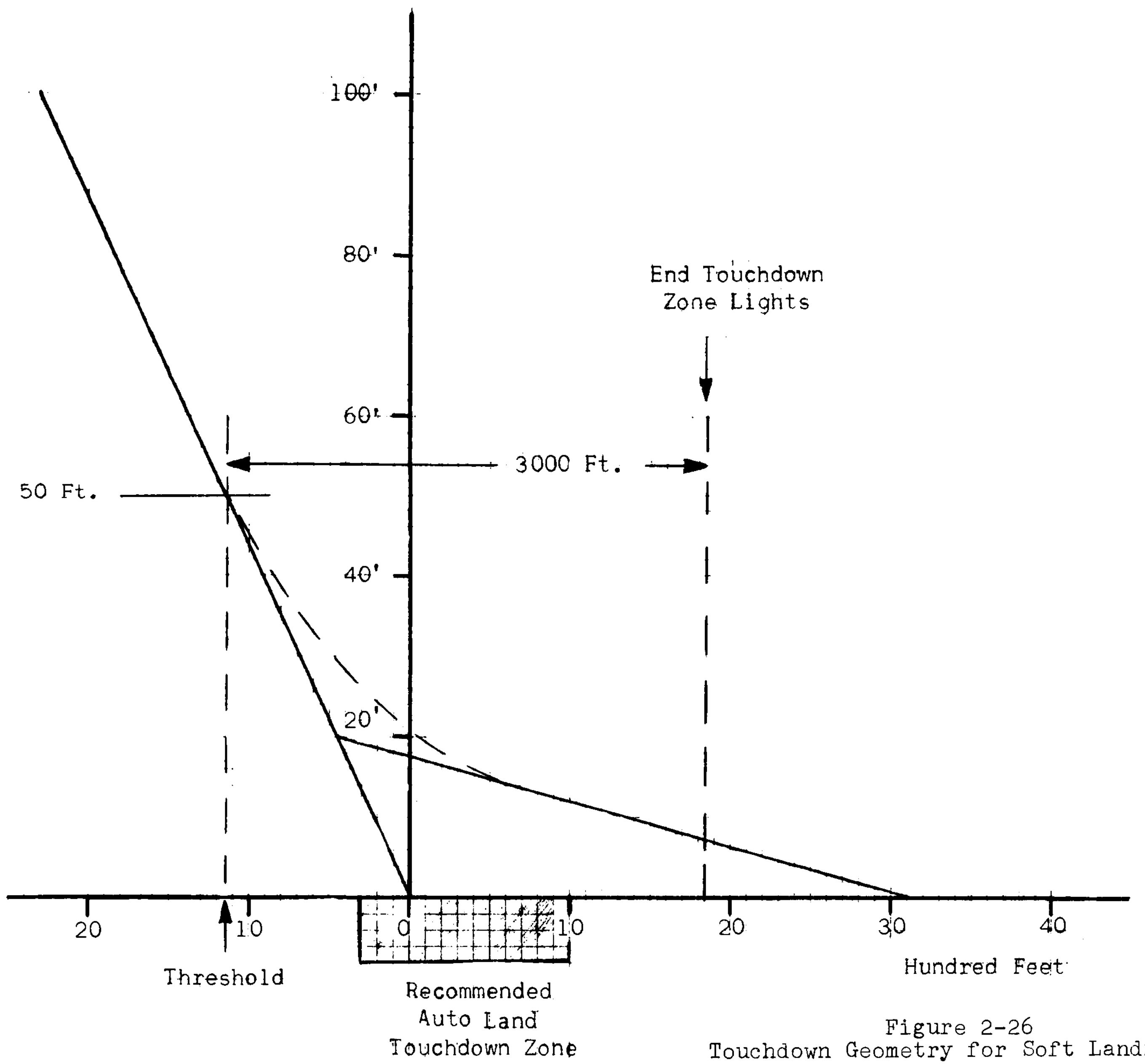


Figure 2-26
Touchdown Geometry for Soft Landing

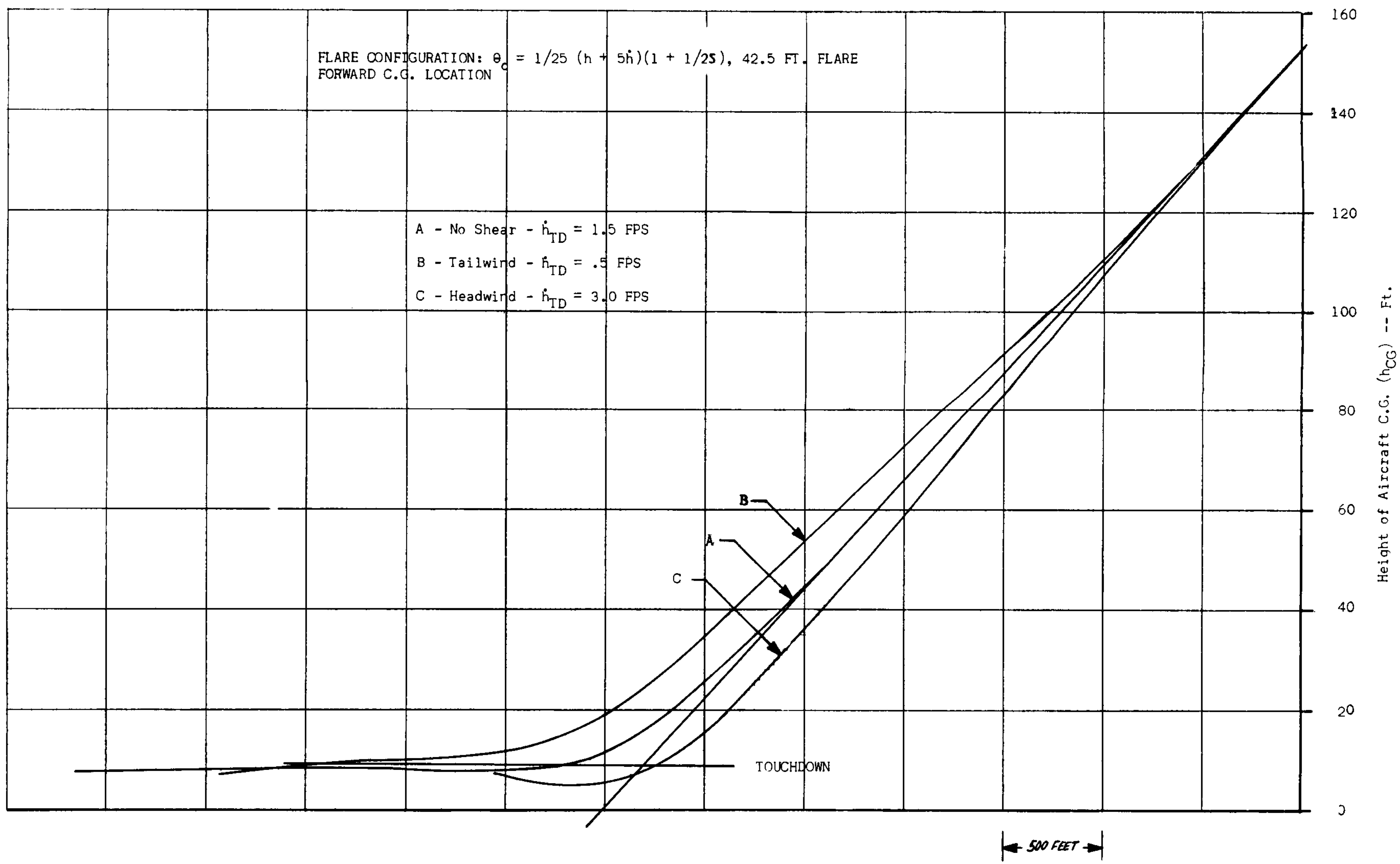


Figure 2-27
Effect of Windshear on Flare Maneuver
(4 knot/100 feet)

that does not appear to be amenable to electronic breakthroughs. It can probably be solved only by runway extensions and many years of operational experience and familiarization with automatic landing system characteristics.

C. THE DECISION-ATTITUDE PROBLEM

1. The Category I, II, and III Standards

The certification of aircraft systems to operate under lower minima conditions has involved an evolutionary pursuit of the IATA categories illustrated in figure 2-28. These categories define decision altitudes as a function of runway visibility range (RVR). Thus, if an aircraft and the related crew and equipment are certified for Category I operations, a descent to 200 feet is permitted before the pilot need make a commitment to abort the approach with a go-around or to take over controls of the aircraft for a manual landing using the airport's visual references. RVR is measured quantitatively on the ground although its accuracy and actual significance at the touchdown area are often questioned. If the airport facility had determined that the RVR is greater than 792.48 meters (2600 feet), then the Category I aircraft will have been given approval to continue its ILS approach to 200 feet; otherwise that aircraft would have been diverted. Little difficulty has been experienced in updating onboard avionics to meet Category I requirements. Airborne equipment requirements are:

- Normal mandatory radio and instrument equipment
- Flight Director or Automatic Approach Coupler (of demonstrated acceptable level of reliability)
- Improved instrument failure warning system or equivalent cockpit procedure ensuring immediate detection of instrument failure

Most of the effort involved in the introduction of Category I operations related to airport equipment. This involved installation of all components required for operational RVR at US airports (high intensity runway lights, standard approach lighting system with flashers, transmissometers, and all weather runway markings or runway centerline lighting.

Category II requirements are similar to those of Category I except that the accuracy problems become much more intense. If the aircraft is allowed to penetrate to an altitude of 100 feet before a visual takeover or a go-around decision is made, the first question to be answered is what will be the source of information that tells the pilot he is at 100 feet. Radio altimeters may be an essential instrument for this function although the terrain peculiarities must be taken into account. Category II certification has been obtained by airlines

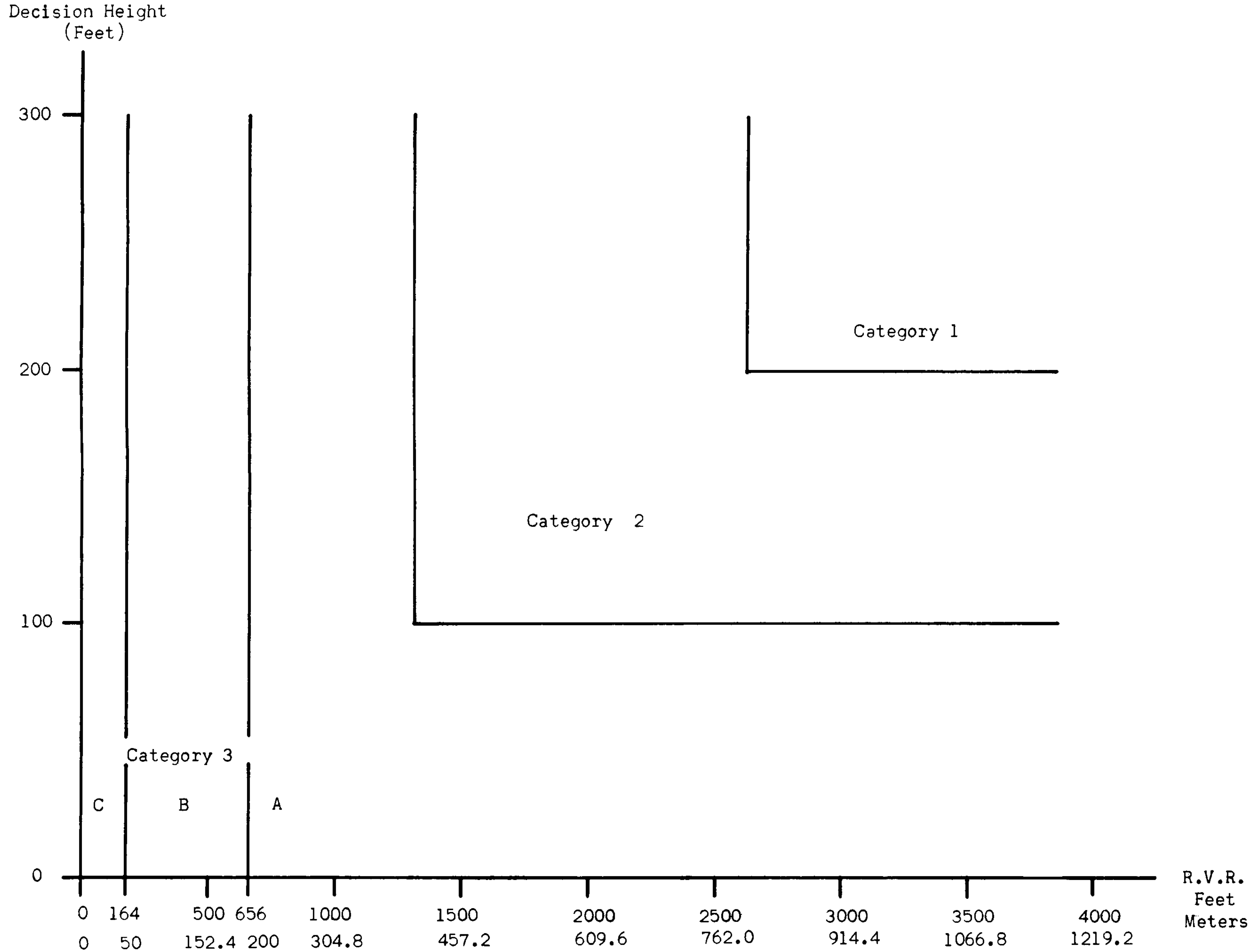


Figure 2-28
IATA Categories

without using radio altimeters as the basic height measurement. This has been done by demonstrating accuracy and reliability of dual barometric altimeters. It also becomes necessary to demonstrate the abort maneuvers from altitudes of 100 feet. The capability of the individual aircraft under engine out conditions and the initial descent rate determine the safety margins of go-around maneuvers. Perhaps most important is the training and proficiency of the pilot and crew. The available time to recover from errors is greatly reduced in Category II situations and coordinated crew responses are essential. The ILS and airport facility standards and the performance of the automatic approach systems have been the major hurdles in achieving Category II certification for various aircraft. (FAA criteria for approval to operate Category II are defined in reference 14.) Figures 2-10 and 2-11, the DC-8 Category II certification flight test records, show the localizer and glide slope accuracy windows that must be attained. To achieve this type of performance, automatic approach computers and flight director computers were improved to include more precise gain programming of the beam signals. Autothrottle systems were added when not already in use. Also, equipment tolerances and accuracies were refined so that they could provide the desired performance and were also compatible with monitoring and failure detection devices. Category II operation has been certified for many aircraft by various carriers so that it would be safe to state that Category II is state of the art in 1967.

The economic advantage of penetrating to lower altitudes than Category II decision altitude is debatable. For example, it has been shown that the introduction of Category I and II operations can take a major cut out of the diversion losses. Figure 2-29 (derived from reference 15) shows the relative incidence of various minima at four airports. These airports are fairly representative although Los Angeles and Seattle, which have a higher incidence of low minima conditions, are not included. It is interesting to note that statistics based on number of hours below specified minima do not provide a direct measure of flight diversions. Some airports, for example, tend to have their fog problems from late evening to early morning hours when the scheduled arrival and departures are not near their maximum density. Other airports that have a more frequent encounter with fog at the peak traffic period will be more vulnerable to economic losses. A summary of the statistics in figure 2-29 yields the following improvements with lower minima equipment:

Minima	Hours Below	Yearly Hours (percent)	Improvement Over 300-3/4 Minima (percent)
300-3/4	270.4	3.1	--
200-1/2 Cat. I	182.2	2.08	33

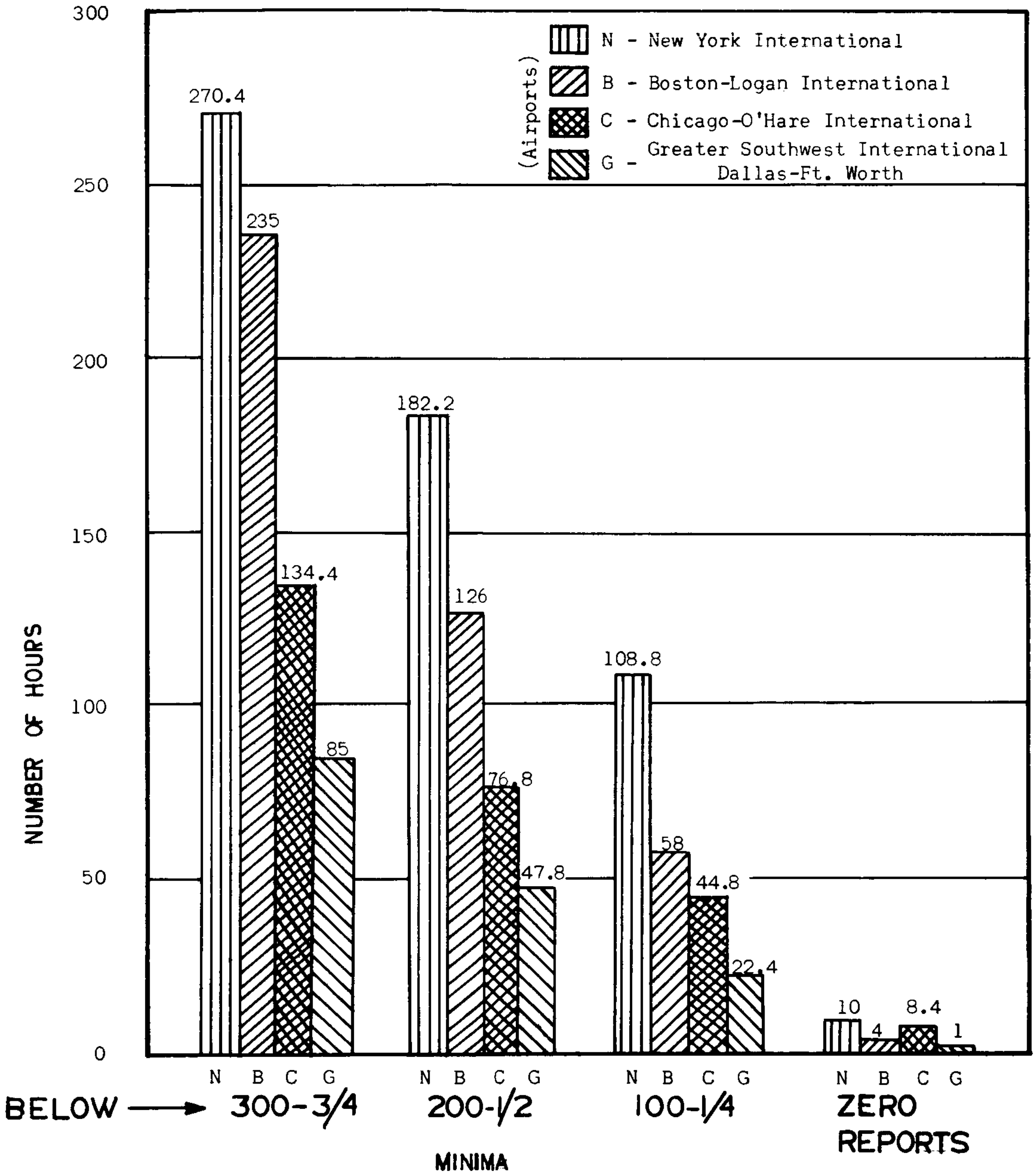


Figure 2-29
Average Yearly Hours Below Landing Minimums
(Also Zero Reports)

Minima	Hours Below	Yearly Hours (percent)	Improvement Over 300-3/4 Minima (percent)
100-1/4 Cat. II	108.8	1.24	60
Cat. IIIa (Autoland) Excludes Zero-Zero Conditions	10.0	0.11	96

For many aircraft, the investment required to improve capability from 60 to 90 percent may not be warranted. However, when one projects to the SST era and considers how a single diversion can disrupt schedules and earning capacity for several flights, the economic necessity of Category III operation is difficult to question.

The demarcation between Category II and Category III in terms of weather conditions is not very significant, but in terms of system requirements it has been a formidable task to identify the various factors and criteria which should be considered. At this time (February 1967), there are proposed requirements for Category IIIA but as yet no definitive specifications or even definitive interpretations of the proposed requirements. For example, the FAA (Federal Aviation Authority and ARB (Air Registration Board - UK) positions may be summarized as follows:

- FAA

  Advisory Circular to be Issued
  Safety Standards Maintained
  (Altitude Loss Factors following equipment malfunction have been specified in FAA Advisory Circular (reference 16). These standards allowed flight path deviations from ILS approach paths in accordance with figure 2-30. For automatic landings the deviation during flare cannot be defined except in qualitative terms such as "excessive" or "unsafe".)

- ARB

Autoland System Design Goal	1×10^{-7}	System Failures/ Landing
Landing Accident Rate Shall not Increase	1×10^{-6}	

The requirements as summarized above do not dictate any specific direction for system implementations. For example, they have been interpreted in the Douglas-Sperry DC-9 Category IIIA Development Program as follows:

- Automatic landing is primary mode of operation

- Flight director is available as a monitor and backup system

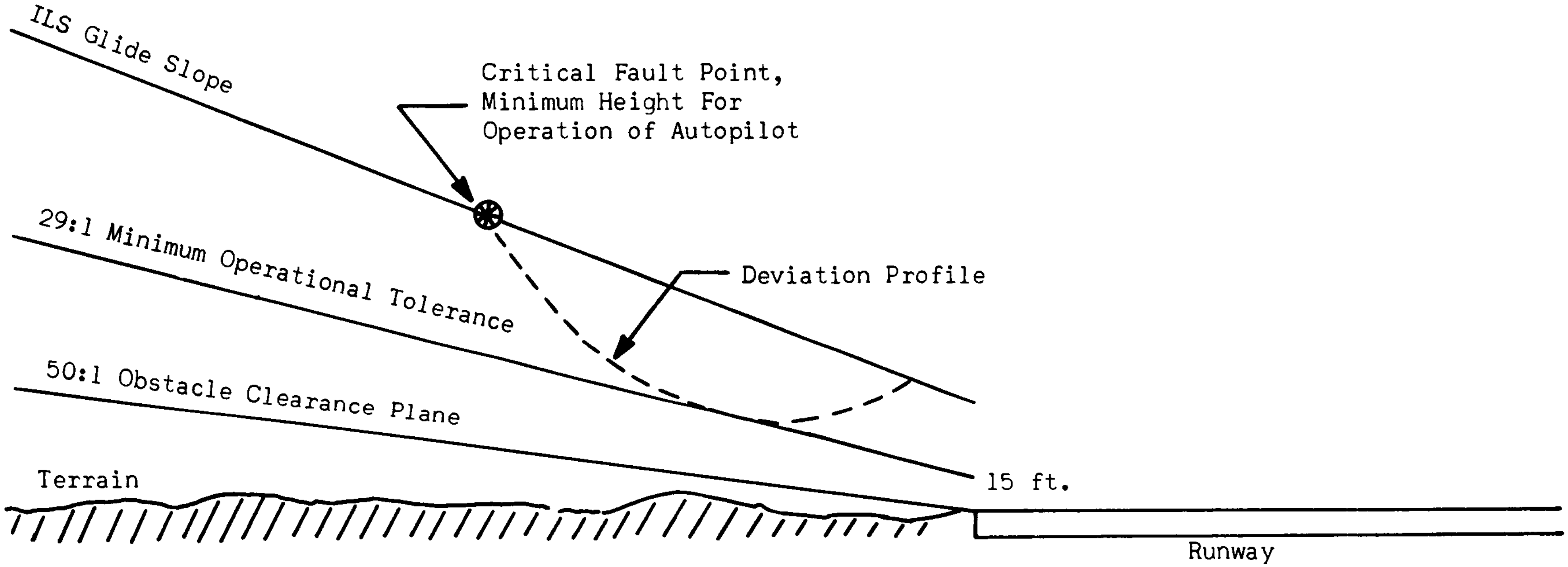


Figure 2-30
FAA Operating Limitations for Autopilots

- Landing system failure above 100 feet.....ABORT

- Automatic landing system failure does not cause appreciable or unsafe flight path deviation

- Manual takeover at touchdown with visual guidance

This particular interpretation was taken because of the relative ease with which the Category II equipment could be updated. The major change between the Category II and III configurations involves the extensive monitoring. Fail-safe operation will be obtained for all failures, but because complete triplication is not used, the configuration is fail-operational for only about 85 percent of possible failures. Thus this interpretation of Category II requirements may necessitate a manual takeover with flight director for the 15 percent failure group if such failures occur below 100 feet. Other approaches do not consider the possibility of a flight director as a backup below 100 feet. In these approaches, the commitment to continue landing after the aircraft has descended below 100 feet depends upon a triplicated, fail-operational automatic landing system (the Smiths system in the Trident, for example).

An interesting observation on the ARB objective of no increase in accident rate with the introduction of all weather landing may be made if we refer to aircraft accident statistics. For example, reference 17 shows the distribution of fatal accidents and major incidents for jet, turboprop, and piston aircraft in recent years. This information is reproduced as figure 2-31. It is apparent that approach and landing represents the major safety hazard in modern aviation. The breakdown of circumstances regarding the approach and landing accidents is given in table 2-3. Note that inaccurate approaches appear to be the largest single contributor. One might conclude that the use of all weather landing guidance and control techniques could have prevented most of these incidents. Thus it is quite reasonable to expect that the introduction of AWL techniques should significantly improve rather than degrade the present record. Indeed, N. E. Rowe, in reference 17, when outlining steps to greater flight safety, lists "Introducing means for automatic landing as a matter of urgency". We therefore see flight safety planners call for automatic landing as an urgent safety requirement and all weather landing planners cautious and hesitant in the introduction of these techniques for fear of degrading flight safety.

Regardless of the system implementation there is always a critical dependence upon the pilot and crew procedures. The tasks of monitoring performance by means of instrument displays and visual real world references and the precise definition of crew procedures in the event of failures or performance anomalies are the critical unresolved problems of Category III operations. These problem areas will be reviewed briefly.

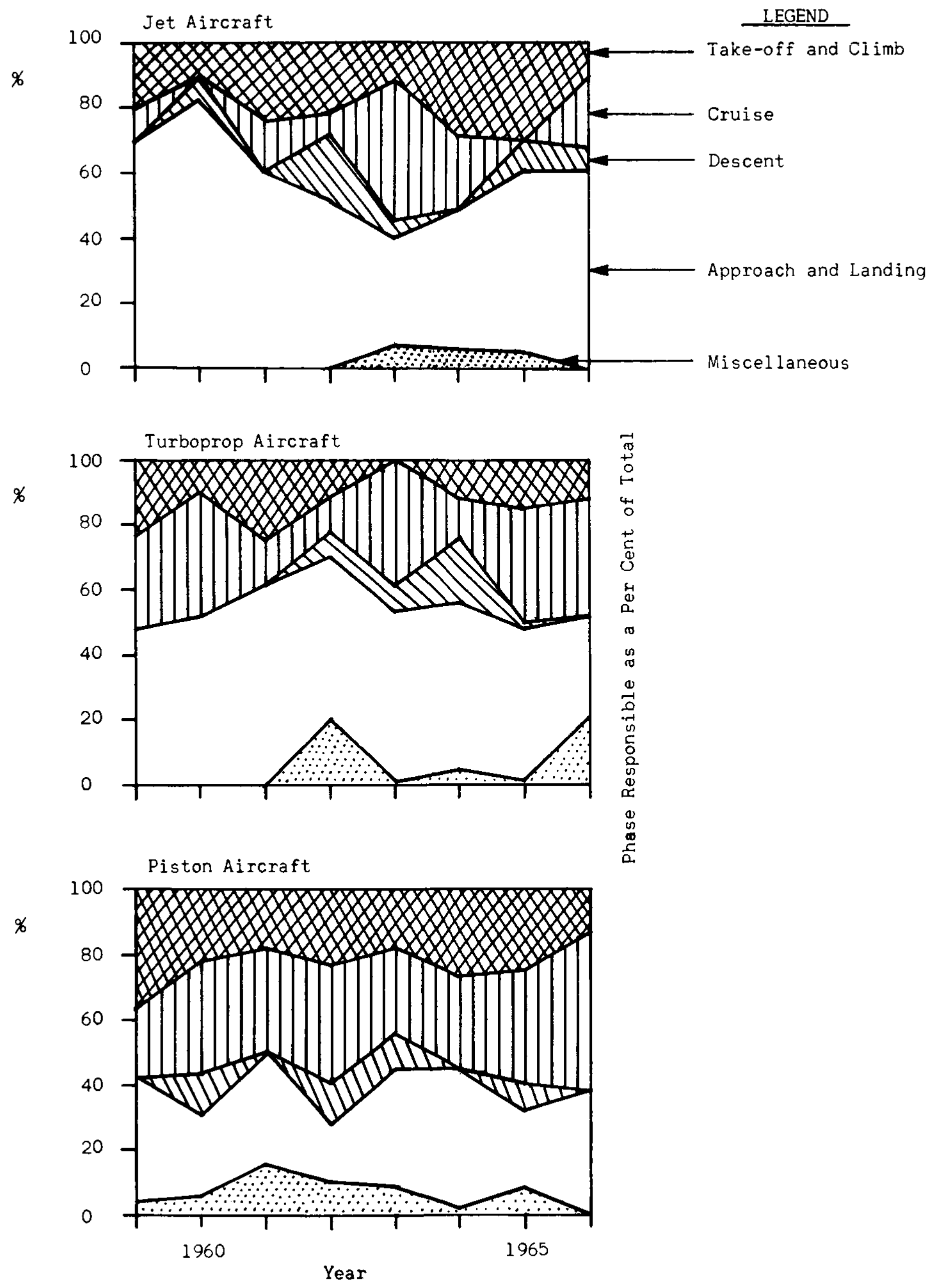


Figure 2-31
Fatal Accidents and Major Incidents by Flight Phase

TABLE 2-3

JET PASSENGER SERVICE FATALITY RECORD

Seven-year totals (1959-1965 inclusive). Training and test flights excluded. Fatalities cover passengers and crew.

Circumstances	Fatalities	Percent of Approach and Landing Fatalities	Percent of All Jet Fatalities
ON APPROACH:			
Hit level ground/water	230	25.6	12.2
Hit high ground/obstruction	539	60.0	28.6
ON LANDING:			
Gear failed to extend	--	--	--
Damaged tire caused landing accident	--	--	--
Undershot	--	--	--
Bounce/hard landing and consequent gear failure	--	--	--
Skid/swerve off runway	17	1.9	0.9
Scraped pod or tip or hit obstruction during landing run	--	--	--
Gear retracted or collapsed during landing run	41	4.5	2.2
Overran			
Crash attempting overshoot	72	8.0	3.8
TOTAL FATALITIES:			
Approach and landing accidents	899	100.0	--
All jet accidents	1886	--	47.7

2. The Information-Decision Altitude Dilemmas

2.1 The Decision Process

The essence of a Category III landing can be described by following the multidimensional decision process along its altitude coordinates. Figure 2-32 shows this viewpoint of the problem at each altitude or instant of time a decision equation must be solved. Figure 2-32 shows the elements of this equation at a time, t_1, and altitude, h_1. The decision, identified as h_1, is seen to comprise three possible outputs. They are as follows:

- Proceed with automatically guided landing
- Manually override system and proceed with landing
- Abort the landing

Each output, in turn, may involve a choice. For example, several abort procedures may be chosen. It is apparent that if we cascade too many alternatives along a given decision output path we would rapidly saturate the decision maker's faculties. A well organized system is one that always operates with a minimum of alternatives. Thus, for example, it may be unwise to offer a choice of abort techniques.

There are six information groupings that may be considered as being the main inputs to the decision making process. As shown in figure 2-32 they are as follows:

- Visual Contact with the Approach Markings and Lights

  Upon receiving adequate visual definition of the landing area, the manual override decision may be made and a manual landing executed.

- Meteorological Status

  These include voice communications of RVR, downwind status, U_W, and crosswind status, V_W, and the onboard interpretation of windshear, $\partial U_W/\partial h$ and $\partial V_W/\partial h$, turbulence and visibility.

- Motion Cues and Controls Response

  These involve sensing of accelerations and observing the autopilot commanded motions of the control column. Evaluation of these cues is usually based on experience and knowledge of what constitutes normal automatic control behavior.

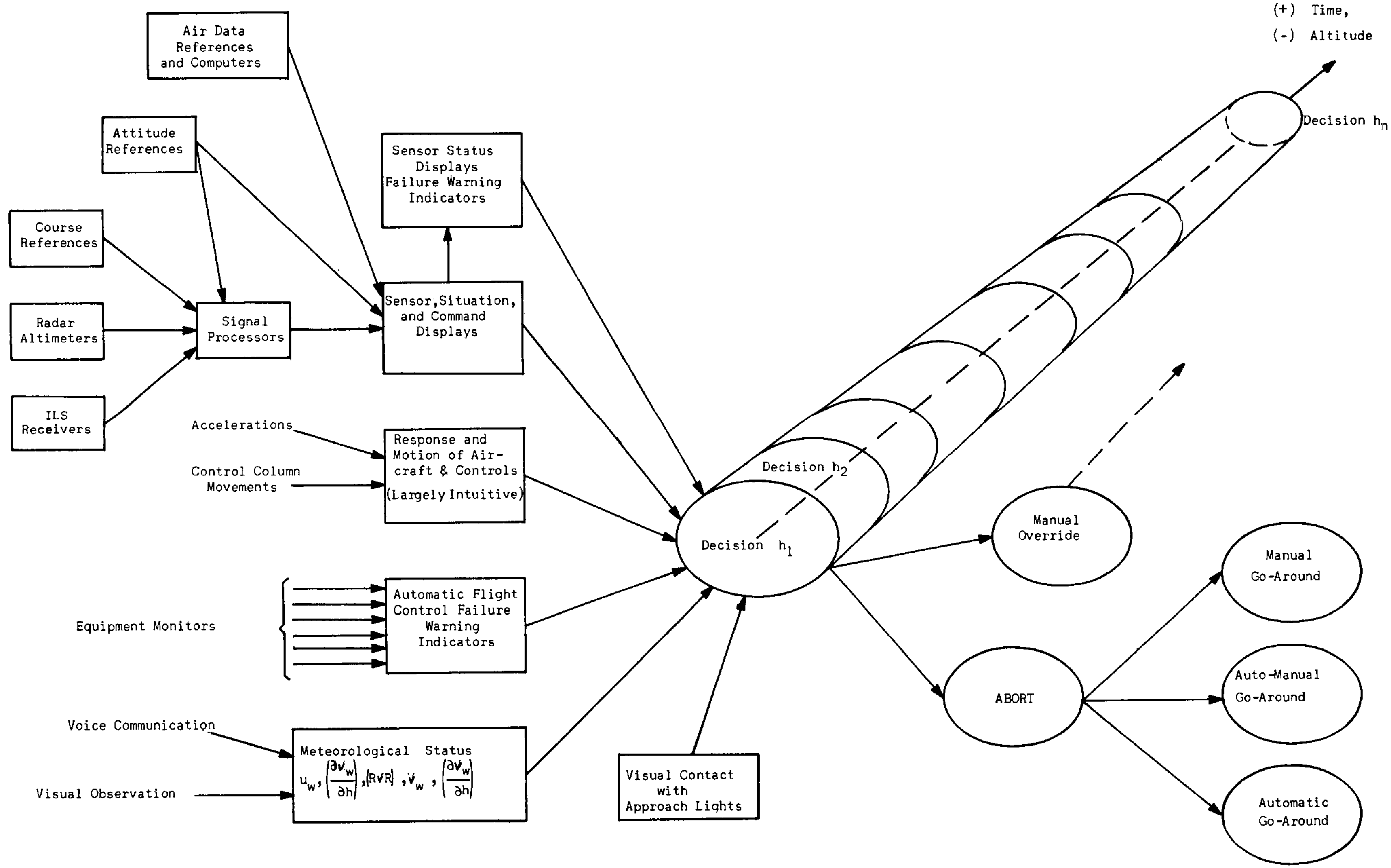


Figure 2-32
The Information Decision-Altitude View of the AWL Problem

- Automatic Flight Control Failure Warning Indicators

  These are specific cues provided by the automatic flight control monitors. Depending upon which phase of the automatic approach has been reached, a specific crew response to each specific failure warning can be defined.

- Sensor Situation and Command Displays

  These are the primary means used by the pilot to monitor the progress of the approach. Displays inform him of his position with respect to the approach paths, attitude, heading, barometric and radio altitude, speed, descent rate, and command errors. Associated with these are progress and mode annunciators which indicate that the proper sequence of automatically switched modes and events are occurring. There is a considerable amount of information redundancy inherent in these instruments. In addition to the built-in failure monitors, the experienced pilot can sense deviations from the norm by incompatibilities between various displays. This is one of the intangible benefits of the pilot's judgment faculties which cannot be mechanized. However, no system will ever be considered for this application if it merely depends upon anything as intangible as pilot's insight which often can be fallible. Specific procedures must be established to define crew action required in response to anomalous situations revealed by the flight control displays.

- Sensor Status Displays and Failure Warning Indicators

  As the use of the flight instruments becomes more critical, the requirement that they be adequately monitored becomes more important. Most flight instruments have always carried warning flags and other status indicators to caution against their usage for certain obvious failures such as loss of power. For AWL operations these are being augmented by more elaborate central instrument monitoring subsystems. Instrument monitors warn against failures of the critical displays and the crew response, again, should be specific for each type of failure warning.

The information obtained from these six groups must be continuously digested in the decision process so that, in effect there is a continuum of decisions...h_1 ...h_n as the aircraft approaches the ground. Some of these h's correspond to major decision altitudes such as the 200-foot Category I and the 100-foot Category II regions. In order to keep this from becoming an overwhelming burden on the pilot's information handling capability, there are obvious methods of weighting the information in terms of priorities and digesting much of the data to assist in making obvious decisions such as those in which an abort is mandatory. For example, the Category II ILS windows are the obvious primary priorities between 200 and 100 feet. If the aircraft has not consistently converged its position to within this window and visual contact with the ground has not been made, an abort would be needed. Likewise, an abort would also be called for if one of the dual pitch autopilot channels displayed a failure above 100 feet and visual contact had not been made.

The decision problem becomes most acute at the Category II altitude. Between the Category II decision altitude and the flare altitude 5 to 7 seconds will elapse. (Refer to table 2-1.) During this period the aircraft has been committed to a landing. This landing will either be automatic (or automanual through the flight director) or will be a manual override in which the pilot intervenes to take over controls from the automatic guidance. The latter action should occur only for the rarest of system failures. In some methods of calculating probabilities of failure, the latter action is considered a catastrophic failure so that a pilot's ability to correct a bad landing at this time will improve the probability of success above the already high value that is generally specified for the automatic landing (one failure per 10^7 landings).

During these final seconds prior to landing, the decision element is secondary for a definite commitment has already been made. In the event of an emergency situation occurring during this period, the emphasis should be less on the decision aspects of the problem and more on crew training and the allocation of crew duties.

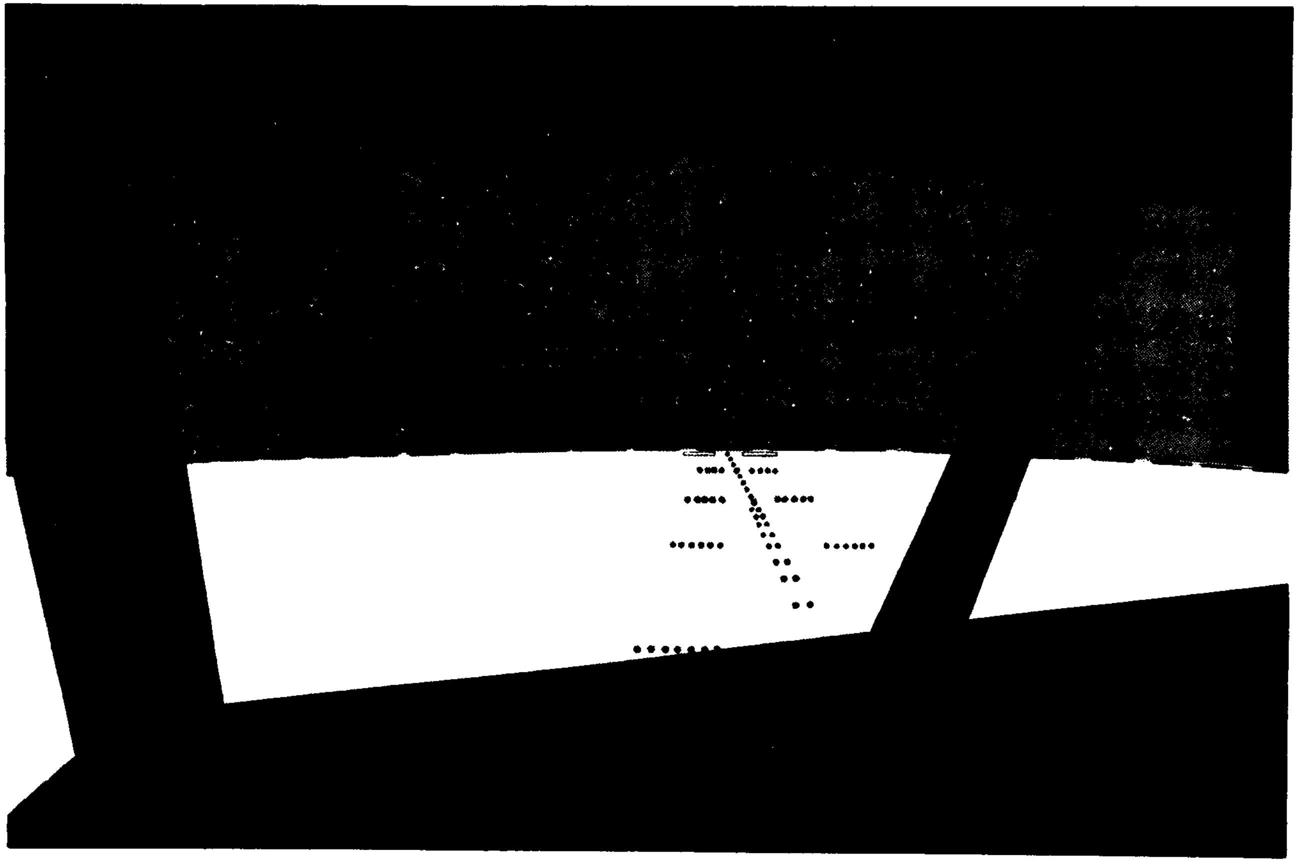
Let us examine the nature of the critical decision at the Category II altitude. If visual contact is made at this decision altitude and all Category II performance criteria have been met, then a transition to manual control and an ensuing successful landing should be accomplished with relative ease. What constitutes meeting Category II performance criteria is therefore the crucial essence of the entire AWL problem. If the aircraft is adequately aligned in terms of its position and kinetic energy at this time,it implies a simple pilot procedure to terminate the landing. In terms of the longitudinal problem it has been shown that it is this initial alignment that determines the character of the automatic flareout. That is, if the alignment is achieved within specified bounds, a successful landing can be executed by the automatic flareout system as well as by manual means.

2.2 Lateral Alignment Accuracy Problems

The lateral problem, however, is somewhat more complicated by the basic accuracy of the guidance information. The allowable relative fore-aft touchdown dispersion is greater than the lateral touchdown dispersion. Runways are

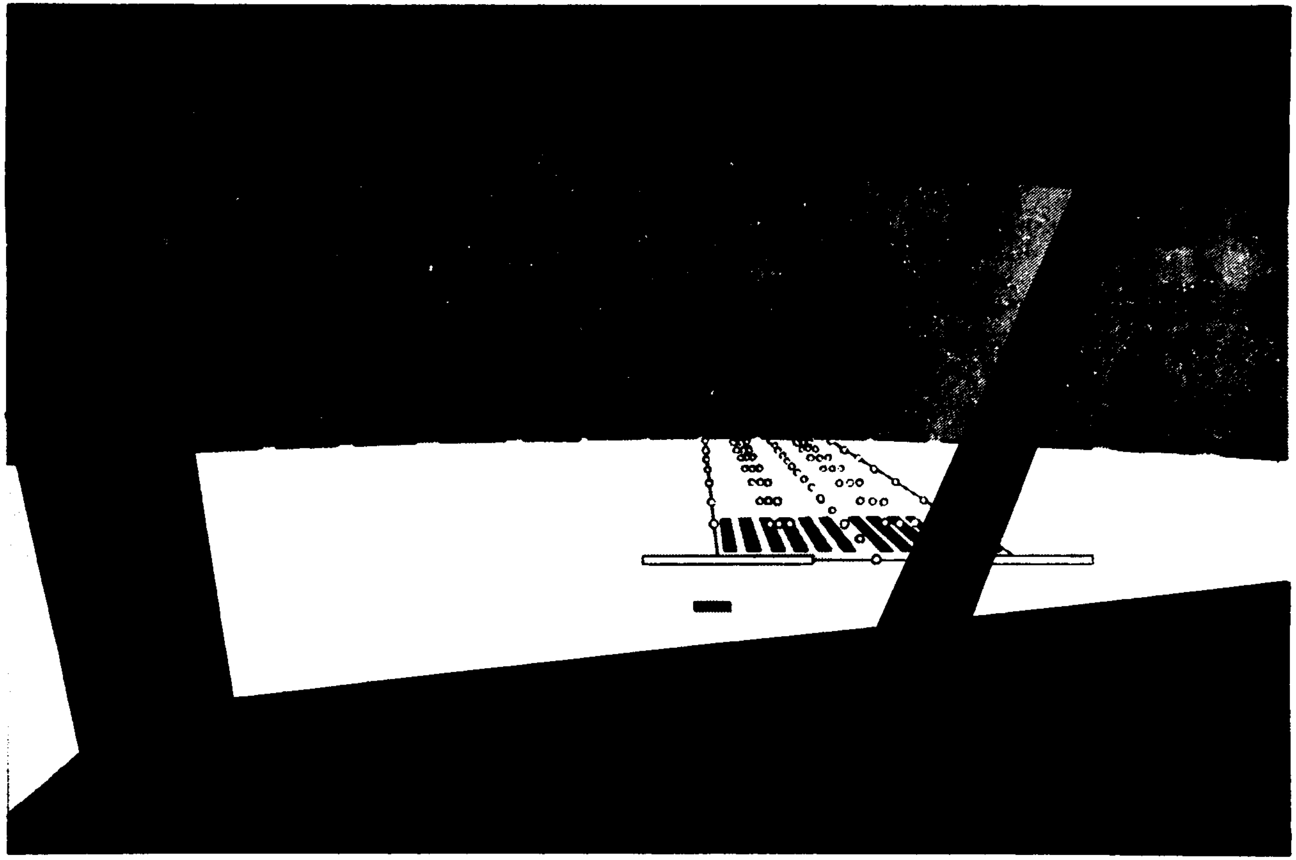
45.72 meters (150 feet) wide and there is little inclination to widen them because aircraft are getting larger. Reaching the 100-foot Category II decision altitude within the specified lateral window still means that lateral corrections may have to be made. (In the longitudinal case, the flight path can proceed to flareout without requiring adjustment.) K. Fearnside and A. P. W. Cane of Smiths Aviation presented an interesting analysis of this problem in reference 18. The problem is essentially one of tolerance accumulations and how they affect the lateral dispersion at touchdown. The pilot's view (derived from reference 18) in Category I weather limits for a 20-percent localizer deflection (30 μa) is shown in figure 2-33. This view of the runway approach lights is seen from the center of a 3-degree glide slope at 200 feet with zero crab angle but with the localizer centerline offset within specification limits. Note that the 20-percent localizer offset is considered to be just outside the acceptable window for Category II. The change in the pilot's view of the runway from the same localizer offset conditions but now at the Category II decision altitude is shown in figure 2-34. Under these conditions, if the landing were committed without a lateral realignment, the aircraft could miss the runway completely. This fact is illustrated by figure 2-35 which is a vertical projection of the glide slope plane intersecting the runway. It shows the position of the 10- and 20-percent localizer deviations from the runway centerline if the present Category II quality localizer is used. Note that the localizer beam tolerances include not only a centerline offset but a fairly significant error in the correlation of position with a given offset indication in microamperes. It is seen from this figure that the 20-percent deviation can miss the runway while the 10-percent deviation can result in a touchdown that is dangerously close to the edge of the runway.

In the Category II situation, the pilot makes a manual landing after establishing visual contact that tells him he is offset from the runway centerline. The question of the pilot's ability to make this lateral correction, referred to as the sidestep maneuver in the United Kingdom work, is the key one for establishing Category II criteria. The results of some of the United Kingdom flight evaluations of this problem (references 18 and 19) are shown in figure 2-36. This figure shows a shaded region of localizer deviations from which pilots made acceptable manual alignments for proper landings. The fact that the recovery limits is a band rather than a line is partially due to the range of bank angles used. Note the more limited ability of a typical autopilot to correct offsets. This is caused by the usual practice of restricting autopilot bank angle commands to about 5.0 degrees during the final phases of an automatic approach. It is this specific autopilot restriction that results in favorable pilot reaction to the so-called supervisory override mode. In this mode, pilot-inserted control wheel force commands add to the autopilot's normal combination of control signals. If the autopilot's flight path corrective capability is slowed by the bank limit restrictions, the supervisory override



WHEEL HEIGHT = 200 FEET
3 DEGREE GLIDE SLOPE
20% LOCALIZER SCALE DEFLECTION
LOCALIZER BEAM OFFSET=15 FEET AT RUNWAY THRESHOLD

Figure 2-33
Pilot's View in Category I Weather Limits



WHEEL HEIGHT = 100 FEET
3 DEGREE GLIDE SLOPE SET TO 60 FEET OVER THRESHOLD (MAXIMUM GLIDESLOPE TOLERANCE)
20 % LOCALIZER SCALE DEFLECTION
LOCALIZER BEAM OFFSET = 15 FEET AT RUNWAY THRESHOLD

Figure 2-34
Pilot's View in Category II Weather Limits

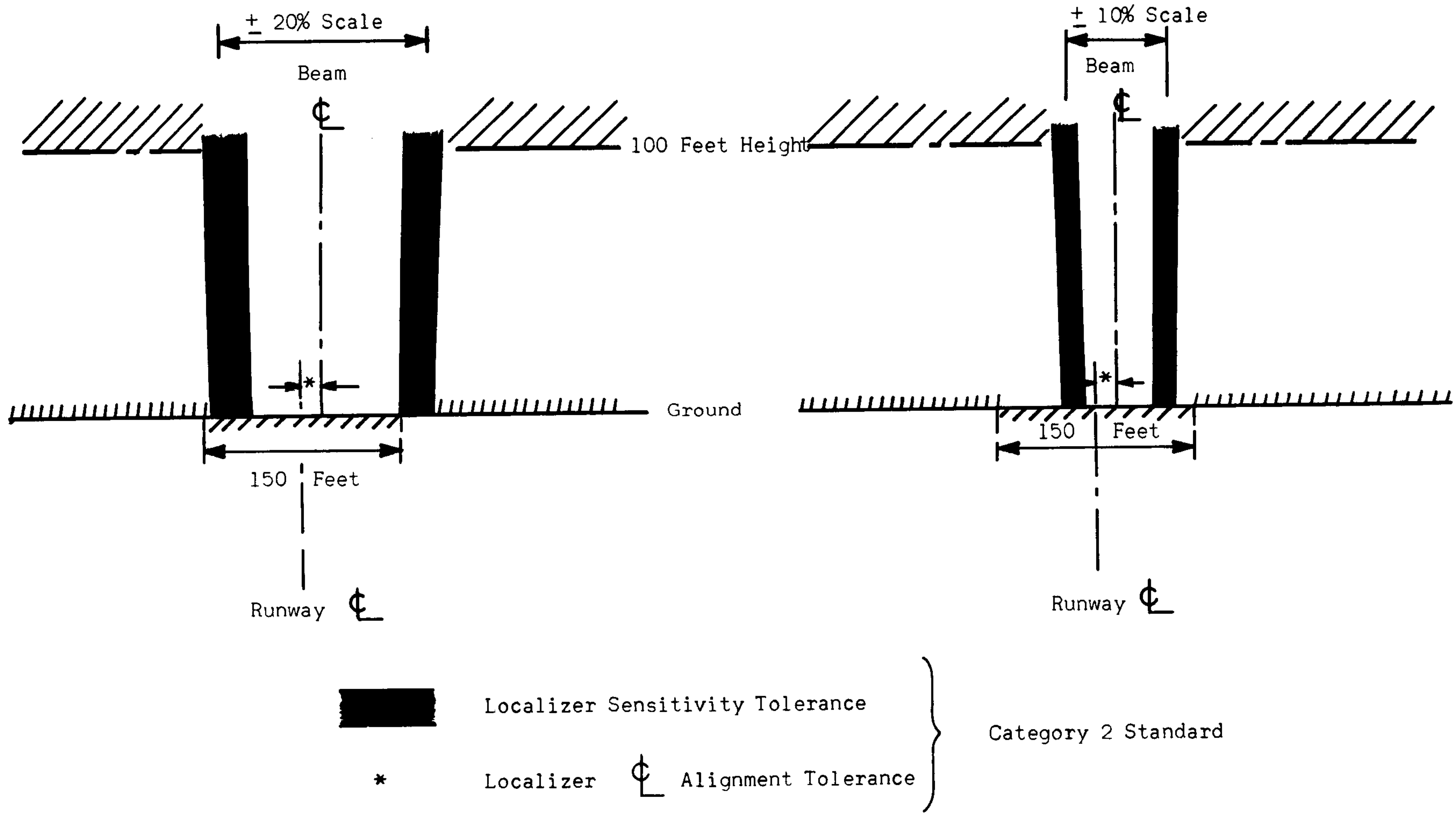


Figure 2-35
10 and 20 Percent Radio Deviations Relative to Runway Edge

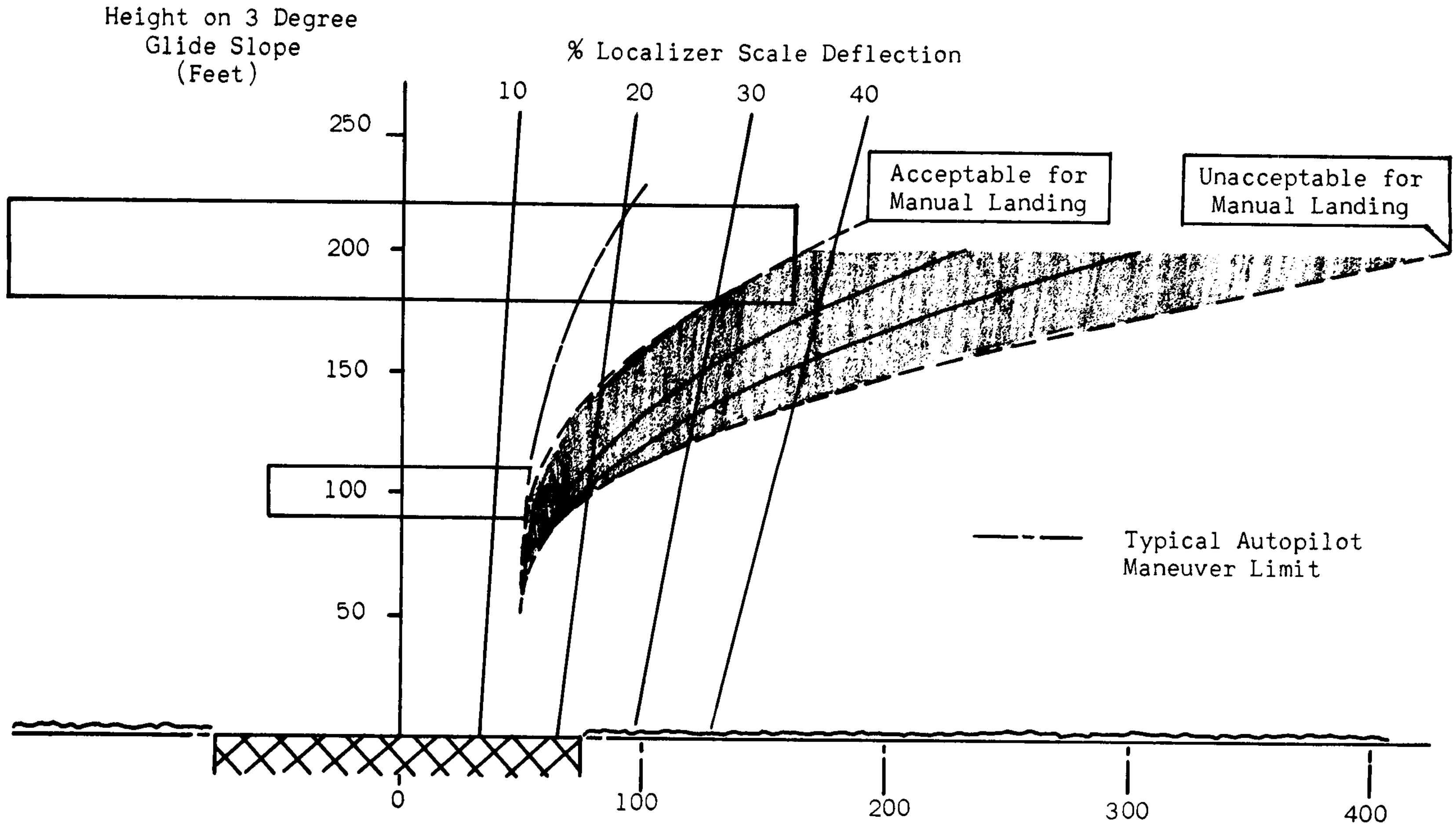


Figure 2-36
Lateral Displacement from Runway Centerline (Feet),
Visual Acceptance Windows

capability allows the pilot to provide a manual compensation for this limitation. If the autopilot's bank limit is raised to that which would be used by the pilot, the supervisory override function would lose its utility.

Figure 2-36 indicates how Category I and Category II acceptance windows relate to localizer guidance accuracy. They are primarily based on the ability of the pilot to recover from errors after visual contact is established. It is seen that the Category II lateral window is not adequate for automatic control to touchdown, especially when the tolerances shown in figure 2-35 are considered. The problem reduces to the accuracy of the localizer. A 10-percent offset would probably be an attainable control accuracy but the variation in the actual location of a 10-percent localizer offset on the runway is a problem. While localizer accuracies can be improved in some specific sites, the fundamental limitations of the localizer radiation techniques have probably been reached. One solution is to widen the runway in the touchdown area. This however is not an avionics solution. There is a strong consensus that the ultimate source of lateral guidance for Category III automatic landings will have to come from techniques other than the existing type of localizer. Meanwhile, the lateral guidance situation for Category III can be considered marginal at best. Some approaches to the problem make use of the split axis autopilot concept in which the longitudinal controls are fully automatic while the pilot controls the lateral flight path manually or with an automanual mode. (The automanual mode is control wheel steering through the autopilot. It can provide the advantage of improved aircraft stabilization and hence better aircraft handling qualities.) The key to this type of operation is the flight control display and how it is utilized in relation to visual cues.

2.3 The Heads-Up and Heads-Down Display Problem

As long as the pilot performs the landing manually as in Category II operations, the pilot is faced with the problem of transition from instrument to visual real world references. The time delays, transient disorientations, and other hazards associated with this critical phase of the landing have been studied, quantified, and evaluated in the extensive human factors research that has been performed on this problem. A strong case has been made for presenting the flight control displays on the windscreen so that the onset of visual contact involves a smooth and natural transition. Thus, if a symbolic runway is projected to align with the pilot's view of the real runway, it is claimed that the ideal integration of automatic guidance and manual control can be accomplished. Windscreen displays have been developed and, in some military applications, are relatively well advanced toward operational use. There are problems with the windscreen concept and some of the solutions are not too easily applied to commercial aircraft (helmet projectors, for example). Evaluation of heads-up displays has been hampered by space restrictions in the aircraft cockpit. It appears that space will always be a difficult problem but one that could be solved if provision for the projection method is made during the cockpit design period. Other problems in the heads-up display technology are in the area of symbology and human factors regarding allowable movements of the symbols.

The question of heads-up and heads-down displays raises the larger question of cockpit operating procedures. The various airlines establish their own cockpit practices in regard to the allocation and division of crew duties. Moreover, the cockpit layout of each aircraft type varies from one airline to another. Much to the aircraft manufacturer's distress, each airline customer has his own ideas about cockpit layouts and flight instruments so that standardized cockpits do not now exist. Whether the present division of crew duties is adequate for the environment of more sophisticated automation when the pilot acts as flight manager has been questioned. The lack of well defined assignments for Category II and III operations has been a major concern of many groups, especially pilots (reference 20). In some approaches to the problem there is a heads-up pilot (usually the captain) who takes over control manually when visual contact is made. Prior to this, the copilot monitors the heads-down flight instruments while the aircraft is being flown by the autopilot. When the flight director mode is the primary means of approach flight control, the pilot may control the aircraft heads-down. The procedures for manual takeover in the event of failure warnings must then be carefully defined. Many pilots have indicated that a heads-up display could minimize the procedural problems. However, we are now approaching an era in aviation development when a more fundamental question may be asked. This question could invalidate the arguments for heads-up displays. For aircraft such as the SST and even the jumbo jets of the Boeing 747 category, the advantage of landing an aircraft using the visual references obtained from the windscreen is in doubt. Any other approach would be a revolutionary challenge to aviation tradition, but it is being suggested that the task of positioning the landing gear on the runway is becoming too difficult a procedure for a pilot observing the runway from the windscreen but located so remotely from the landing gear.

It is perhaps sacrilegious to suggest that a solution to the problem could be obtained by an aircraft design concept such as that shown in figure 2-37. Here the primary cockpit location is near the tail of the aircraft with a direct view of the landing gear and runway for approach and takeoff. A secondary crew location is at the nose. This second station, if desired, can serve for navigation monitoring and possibly as a primary location for a cruise pilot. The advantages from the viewpoint of mechanical flight control designs will also be enormous but it is doubtful that such a shattering of aircraft tradition could be introduced before avionics will find a way to cope with the problem for the presently envisioned aircraft configurations. Thus, in lieu of locating the pilot behind the landing gear, the Boeing Company SST project is experimenting

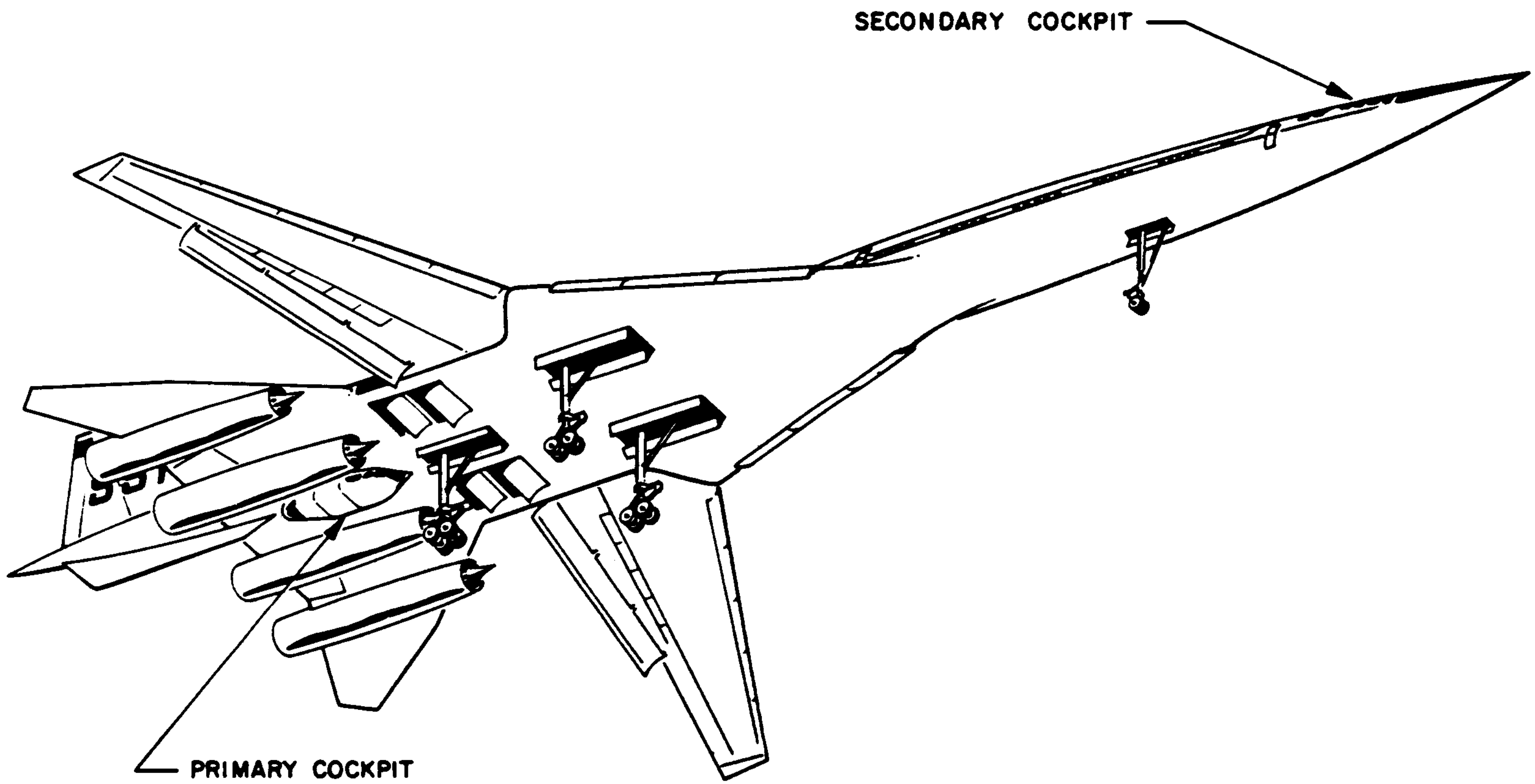


Figure 2-37
A Non-Avionics Solution to the Problem of
SST Landing Handling Qualities

with operational concepts based on mounting a television camera in that rear location. The aircraft would then be flown heads-down with reference to the television display. Combined with this television display would be an electronic attitude director for automatic guidance in low visibility situations. Such a concept represents the most severe challenge to avionics reliability. If an aircraft is normally landed with reference to this remote camera view, can an electronic failure be tolerated? If it can, the pilot must land the aircraft by conventional means, obtaining his visual references through the windscreen. How does he obtain proficiency at this task which is admittedly too difficult for use as a primary operating technique (otherwise the television technique would not have been used)? It is ironic that the more we depend upon avionics, the greater is the demand on pilot's flying skill, and the more difficult it becomes to ensure that skill.

D. AN APPRAISAL OF LESS CONVENTIONAL TECHNIQUES FOR LANDING GUIDANCE

The preceding discussions of the landing problems have stressed guidance and control concepts that would be considered conventional because they are in operational use. Also, the analytical design techniques used to synthesize and evaluate these systems are based on the automatic control theory that reached its maturity in the 1950's. Control theory and guidance concepts have spread out in many directions since that time. It would be appropriate to inquire whether any of the newer techniques are applicable or offer any advantages in the AWL problem.

An extensive technology in the area of explicit guidance has been developed for the missile steering problem. Explicit guidance makes use of the knowledge of the vehicle's desired final state and its instantaneous state to continuously compute a steering vector that will satisfy the terminal requirement. This provides an infinity of permissible flight paths always depending upon the instantaneous conditions. The applicability of this type of guidance to an aircraft's automatic approach phase of flight can be rejected for philosophical reasons. With explicit guidance, the aircraft could be following any path toward the runway. Not only are there practical problems (such as throttle control difficulties), but the very concept of an automatic approach involves the accuracy of the aircraft's adherence to a fixed path in space. The preceding discussions have emphasized the fact that the precision of the approach or landing alignment phase is the essential criterion for the ensuing landing maneuver. The concept of a decision altitude is based on the pilot being able to judge whether he has achieved an adequate alignment; hence the concept of the Category II glide slope and localizer windows. With explicit guidance, there is no absolute standard by which the future success can be appraised other than the judgments of the onboard computer. One consequence of this is that a ground-based radar monitoring system could not determine whether the aircraft's approach trajectory is adequate for landing. Using the ILS fixed path in space concepts, a ground monitor can report misalignments from the localizer and glide slope. Likewise, the pilot can determine whether he has achieved his desired path and whether he is holding it adequately. Thus, it would be very difficult to argue for an approach guidance concept that does not use the fixed path in space approach. It has been shown that the control problems associated with this concept are of a secondary effects nature. The major source of error was related to the accuracy deficiency and other peculiarities of the beam position data.

The flareout problem, on the other hand, does not have a fixed path in space requirement. Explicit guidance techniques could be applied to this phase of the problem without incurring any philosophical rejections. Two approaches to the flareout other than the conventional concepts described previously will be examined here. The first is an application of optimization theory. The second is the terminal controller concept.

1. The Application of Optimization Theory to Flareout

Most of the literature on optimum control theory is more concerned with the mathematical elegance of the technique than with the problem being solved. For this reason, the rather trivial second-order system is usually studied and the problem situations are contrived to demonstrate the theory. In essence, an optimum control system as defined by modern control theory, has very little to do with an optimum in terms of practical requirements. It is optimum because the control activity is always based on a closed form performance index. The index is arbitrary, generally based on an integral a quadratic error criterion. The interpretation of error when the problem is of the multiloop nature is also arbitrary since there is no way in which an optimum weighting of the various factors can be defined. A good summary of different techniques found in modern control theory is given in reference 21.

Since we are concerned with flareout, it is fortunate that one of the best expositions of the application of optimization techniques used the aircraft flareout as an example problem. In this work by Ellert and Merriam (reference 22), the Merriam Parametric Expansion method is applied to the closed loop control of a flareout trajectory. In this particular approach, the authors defined a reference altitude trajectory or desired altitude as a function of time. This desired altitude, $h_d(t)$, is shown in figure 2-38. It is initially an exponential

$$h_d = 100\ e^{-t/5} \text{ for } 0 \leq t \leq 15 \qquad (2\text{-}36)$$

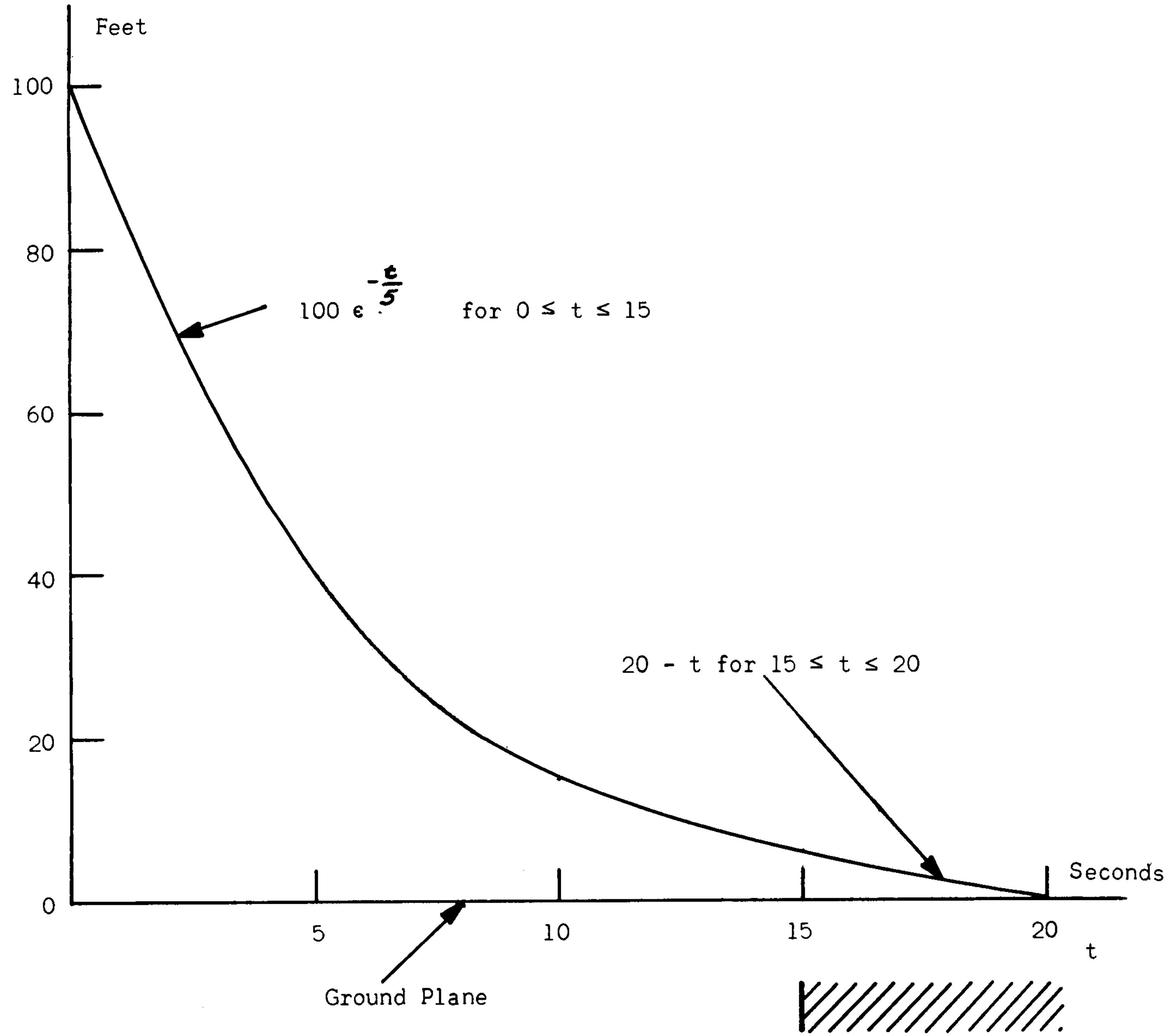


Figure 2-38
Reference Trajectory for the Ellert-Merriam
Optimized Landing System

and after the nominal path reaches the runway, the final path is a straight line (constant rate of descent) equal to

$$h_d = 20 - t \text{ for } 15 \leq t \leq 20 \tag{2-37}$$

A flight path normally exists in space, not time. However, optimization techniques are based on linear differential equations in the time domain so that only time parameters have theoretical use. The time space relationship is easily solved if velocity were constant. Ellert and Merriam recognized that one restriction on the validity of their results was the assumption of constant velocity. They assumed that a throttle control could minimize the speed perturbations and thus speed variations would cause only secondary effects. (Refer to previous discussion under Throttle Controls regarding throttle authority restrictions and their use for long-term speed adjustments only.) However, they did not recognize that aircraft land at speeds that are selected to correspond to their weight at the time of landing. To cope with this problem, equations (2-36) and (2-37) would have to be shifted for each velocity. This would only add minor complications to the digital program which is probably near the capability limit of the most powerful machines if a real time solution is attempted. More significant, however, is the fact that most aircraft flareouts deliberately exploit the speed decrease associated with the flare path change and in some procedures, throttles are actually cut back during the flareout.

In the formulation of the problem, the equation of state

$$x'(t) = Bx(t) + C\overline{M}(t) \tag{2-38}$$

is based on four measurable state variables that define the x(t) state signal vector:

$$x(t) = \begin{bmatrix} x_1(t) \\ x_2(t) \\ x_3(t) \\ x_4(t) \end{bmatrix} \tag{2-39}$$

where $x_1(t)$ = pitch rate, $\dot{\theta}$ or θ'

$x_2(t)$ = pitch angle, θ

$x_3(t)$ = vertical speed $\dot{h}$ or h'

$x_4(t)$ = vertical position, h

The control vector, $\overline{M}(t)$, is the elevator deflection (assumed to have ideal characteristics as a rigid body, moment producing control only). The B matrix and C matrix

$$B = \begin{bmatrix} b_{11} & b_{12} & b_{13} & 0 \\ 1 & 0 & 0 & 0 \\ 0 & b_{32} & b_{33} & 0 \\ 0 & 0 & 1 & 0 \end{bmatrix} \tag{2-40}$$

$$C = \begin{bmatrix} c_{11} \\ 0 \\ 0 \\ 0 \end{bmatrix} \tag{2-41}$$

are defined in terms of coefficients that are based on the simplified second-order representation of the pitch rate response to an elevator deflection

$$\dot{\theta}(s) = \frac{K_S(T_\gamma S + 1)}{\left[\left(\frac{S^2}{\omega_s^2} + \frac{2\zeta S}{\omega_s}\right) + 1\right]} \delta_e(s) \tag{2-42}$$

and the relationships used in the simplified flight path control analysis given previously as equations (2-14) and (2-15). In terms of these simplifications, the coefficients of the B and C matrices are given in the following table

$$b_{11} = \frac{1}{T_\gamma} - 2\zeta\omega_s \qquad b_{32} = \frac{V}{T_\gamma}$$

$$b_{12} = \frac{2\zeta\omega_s}{T_\gamma} - \omega_s^2 - \frac{1}{T_\gamma^2} \qquad b_{33} = -\frac{1}{\tau_\delta}$$

$$b_{13} = \frac{1}{VT_\gamma^2} - \frac{2\zeta\omega_s}{VT_\gamma} + \frac{\omega_s^2}{V} \qquad c_{11} = \omega_s^2 K_s T_\gamma$$

Note that the term K_s in equation (2-42) is a function of the pitching moment coefficient, C_{m_δ}, which is usually very nonlinear for large excursions, especially those that were encountered in the reference 22 study. K_s is also a function of dynamic pressure and will therefore change because of the speed changes that must occur.

The optimization process consists of minimizing an instantaneous measure of errors in the various state variables. For the landing problem the appropriate error measure is of the form

$$e_m(t) = \phi_h(t)[h_d(t) - h(t)]^2 + \phi_{\dot h}(t)[h_{d'}(t) - h'(t)]^2$$
$$+ \phi_\theta(t)[\theta_d(t) - \theta(t)]^2 + \phi_{\theta'}(t)[\theta'_d(t) - \theta'(t)]^2 \qquad (2\text{-}43)$$
$$+ [\delta_e(t)]^2$$

where $\phi_h(t)$, $\phi_h'(t)$, $\phi_\theta(t)$, and $\phi_{\theta'}(t)$ are the time varying weighting factors which indicate the relative importance of the various terms in the error measure. Various rationalizations were used to select a proper relationship between weighting factors. Of three sets of weighting factors chosen for study in reference 22, two consistently performed badly while one gave generally good results. The one that worked used a constant value for ϕ_h with all other ϕ's zero. That is, the only criterion was conformity to the reference path of figure (2-38). The error index used was

$$e(t) = \int_t^{20} \{\phi_h[h_d(\sigma) - h(\sigma)]^2 + [\delta_e(\sigma)]^2\} d\sigma \qquad (2\text{-}44)$$

where σ is a dummy time variable. The mathematical optimization process is a closed form solution which provides a control, $\delta_e(t)$, that minimizes the integral in equation (2-44). The use of Merriam's Parametric Expansion method to solve this problem yielded an optimum elevator deflection as a function of time:

$$\delta_e(t) = \omega_s^2 K_s T_\gamma [k_1(t) - k_{11}(t)\theta'(t) - k_{12}(t)\theta(t)$$
$$- k_{13}(t)h'(t) - k_{14}(t)h(t)] \qquad (2\text{-}45)$$

In this expression the various k's are feedback gains of the state variables while $k_1(t)$ is equivalent to the combination of the desired reference values. An extraordinary observation can now be made. The feedback terms are identical to those used in the conventional state-of-the-art systems discussed previously; namely h, $\dot h$, θ, and $\dot\theta$. Moreover, of the various weighting function strategies tried, the only one that did not result in aircraft crashes was the one that yielded constant gains for these feedbacks. That is, the gains were constant except during the last 2 seconds when they all decayed to zero. At that time the errors in the zero disturbance environment were zero so the problem solution called for gains of zero. This is, of course, completely unacceptable since the aircraft is now totally uncontrolled and unable to correct for any disturbances (gusts and ground effects).

An important question must now be resolved. Are the results of this system synthesis procedure to be used as the basis for a landing system design or does the landing system design include the synthesis process as a sort of adaptive mechanization. If this is intended only as a design tool, then there can be little argument except to note that it is philosophically wrong to attempt the design of a system using such unreasonable simplifications. The design of an aircraft landing system can only be accomplished by including all of the so-called secondary effects, disturbances and initial condition variations since it is these factors that determine success or failure of a system. If the synthesis procedure is to be part of the onboard mechanization, then the question of what is being bought for the enormous complexity must be raised. The remainder of this discussion assumes that sufficiently powerful onboard computers are available for the in-flight synthesis task and that this is the desired method of implementation.

The authors of this work recognized many of the limitations of their simplifications. They proposed additional study of system sensitivity to wind gusts and measurement noise but the results of any such studies, if they were performed, have not been found in the literature. This work obviously contained many invalid assumptions. However, it contains two major philosophical defects (as does most of the work in recent control theory). First, the problem solution involves a knowledge of the coefficient and driving matrices of the process [equations (2-40) and (2-41)]. This implies that the onboard computer knows all of the aircraft coefficients. (We often do not know them after extensive flight tests.) Moreover, it implies that these coefficients are linear. They usually are not in the landing problem. Second, and of even greater significance, is the implicit assumption that the measurement of the state variable and the action of the controller (δ_e) is accomplished with infinite bandwidth. This is almost a necessary assumption for practical time domain approaches to control system synthesis. For example, in the application of Merriam's method in reference 22, a set of 15 independent, first-order differential equations had to be solved continuously to determine the k parameters. The number of equations is determined by the order of the process under consideration. In reference 22, the simplification of the landing problem reduced it to a fourth-order system. When measurement and control dynamics and speed dynamics are considered, the landing problem reaches tenth order even without inclusion of the nonrigid dynamics in the aircraft representation. The number of simultaneous equations requiring solution in Merriam's method is $1 + N + (N/2)(N + 1)$, where N is the order of the system. Thus, a tenth-order system would involve the continuous solution of 66 simultaneous equations.

A final comment on the significance of ignoring the higher order dynamics of a problem can be made with regard to the conventional closed loop flight path control system discussed previously. The root locus of figure 2-16b can be moved into a region of excellent damping as the gains become infinite if we add an additional lead compensator on the $\dot{h}$ term. We need not go to very high gain levels before we appear to have a stable 1.0 Hz flight path control loop. Such a system would hold the aircraft on a reference path for any reasonable level of disturbance. It would rapidly and smoothly acquire the reference path when initial errors are imposed. By making no more assumptions than those made in reference 22, this linear system could out-perform any of the optimum systems investigated in that work. Unfortunately, bandwidth limitations (and velocity saturations) prevent the practical attainment of this level of performance. Also, elastic effect dynamics are encompassed by the wide bandwidth, high gain systems, and even if tight performance in terms of the rigid body representation of the problem can be attained, the actual result is usually an awesome spectacle of elastic mode instabilities.

2. <u>Terminal Controllers</u>

The terminal control concept appears attractive for the flareout application because in principle it offers a means of overcoming one of the difficulties of exponential flare controllers. That difficulty is the inability to maintain a tight constraint on touchdown dispersion in the presence of gusts, windshear, or other nonpredictable disturbances to the nominal problem. In principle, a terminal controller will always adjust the control law to meet the terminal requirements. The exponential flare controller also attempts to meet a terminal requirement but it does not adjust its control law to achieve this end. The exponential flare system attempts to meet only the $\dot{h}$ terminal constraint; "terminal" interpreted here as $h = 0$. The terminal controller can exercise some additional constraint on touchdown point because it includes time-to-go as one of its variables. To the extent that time-to-go will correspond to position, the touchdown position is included in the control law.

The theoretical basis of the terminal control concepts have been developed in references 23 and 24. The application of these concepts to the automatic landing problem was studied by Bendix (reference 25) and systems employing the terminal control principles have been built and flight evaluated by Autonetics (reference 26). Terminal controllers are actually very little different from other closed loop control systems when viewed in terms of the actual control law implementation. The difference lies primarily in the interpretation of the control laws. Terminal controllers (or final value systems) are synthesized from the viewpoint of a predictive process. If we ignore the effects of wind disturbances, the aircraft-autopilot combination is a deterministic system. In the conventional closed loop control system such functions as lead compensators could therefore be interpreted as predictive controls although the rationalization for their use will generally be related to a system phase characteristic or other property relating to system stability. In the design of a terminal controller the predictive aspect is paramount while stability factors are considered as a secondary problem.

A terminal control flareout system can be derived from simple geometric considerations. For example, assume that the aircraft is on the reference trajectory shown in figure 2-38. Forgetting the interpretation of that curve as a reference path, let us describe the trajectory (as a function of time-to-go) that the aircraft must follow to reach the ground at the terminal time and with a terminal constraint in $\dot{h}$. If we change the coordinates in figure 2-38 from time t to time-to-go where $\tau = (T - t)$ or $(20 - t)$ for figure 2-38, then h as a function of τ can be described by the power series

$$h = a_0 + a_1\tau + a_2\tau^2 + a_3\tau^3 + \ldots \tag{2-46}$$

$$\frac{dh}{d\tau} = a_1 + 2a_2\tau + 3a_3\tau^2 + \ldots \tag{2-47}$$

$$\frac{d^2h}{d\tau^2} = 2a_2 + 6a_3 + \ldots \tag{2-48}$$

$$\dot{h} = \frac{dh}{d\tau} \text{ and } \ddot{h} = \frac{d^2h}{d\tau^2} \tag{2-49}$$

The coefficients of the power series expansion in h [equation (2-46)] can be determined by inserting the terminal constraints at $\tau = 0$. Thus

$$a_0 = h_T \text{ (altitude at } t = T \text{ or } \tau = 0) \tag{2-50}$$

$$a_1 = \dot{h}_T \text{ (altitude rate at } t = T) \tag{2-51}$$

By substituting these values of a_0 and a_1 into equations (2-46) and (2-47), the coefficients for a_2 and a_3 are determined. We then have an expression for $\ddot{h}$ that can be interpreted as the instantaneous value of vertical acceleration required to follow the trajectory that will satisfy the terminal constraints. (Note that two terminal constraints exist; they are time-to-go, τ, which imposes an approximate position constraint, and vertical speed, $\dot{h}$.) Substituting the values of a_2 and a_3 into the equation for h, we get an effective acceleration control law given by

$$\ddot{h}_{command} = \frac{6(h_T - h - \tau\dot{h})}{\tau^2} - \frac{2(\dot{h}_T - \dot{h})}{\tau} \tag{2-52}$$

It is noted that if the power series expansion for h were to include higher order terms than τ^3, the $\ddot{h}$ command would involve higher derivatives of h in the control law. For example, if a τ^4 term were included in the power series, an $\dddot{h}$ term and an $\ddot{h}$ terminal constraint would appear in the control law. Experience with terminal controllers (references 25 and 26) indicates that the higher order terms are not desirable from a complexity and noise point of view. Also, they do not appear to make any significant contribution to system accuracy. Hence the control law form given in equation (2-52) is the type that has received the most consideration.

Two important observations are made concerning equation (2-52). First, the feedback quantities are h and $\dot{h}$ as in the exponential flareout systems. The relative weighting of the h and $\dot{h}$, however is variable. Second, the gains approach infinity as time-to-go approaches zero. These gains must be programmed as a function of time-to-go. This implies the computation of time-to-go. Such a computation can be made if we had precision DME to tell us the distance to go and a precision measurement of aircraft ground velocity. An alternate is to program the gains in an open loop manner. This timing function could be chosen on the basis of a nominal velocity and nominal flareout trajectory. Other compromises have to be made to achieve a practical mechanization. The gains must be restricted to realistic values. The first part of the control law should be eliminated at the final phase of the flareout so that a straightforward $\dot{h}$ control (with a touchdown bias) remains. The terminal control system, therefore, in its practical implementation looks even more like certain modified exponential flare law controllers. The study and flight results have not indicated any clear improvements over exponential flare law systems. It can therefore be concluded that terminal controllers represent a reasonable approach to practical flareout systems but whether or not they are the most desirable approach will depend upon the specific aircraft flight characteristics and its related avionics equipment.

E. SUMMARY OF CONCLUSIONS

1. Low visibility automatically controlled approach and landings are technically feasible and are being demonstrated continuously in modern jet aircraft. The main impediments to the full operational deployment of automatic landing systems involve questions of assuring equipment reliability and defining operating procedures for normal and emergency situations.

2. Approach guidance provided by the present ILS system is satisfactory for most automatic landing requirements. However, there are cases where the lateral guidance provided by localizer beams is marginal at best. The marginal accuracy is related to the accumulation of tolerance errors in the localizer alignment, receiver linearity and spurious radiation phenomena that produce beam bending effects. These tolerance factors combined with aircraft flight path control

errors in the presence of windshears and other disturbances can produce marginal lateral positioning accuracy for automatic touchdowns.

3. Guidance schemes that can be considered as alternates to the present ILS should provide the ILS equivalent of a fixed "highway in the sky". Concepts that allow any approach flight path provided that terminal conditions are met must be considered objectionable because there is no way to monitor performance of such systems with raw position measurement data. In general, a ground-based Precision Approach Radar (PAR) should have the capability of monitoring an automatic approach by comparing the measured aircraft position with an allowable window.

4. Performance deficiencies in 1967 state-of-the-art automatic approach and landing systems are not, in general, caused by inadequacies in the control concepts being used. They result from inadequacies in the primary guidance information available to define the desired flight path. Adapting systems to operate with deficient information results in performance compromise.

5. A major source of these compromises relates to the lack of an integrated DME or distance-to-touchdown information. The need for this type of data is especially important when the guidance information is derived from angular or converging beam flight path references such as those provided by the standard ILS localizers and glide slopes. To overcome the lack of distance information, present day systems make approximate determinations of this parameter by means of timing programs, crude altitude measurements, or discrete position approximations provided by marker beacons.

6. Another source of performance compromise relates to the lack of precise cross course velocity information for lateral control. Methods of synthesizing the necessary cross course velocity data from radio and inertial references lead to minor difficulties that appear as tendencies to hold small position offsets or as excessively active control wheel commands.

7. The lateral velocity problem and the lack of usable DME combine to make lateral performance dependent upon the initial localizer beam intercept angle and distance from the runway when the localizer flight path reference is captured. Significant variations in localizer capture performance can result from small changes in the initial conditions of beam intercept.

8. The lateral alignment or rollout alignment maneuver immediately prior to touchdown looms as a more difficult problem for the jumbo jets and SST. Present day jets can touchdown without prior removal of fairly large crab angles that result from the approach flight in the presence of crosswinds. An automatic decrab yawing maneuver or a sideslip approach will probably be needed to land the SST in a crosswind. An automatic decrab system imposes some additional

accuracy requirements on the vertical guidance references and some tighter constraints on the vertical flareout path. A sideslipping approach technique can eliminate the crab angle problem but may compromise runway alignment accuracy.

9. Vertical path control performance on automatically guided approaches to a flareout window are dependent upon procedures that are often beyond the control of the automatic system. Thrust setting, flap extension, and landing gear deployment influence flight path control accuracy. Automatic throttle controls have been introduced to remove non-uniform speed variations as one of these uncertainties. An ideal automatic system would also automate flap and gear deployment but system complexity penalties seem to dictate against considering such techniques.

10. For vertical flight path control as well as lateral control, the precision of the aircraft's position and velocity alignment at the flareout altitude determines the success of the landing maneuver. The closed loop flareout control laws have a limited ability to make corrections from off-nominal alignments prior to flare initiation.

11. Under proper conditions of alignment, excellent flareout performance can easily be obtained for even severe conditions of windshear disturbances if the criterion of success is touchdown velocity. If a tight constraint in touchdown position is also applied, then two undesirable penalties result. First, the nominal touchdown velocity will increase and there will be a statistical shift toward hard landings. Second, flareout initiation must be delayed to occur at lower altitudes. Pilots consider automatic flareout initiation at low altitudes objectionable.

12. The state-of-the-art radio altimeter is an adequate source of flareout guidance but its utility is often limited by the terrain profile along the approach to the runway. If the radio altitude signal characteristics are such that wide bandwidth vertical velocity cannot be derived without excessive noise penalties, then inertial measurements can be used to augment the radio altimeter derived rate data. However, this method of synthesizing wide bandwidth vertical speed can cause difficulties if the approach terrain is irregular.

13. Category II low visibility operations which define a decision altitude of 100 feet when runway visibility range is 396.24 meters (1200 feet) can be considered to be the state of the art in 1967. In order to achieve Category IIIa operations, automatic guidance to touchdown is an essential requirement. The major problems which must be solved before Category IIIa can become operational relate to system failure characteristics and failure procedures. Automatic systems employ redundancy and sophisticated monitoring equipment to achieve the fail-operational capability that would be compatible with the safety requirements.

14. The methods of using the crew to monitor performance of the automatic equipment and a definition of crew procedures for various failure situations are critical problems which remain to be worked out for Category IIIa. The solutions are intimately related to the types of displays that are used to present flight situation and equipment status information.

15. All weather landing flight control displays used on present day jet transports are evolutionary improvements over the instruments that have been used for less critical flight control tasks. Questions exist regarding the compatibility of these heads-down instruments with such tasks as manual takeovers from the automatics immediately prior to touchdown or immediately after touchdown.

16. Heads-up displays that project a pictorial presentation of the flight control situation on the windscreen are usually suggested as the solution to the manual-automatic monitoring and takeover problem at Category IIIa altitude. Such displays have been developed but they have made little progress in being applied to operational aircraft. In addition to some technical problems and some controversy regarding symbology and limitations in viewing area, the heads-up displays have been impeded primarily by cost and space restrictions in existing cockpits.

17. In aircraft such as the SST a fundamental question regarding the desirability of a heads-up display for landing relates to the limitations of the windscreen itself as a means of obtaining an adequate real world view. It has been suggested that the SST can be landed and taxied by obtaining visual cues from remote television presentations of the landing gear and runway.

18. Very little progress has been made on automatic guidance techniques for Category IIIb, c conditions when the runway visibility range approaches zero. The incidence of true zero-zero conditions may be sufficiently low that the economic investment required for additional facilities to guide the aircraft along the runway cannot be justified.

19. A fundamental dilemma arises in regard to the increasing use of automatic guidance and control to land aircraft and the use of the pilot as a systems manager. We are still dependent upon the pilot to assume control under abnormal and adverse conditions but we may no longer afford him the opportunity to acquire the manual control proficiency that comes with flight experience. It appears that new concepts in pilot training will be needed for the era when automatic equipment will be allowed to exercise a dominant role in aircraft flight control.

REFERENCES

1. Action by the ICAO Seventh Com. Division Relevant to All Weather Landing and Take Off, Working Paper 4, 15th Technical Conference, Lucerne: All-Weather Landing and Take Off, Volume I, April 1963.

2. Hostek, F. H.; Griffin, C. E.; and Piwarzyk, R. W.: "FAA Certification of All Weather Landing Equipment Installed in the DC-8 Aircraft", Volume I, Douglas Aircraft Division Report No. DEV-3702, Revision A, June 27, 1966.

3. "Dynamics of the Airframe", Bureau of Aeronautics Report AE 61-4-II, September 1952.

4. Popik, M. J.; and Gaylor, R.: "Development of an Automatic Throttle System for Automatic Landing Application", FAA International Aviation Research and Development Symposium, Atlantic City, New Jersey, September 1963.

5. Lang, G.; and Ham, J. M.: "Conditional Feedback Systems - A New Approach to Feedback Control", AIEE Transactions Volume 74, Part II, July 1964.

6. Foss, R. L.; Magruder, W. M.; Litrownik, I.; and Wyrick, D. R.: "Low Speed Aerodynamic Characteristics of the Double Delta Supersonic Transport", Journal of Aircraft Volume 2, No. 6, November-December 1965.

7. Schaeperkoetter, R. L.: "Automatic Flareout and Landing Study for the Boeing 727", Sperry Phoenix Company Report TM LJ-1261-0255, September 1963.

8. Woodward, J. E.: "Status Report on Flarescan", IATA 15th Technical Conference, Lucerne, Working Paper 44, April 1963.

9. Woodward, J. E.; and Litchford, G. B.: "A Progress Report on the Advanced Integrated Landing System (AILS)", AIAA/ION Guidance and Control Conference, University of Minnesota, August 1965.

10. Litchford, G. B.: "Some Interrelationships Between ILS Reference Points, Flare Guidance and the Need for Runway Extensions for All Weather Landing", IATA 15th Technical Conference, Lucerne, Working Paper 133, April 1963.

11. Kramer, K. C.: "An All Weather Landing System for the Caravelle", IATA 15th Technical Conference, Lucerne, Working Paper 105, April 1963.

12. "Sperry All Weather Landing, A Progress Report", IATA 15th Technical Conference, Lucerne, Working Paper 132, April 1963.

40. Woodward, J. E.: "Guidance System for Approach and Landing", FAA Second International Aviation Research and Development Symposium on All-Weather Landing Systems, September 1963.

41. Jacobson, R. S.; and Stotz, C. C.: "Beacon Vision", Sperry Engineering Review, Volume 17, No. 3, Fall 1964, pages 40-46.

42. Bull, G.: "Longitudinal Dynamics, A Critical Piloting Problem in Landing the Supersonic Transport", Society of Experimental Test Pilots (SETP) Sixth Annual Symposium, September 28, 1962.

43. "Design Objectives for Flying Qualities of Civil Transport Aircraft", SAE Aerospace Recommended Practice, ARO 842, August 1, 1964.

44. Tobie, H. N.: "A New Longitudinal Handling Qualities Criterion", Advancements in Adaptive Flight Control, 1966 NAECON.

45. Pinsker, W. J. G.: "Features of Large Transport Aircraft Affecting Control During Approach and Landing", IATA 15th Technical Conference, Lucerne, April 1963.

46. Stapleford, R. L.; Johnston, D. E.; Teper, G. L.; and Weir, D. H.: "Development of Satisfactory Lateral-Directional Heading Qualities in the Landing Approach", NASA CR-239, July 1965.

47. Drake, D. E.: "Direct Lift Control During Landing Approaches", AIAA Paper 65-316, July 1965.

48. Smith, L. R.; and Prilliman, F. W.: "Direct Lift Control as a Landing Approach Aid", AIAA Paper No. 66-14, January 1966.

49. AM&M Interview with Colonel Joseph F. Cotton, USAF - The Subject: The World World of Mach 3; The Experts: Pilots of the XB-70, March 1966.

50. Ashkenas, I. L.; and McRuer, D. T.: "A Theory of Handling Qualities Derived from Pilot-Vehicle System Considerations", Aerospace Engineering, February 1962.

51. Notess, C. B.; and Gregory, P. C.: "Requirements for the Flight Control System of a Supersonic Transport", SAE 679C, April 1963.

27. Wulfeck, J. W.; Wlisz, A.; Rabin, M. W.; and Emerson, G. O.; "Vision in Military Aircraft", USAF WADC Technical Report TR-58-399, 1958.

28. Fogel, L. J.; Biotechnology: Concepts and Applications, Prentice Hall Inc, 1963.

29. Electronics Laboratory General Electric Company, Syracuse, N. Y.: "Final Report, Digital Contact Analog", Contract No. NOnr 4057 (00), June 1966.

30. Baxter, J. R.; and Workman, J. D.: "Review of Projected Displays of Flight Information and Recommendations for Further Development", IATA 15th Technical Conference, Lucerne, Working Paper 13, April 1963.

31. Kear, Frank G.: "Instrument Landing at the National Bureau of Standards", IRE Transactions on Aeronautical and Navigational Electronics, Volume ANE-6, June 1959, pages 61-66.

32. Cutrell, E. A.: "Operational Flight Testing of Early Instrument Landing Systems", IRE Transactions on Aeronautical and Navigational Electronics, Volume ANE-6, June 1959, pages 67-70.

33. Metz, Henry I.: "A Survey of Instrument Approach Systems in the United States", IRE Transaction on Aeronautical and Navigational Electronics, Volume ANE-6, June 1959, pages 85-94.

34. Jackson, W. E.: "Improvements on the Instrument Handling System," IRE Transaction on Aeronautical and Navigational Electronics, Volume ANE-6, June 1959, pages 85-94.

35. Sandretto, Peter C.: Electronic Aviation Engineering, International Telephone and Telegraph Corporation, New York, 1958.

36. Gregory, E. C.: "High Density Landing Systems Study", Cook Research Laboratories Final Report, Contract No. DA-36-039-sc-84513, March 1960.

37. Iden, Floyd W.: "Glide-Slope Antenna Arrays for Use Under Adverse Siting Conditions", IRE Transactions on Aeronautical and Navigational Electronics, Volume ANE-6, June 1959, pages 100-111.

38. Friedman. A.; and Doniger, J. (Bendix): "Analog Computer Study of Safe Low Approach Decision Regions and Effectiveness of Glide Path Extension Techniques", Phase II Final Report - FAA Contract No. ARDS 451.

39. Wulfsberg, Paul G.: "Solid State ILS Receiver Developments for Low Approach", Second International Aviation R&D Symposium, All-Weather Landing Systems, September 1963.

13. KLM Research Department: "Studies Related to Flareout Guidance", IATA 15th Technical Conference, Lucerne, Working Paper 68, April 1963.

14. "Criteria for Approval of Category II Landing Weather Minima", FAA Advisory Circular AC120-20, June 6, 1966.

15. Cutrell, E. A. (R. Dixon Speas Association): "Airport Statistics Indicating the Frequency of Low Limits and Zero-Zero Conditions and Local Fog Characteristics", IATA 15th Technical Conference, Lucerne, April 1963.

16. "Automatic Pilot Systems Approval", FAA Advisory Circular AC25.1329-1, October 1, 1965.

17. Rowe, N. E.: "Flight Safety in the New Jet Era", Astronautics and Aeronautics, September 1966.

18. Fearnside, K.; and Cane, A. P. W.: "Monitoring Information and Failure Warning to the Flight Crew During Autopilot Operation", Royal Aeronautical Society Centenary Congress and 5th Congress International Council of Aeronautical Sciences, September 1966.

19. Perry, D. H.; Port, W. C. A.; and Morrel, J. C.; "A Flight Study of the Sidestep Manoeuvre During Landing", R. A. E. Report Aero 2654, July 1961.

20. Beck, R. H.: "1200 RVR, Clear to Land", The Air Line Pilot, August 1966 (Air Line Pilots Association publication).

21. Tou, J. T.: Modern Control Theory, McGraw-Hill Book Company, 1964.

22. Ellert, F. J.; and Merriam III, C.W.: "Synthesis of Feedback Controls Using Optimization Theory - An Example", IEEE Transactions on Automatic Control, Volume AC-8, No. 2, April 1963.

23. Matthews, M. V.; and Steeg, C. W.: "Terminal Controller Synthesis", Princeton Symposium on Non-Linear Control Systems, March 1956.

24. Boonton, R. C. Jr.: "Optimum Design of Final-Value Control Systems", Proceedings Polytechnic Institute of Brooklyn, Symposium on Non-Linear Control Systems, 1956.

25. Doniger, J.; and others (Bendix Corporation-Eclipse Pioneer Division): "Automatic Landing System Study", ASD-TR-61-114, Part J, February 1962.

26. Velander, E. W.: "Autoflare Manual and Automatic Landing", IATA 15th Technical Conference, Lucerne, April 1963.