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DESIGN EVALUATION CRITERIA FOR COMMERCIAL STOL TRANSPORTS

By R. L. Allison, M. Mack, P. C. Rumsey

D6-40409

JUN 1972

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Prepared under contract NAS2-6344 by

THE BOEING COMPANY
Commercial Airplane Group
P. O. Box 3707
Seattle, Washington 98124

for
Ames Research Center
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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SUMMARY

Handling qualities criteria and operational performance margins have been determined for the landing phase of commercial short-takeoff-and-landing airplanes. The requirements are the result of a literature survey, analysis of areas found to be inadequately covered by current criteria, and a subsequent piloted simulator investigation of critical criteria requiring substantiation. Three complete simulator models were used, each describing the characteristics of a different high-lift system, the externally blown flap, the augmentor flap, and the internally blown flap.

The proposed criteria are presented with substantiating discussions from currently available data or directly from the results of this simulation work where it is applicable. Further work is required in some areas where time limitations prevented full investigation of all three concepts or complete analysis of a given criteria topic.

The requirements are offered here as a starting point, and a base for the evaluation of competitive STOL designs.

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Detailed breakdown of each section will be found on the pages listed.

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1.0 INTRODUCTION

The work reported in this document was performed under Contract NAS2-6344 to NASA-Ames Research Center. The purpose was to conduct research into the flying qualities of some typical commercial STOL transport designs in order to contribute towards a statement of criteria for the performance margins and handling qualities required for STOL operations. These criteria could then be used in the design definition and evaluation of competing STOL high lift concepts.

The research was restricted to the landing flight phase including approach, landing flare, roll-out, and go-around. The investigation concentrated mainly on the longitudinal performance and handling qualities since the lateral/directional requirements were already well covered by existing literature. A small amount of engine failure work was accomplished.

A literature survey was conducted to establish current regulations and criteria and to define the areas requiring further investigation. Direct comparisons of existing requirements from various sources helped in the definition and planning of the piloted simulation work that is reported in this document.

Three proposed STOL transport configurations were modeled in the simulator: the externally blown flap (EBF), the internally blown flap (IBF), and the augmentor wing (AW). A common approach speed was chosen at 80 knots which was considered to be consistent with the 2000 foot field length for the gross weight range of these airplanes. The augmentor wing airplane was designed to pass 80% of the total engine air flow to the trailing edge flaps. Space requirements for the ducting in the wing necessitated a low wing loading by CTOL standards for this configuration. The internally blown flap configuration used 12 to 15% of the engine air to blow the leading edge and trailing edge flaps, and the externally blown flap used an 8% bleed to the leading edge only.

The engine positioning on the EBF wing was carefully chosen to give favorable thrust impingement on the flaps, whereas on the IBF the engines were placed to avoid jet exhaust interference with the trailing edge flap system.

Seperate longitudinal Stability Augmentation Systems (SAS) were designed for each configuration. The SAS feedbacks, feedforwards, and control interconnects were used to vary the airplane handling characteristics so as to define acceptable and unacceptable values of postulated handling qualities parameters. Different SAS types were used to evaluate different piloting techniques for control of flight path and speed. Two main SAS types emerged from these evaluations and their form is described in detail in Section 5 of this document. In other sections, the two SAS types are referred to as:

(a) Minimum SAS - describing an augmentation system that has pitch rate damping and attitude hold features tied to the elevator, and flight path rate fed back to the engines to improve the control of flight path angle with throttles.

(b) Full SAS - describing an augmentation system using similar feedbacks to (a) but including also the use of auxiliary flap or main flap modulation controlled by stick and throttle movements, and having speed, angle-of-attack, and flight path acceleration feedback signals. This SAS also includes an interconnect from the column into engine thrust.

Neither of these augmentation systems included typical amplitude and rate limits, except those imposed by full control deflection, maximum engine thrust or acceleration capability. The SAS was used simply to modify airplane handling qualities for the purposes of this research.

The aerodynamic data used in modeling the three concepts had different sources which influenced the flying qualities of each type. The Augmentor Wing was modeled on wind tunnel data (from the Ames 40 x 80 tunnel) corrected to the chosen wing planform and flap configuration. The EBF data was derived from theoretical techniques modified empirically by data from NASA and Boeing wind tunnel tests. The IBF aerodynamic build-up was almost purely theoretical using jet-flap theory with some small corrections from British RAE data. The resulting Augmentor Wing data retained significant non-linearities associated with the wind tunnel model from which it was derived, whereas the EBF data was averaged from a number of tests with the non-linearities faired out. Differences in handling qualities between the airplane types are therefore mostly due to the different levels of accuracy of the aerodynamic data modeling. The results of the tests are not presented as a judgement of one concept against the other, but rather as an indication of the effects that non-linear data can have on total airplane handling qualities.

The three configurations covered a range of possible aerodynamic characteristics against which the proposed criteria could be judged. Further work is required to fully prove these criteria, and statistically designed experiments are needed to justify the rational approach to landing field length determination.

The criteria are specifically identified in each section of the document, and background material for each is presented in the discussion sections which appear after each stated criteria. This layout gives freedom to include as much of the simulation results as possible in this document. The proposed criteria are offered as a starting point for further research, and as a base for the evaluation of competitive STOL designs.

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1.1 List of Symbols

C_J	= $\frac{\text{Blowing Thrust (lbs)}}{\text{Dynamic pressure (lbs/ft}^2\text{) x Reference wing area (ft}^2\text{)}}$, blowing momentum coefficient
C_L	Coefficient of lift
$C_{L_{MAX_{C_J}}}$	Maximum lift coefficient available at the value of C_J at which the airplane is flying.
$C_{L_{MAX_{C_{J_{MAX}}}}}$	Maximum lift coefficient available at the value of C_J computed with maximum thrust.
$C_{L_{WB}}$	Lift coefficient of wing and body alone
$C_{L_{\delta e}}$	Lift coefficient due to elevator deflection
$C_{M_{\delta e}}$	Pitching moment coefficient due to elevator deflection
F_{G_B}	Blowing thrust value, lbs.
F_S	Pilot's stick force, lbs.
h	Airplane altitude, feet
$L_{u,v,w}$	Characteristic length of turbulence, feet.
L_δ	Rolling moment due to control deflection, ft-lbs/deg
N_Z, n_Z	Normal acceleration, ft/sec^2
P_{osc}/P_{AV}	Ratio of the oscillatory roll rate response to the average roll rate response to a step wheel input
Q	Pitch rate, deg/sec
t	time, secs
$t_{\dot{h}}$	The time required to achieve a positive change in rate of climb following an aft column step input, secs.
t_ϕ	The time taken to reach a bank angle ϕ in a maximum control roll maneuver, secs.
T_D	Dutch roll period, secs.
T/W	Thrust to weight ratio

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T_{ψ}	The time required to achieve a positive change in heading angle following a right wing down roll input, secs
V_e	Equivalent air speed, knots
V_{10}	Crosswind velocity at 10 feet altitude, knots
V_{WIND}	Steady wind speed, knots
V_{MIN}	Minimum demonstrated speed in stalling maneuver, knots
V_{MCA}	Minimum speed at which straight flight can be maintained with one engine failed and less than 5° of bank angle, kts
W/S	Wing loading, lbs/ft^2.
W_o	Initial rate of descent, ft/sec
Z_W	Normal force due to vertical velocity, heave damping, lbs/ft/sec.
α	Angle of attack, degs.
α_W, α_{WING}	Wing angle of attack, degs.
α_B	Angle of attack referred to body axes, degs
β	Sideslip angle, degs.
δ_{AUX}	Auxiliary flap deflection, degs
δ_{COL}	Column deflection, ins
δ_P	Pedal deflections, ins.
δ_{THL}	Throttle deflection, ins.
δ_W	Wheel deflection, degs or inches
γ	Flight pathangle, degs.
η_T	Effective inclination of thrust vector including interference effects with wing aerodynamics, degs.
$\sigma_{u, v, w, ref}$	r.m.s. level of random turbulence, ft/sec
μ	Coefficient of braking friction on runway
ζ_d	Dutch roll damping ratio

τ_{N_Z}	Effective single degree of freedom time constant of load factor response, secs
τ_R	Effective single degree of freedom time constant of roll response, secs
ω_{n_D}	Frequency of dutch roll oscillation, rads/sec
ω_{SP}	Frequency of short period oscillation, rads/sec
θ,	Pitchangle, degs.
ϕ	Roll angle, degs
ϕ_{1}/δ	Roll angle achieved in one second due to a unit deflection of roll control, degs/inch
ϕ_{osc}/ϕ_{AV}	Ratio of the oscillatory bank angle response to the average bank angle response to a pulse wheel input
ψ	Heading angle, degs

Common subscripts: -

SS	-	steady state
Max	-	maximum value
app	-	at approach trim conditions
$(\dot{\ })$	-	$\frac{\partial}{\partial t}(\ )$
$(\ddot{\ })$	-	$\frac{\partial^2}{\partial t^2}(\ )$

Page

2.0 OPERATIONAL ENVIRONMENT

This section defines the assumed operational environment used in the design of the three STOL airplanes and in the planning of the test conditions for the piloted simulation work. This study was directed towards the landing and go-around phases of the flight profile and only these conditions are dealt with.

At this stage in the development of commercial STOL transportation systems very little is known of the likely airport sizes or sites which may be available. The choice of turbulence levels, clearance planes, runway sizes, etc. is therefore an arbitrary one, although necessary for the assessment of handling qualities and operational suitability of proposed STOL designs. The following definitions are therefore a first cut which the three STOL airplanes were designed to meet without excessively large penalties.

2.1 Wind and Turbulence

A modified version of the wind model defined in Reference 1 was used. This assured consistency with previous STOL handling qualities investigations performed at NASA Ames Research Center.

2.1.1 Steady Winds

The tendency in airport planning towards single direction operation, and the further restrictions which will occur with STOL strips situated in narrow access corridors, is causing airlines to demand higher and higher crosswind capability from short-field airplanes. Ransome has suggested (in Reference 2) that the design cross-wind be at least 25 knots and perhaps as high as 35 knots. Other airline feedback to the Boeing Company has suggested that a 30 knot crosswind measured at a height of 10 feet is an acceptable minimum for future airplane design. As discussed in Section 7.1.2, a 30 knot cross-wind was chosen for this study since the design penalty for the required rudder and lateral control power seemed reasonable at approach speeds above

70 knots.

The wind shear profile used is shown in Figure 2.1-1, and was recommended in Reference 1. For the design crosswind level, V_{10} = 30 knots.

2.1.2 Random Turbulence

The Reference 3 turbulence model was used with the Dryden spectral form. The r.m.s. turbulence levels in two axes, σ_u and σ_v, are kept constant at the reference level, $\sigma_{Ref.}$. The variation of σ_w with altitude is shown on Figure 2.1-2, and of L_u, L_v and L_w on Figure 2.1-3. The maximum turbulence level for normal handling qualities was set at the .01 probability level for clear air turbulence (non-storm) as defined in Reference 3.

2.2 Weather Minima

The majority of the piloted simulator investigation was performed in VFR flight conditions. Critical flight path control tasks were repeated under simulated IFR conditions with breakout at the Category II minima. No investigation has been made of Category III requirements.

2.3 Terrain Clearance Plane

The FAA has defined minimum flight path requirements and protection surfaces for STOLports in Reference 4. Ransome (Reference 2) has requested at least a 6° climb-out capability with all engines operating, and a 1% gradient above the FAA protection surface for the engine failed condition. This latter case would appear to be a reasonable minimum, and also consistent with the need for minimizing noise on the ground and providing a flexible performing airplane that can fit into and around existing ATC routes.

2.4 Allowable Noise Footprint

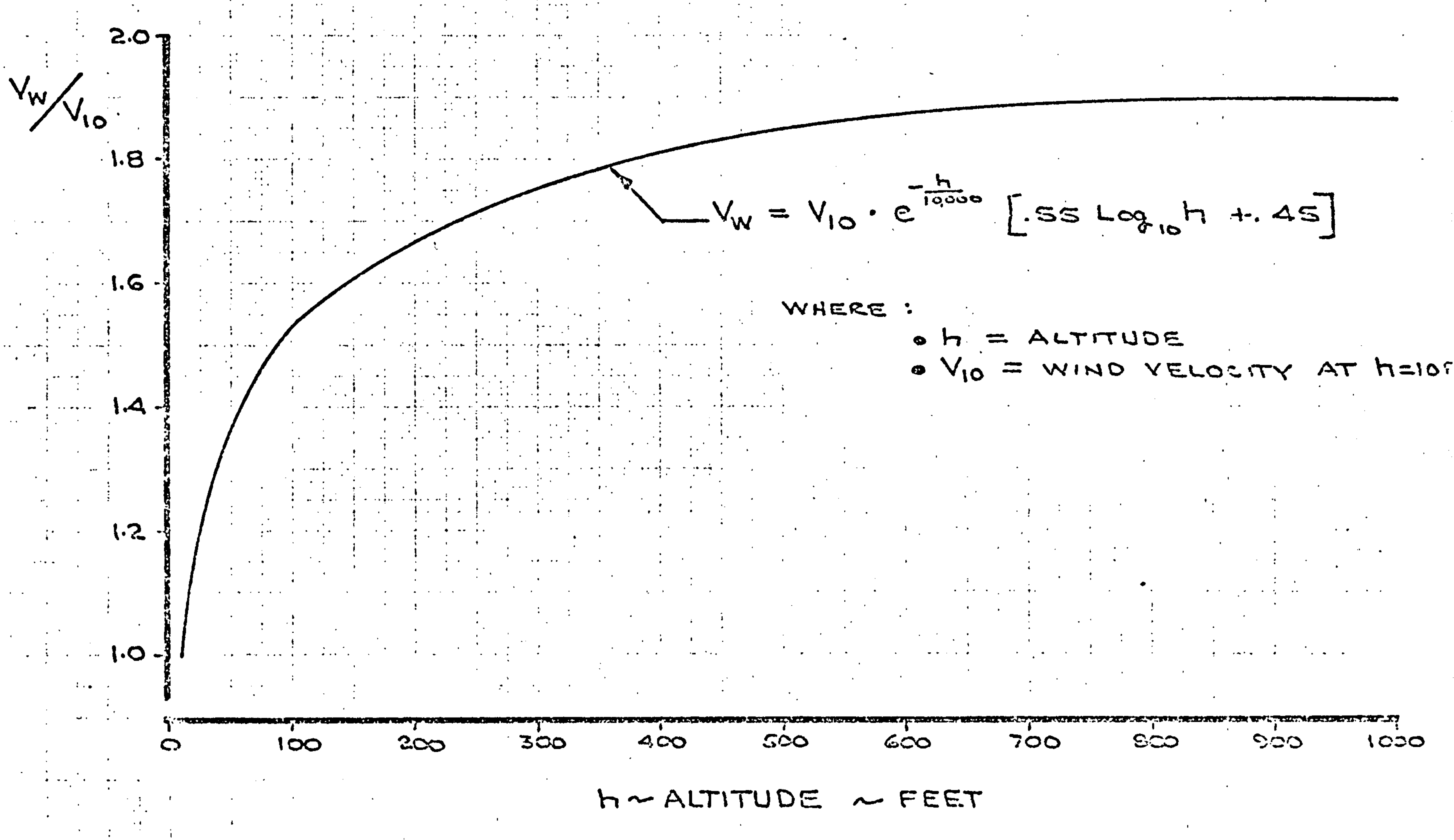
The aim of introducing jet-powered STOL aircraft into short fields built inside existing communities is going to require considerable reduction in the noise levels emitted by these airplanes. A suitable goal would seem to be to produce a noise level below the residual level of the surrounding community. A first definition of this would be to confine the 80 EPNdB contour within a box 20,000 feet long by 4,000 feet wide centered on the runway.

2.5 Runway

Runway width and markings are laid out in Reference 4. The glide slope location relative to the threshold and the touchdown aiming zone have an impact on touchdown dispersions as discussed in Section 3.5. It should be expected that STOL runways will be grooved, and possibly heated, to maintain a reasonable minimum friction coefficient, $\mu = .2$.

2.6 Aircrew

Cockpit layout and systems design should be consistent with the concept of a two man crew. It should be possible for either crew member to fly and land the aircraft in an emergency. The piloting task should not require exceptional skills or extensive special training.



LOW ALTITUDE WIND SHEAR PROFILE FOR STEADY WIND MODEL

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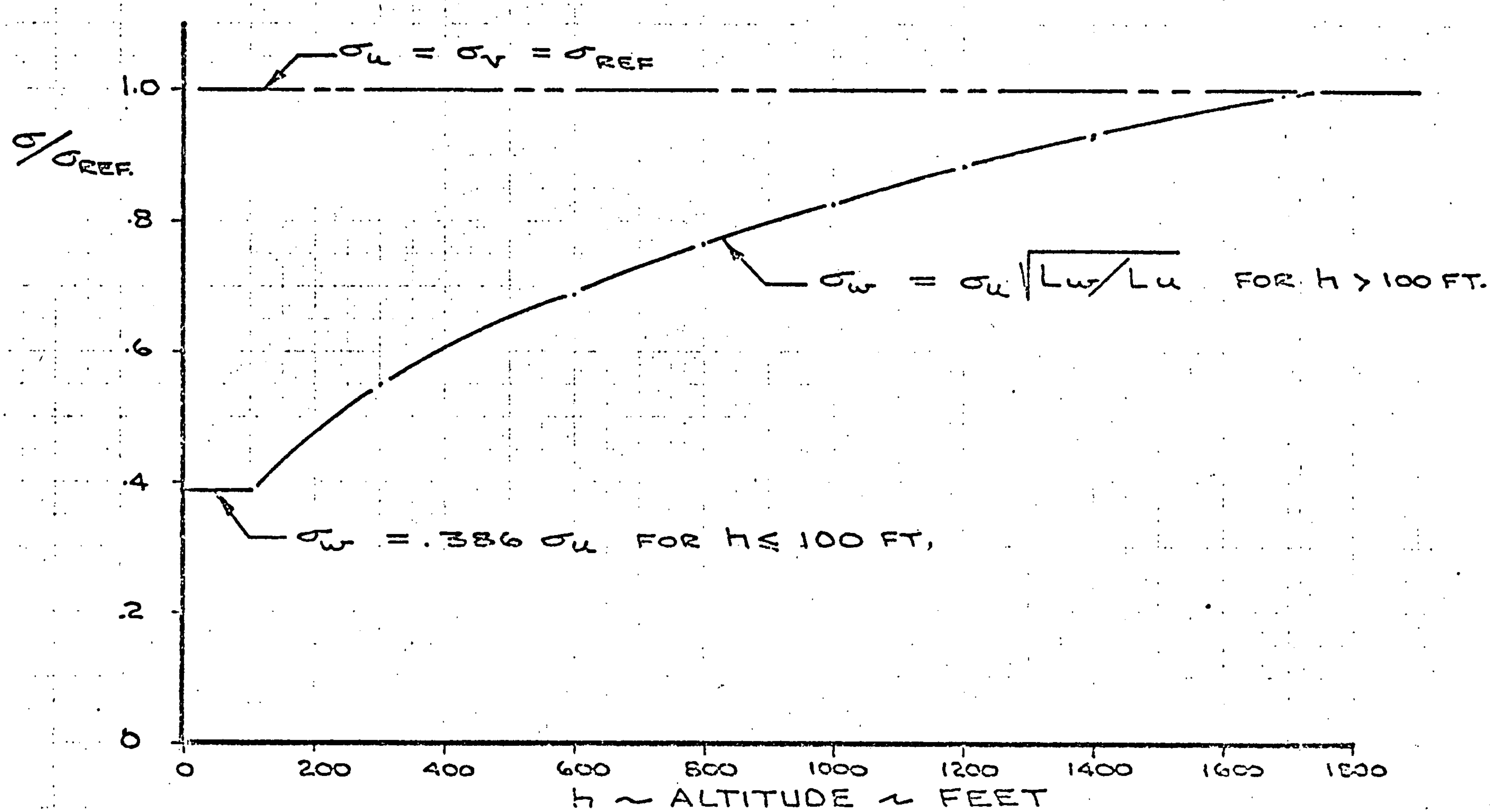
FIG 2.1-1



NOTE: $\sigma_{REF.}$ = REFERENCE RMS VELOCITY

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RMS VELOCITIES FOR RANDOM TURBULENCE MODEL

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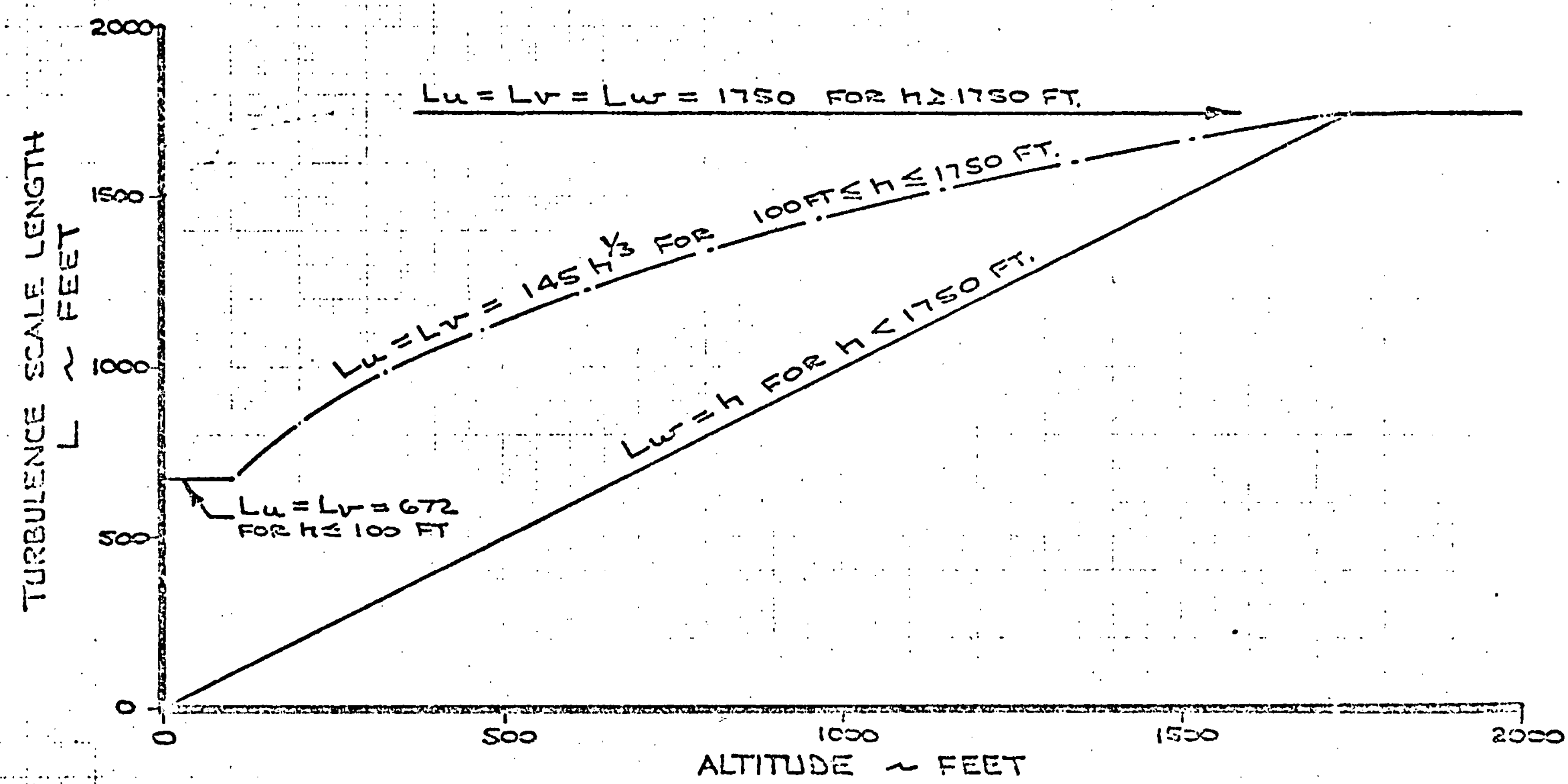
FIG. 2.1-2



SCALE LENGTHS FOR RANDOM TURBULENCE MODEL

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FIG. 2.1-3

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3.0 OPERATIONAL MARGINS

3.1 General

Criteria: The choice of operational margins must provide protection from uncontrollable flight conditions yet allow sufficient maneuvering for completion of approach positioning, landing, or the go-around procedure. The requirements may be classified into four categories:

1. Provide adequate maneuver capability to perform the functions that are required of the airplane.
2. Provide a margin in angle-of-attack to prevent dangerous loss of lift or loss of control due to atmospheric disturbances.
3. Provide a margin in speed to prevent dangerous loss of lift or loss of control due to speed deviations from the reference speed.
4. Provide capability to maintain the desired flight path at speeds less than the target speed.

The choice of operational landing field lengths should reflect the consistency with which the flare and braking performance required can be demonstrated. Typical environmental conditions that may be encountered in routine commercial operation must be taken into account as well as the effects of likely powerplant and system failures.

Discussion: The operational margins used by today's airplanes have been derived and developed from many hours of commercial jet operations, accumulated over a number of years flying into and out of a variety of airports in extremes of atmospheric conditions. In terms of lift margins, today's requirements are simple and are quoted as speed margins from a stall speed measured under carefully defined maneuvering conditions. Climb gradients and field length requirements have grown from operational experiences and are known to allow operation from most of the world's airports without large off-loading penalties.

The related lift, thrust, and angle of attack characteristics of a typical powered lift STOL airplane renders this simple concept of speed margins inadequate. The basis for the proposals in this section is that the capability of current conventional aircraft should be matched in terms of maneuvering margin and margin from stall in gusts, but not necessarily in the speed margin. This latter exception is partially justified by noting that in-service flying records demonstrate that speeds below the reference value are rarely recorded. In fact, pilot's make every effort to stay fast by the addition of speed in all critical cases such as gusty weather, high winds or for heavy weight conditions. Typical records of landings show that touchdown speeds average only a 5 to 10% speed decrease from the speed on approach. The conclusion is that the currently required speed margin is more truly indicative of protection from elements other than inadvertent speed decreases. Such decreases must still be taken into account, however, but the existing margins could possibly be reduced or replaced by other equivalent protection.

In the same way, the landing roll-out and take-off operation for STOL is very different from conventional operation. When aircraft body lengths approach 10% of the total available runway a more rational approach has to be taken to generating field length margins than the existing application of an overall factor. Following the approach outlined in Reference 5 the present study ran a number of simulated landing tests, and the resulting touchdown and roll-out data has been analyzed on a statistical basis to help draw conclusions concerning the important parameters affecting consistency of touchdown performance.

These analyses are simply a beginning. It is obvious that much more data in the simulation area, and in actual flight operations, will be necessary before hardened criteria can be adopted concerning operational margins. But, the proposed criteria have been demonstrated to be sufficient by the testing conducted in the present study, and they are therefore offered for use as a baseline for evaluation of competitive STOL designs.

3.2 Lift Margins

Criteria: The choice of reference speed and configuration for the approach shall provide a margin of lift from stall warning to cater for the acceleration requirements of Sections 5.4.1.1 and 5.4.2.1, and the flight path control requirements of Section 5.4.1.2.

The choice of reference speed and configuration for the approach shall provide a sufficient margin of lift and angle-of-attack from wing stall or from loss of control about any axis to cater for:

1. A maneuver up to an incremental .5 g, with all engines operating at maximum power.
2. A maneuver up to an incremental .35 g, with all engines operating at the trim power for the landing approach.
3. A step gust of 20 knots T.A.S. normal to the flight path, with one engine inoperative and the remaining engines set at the power required to maintain the approach glide slope at the reference speed.

Discussion: - The airplanes simulated in the present study were designed to meet a series of conservative margins formulated from a comparison of the capabilities of conventional jet commercial transports. The single lift margin used by current airplanes acts as a margin for maneuver, speed errors, angle-of-attack protection in gusts, and for recovery from wind shear. The interaction between thrust setting and lift for the propulsive lift systems used by STOL airplanes requires that these individual margins be treated and applied separately with carefully defined power settings, speeds, and configurations. This approach assumes that the margins used in today's jet operations are critical in all phases and that the future of jet STOL operations will occur in similar environmental conditions. Neither of these assumptions is necessarily correct.

Section 2.0 of this criteria document defines the environment selected to represent a first proposal for the definition of STOLport weather conditions. It was an aim of the present study to define more clearly the margin requirements for STOL operation in this selected environment.

3.2.1 Maneuver Requirements

The normal maneuver and control requirements for the STOL landing and approach phase are presented in detail in Sections 5.4.1.1 and 5.4.2.1. Because these are normal operational maneuvering requirements they are expressed as margins from stall warning (inherent or artificial), not from $C_{L_{max}}$.

Other maneuvering requirements, more usually measured as margins from $C_{L_{max}}$, are for protection from gusts or for gross collision avoidance maneuvers. However, the interconnected lift and thrust characteristics that are a vital feature of the STOL concept allows the available lift margin to vary with power setting. These characteristics may lead to a new requirement for a margin to cover likely operational conditions where flight parameters may momentarily reduce otherwise specified margins to very low values. The equivalent case for CTOL is covered by the knowledge that the lift margin available is independent of engine thrust setting, and monitoring of speed alone will ensure sufficient maneuver capability in all flight phases.

Theoretically speaking, the 30% speed margin currently used for CTOL operations should yield an incremental acceleration capability of .69 g before stall at constant speed. However, due to the dynamic nature of the demonstration maneuvers used to define the reference speed, V_{MIN}, the actual normal acceleration margin to the break in the lift curve slope ranges from about .42 to .59 incremental g for a typical family of commercial jet airplanes. If it is assumed that these margins are critical for safe operation, and that the requirement is independent of approach speed, the STOL airplane should be designed to match this capability. To understand the validity of this assumption, the three high-lift STOL types used in the present study were designed to meet the most critical of a set of performance margins which equal the maneuver capability and gust penetration ability of today's jet transports. These margins are listed in the criteria comparison of Section 9.0. The lift and angle-of-attack margins which resulted (for level flight and for descent on the glide slope) are shown on Figure 3.2-1. Operational type evaluations were then made of these airplanes, although limited simulation time allowed a thorough examination of only two of these types, the augmentor wing and the externally blown flap aircraft.

The variation of lift margins during operational maneuvers in calm air and in turbulence was investigated by means of continuous recordings of the following parameters:

(i) Instantaneous lift coefficient, C_L, measured at the instantaneous α_w and power setting.

(ii) Maximum lift coefficient, $C_{L_{MAX_{C_J}}}$, available at the instantaneous power setting and speed.

(iii) Maximum lift coefficient, $C_{L_{MAX_{C_{J_{MAX}}}}}$, available at maximum power at the instantaneous speed.

All lift coefficients are for the tail-off airplane including ground effects. For easy comparison, the first two parameters were recorded on time shared traces, and the instantaneous lift coefficient, C_L, was also displayed in the ratios $C_L/C_{L_{MAX_{C_J}}}$ and $C_L/C_{L_{MAX_{C_{J_{MAX}}}}}$.

The variations of these parameters help to illustrate the advantages and disadvantages of the powered lift concepts from the point of view of lift margins, and a number of examples are given here as background material.

LIFT MARGIN SUMMARY - NASA AMES JET STOL AIRPLANE

TRIM SPEED 80 Kts	TRIM α w	TRIM Thrust %	TRIM $C_L = \frac{W}{q S_W}$	$\frac{C_{L_{MAXWB}}}{C_{L_{WBTRIM}}}$ at trim thrust	$\frac{C_{L_{MAXWB}}}{C_{L_{WBTRIM}}}$ at max. thrust	$\Delta\alpha$ at Trim Thrust degs.	$\Delta\alpha$ at Max. Thrust degs.
Augmentor Wing Fwd c.g.							
γ = 0	-2.50	74.0	3.58	1.81	1.98	35.2	37.7
γ = -6	5.0	45.5		1.64	2.05	32.8	30.2
Externally Blown Flap Aft c.g.							
γ = 0	7.7	65.9	4.64	1.63	1.82	23.9	23.9
γ = -6	12.7	39.7		1.40	1.86	17.3	18.9
Fwd c.g.							
γ = -6	13.4	43.3		1.39	1.78	17.4	18.2
Internally Blown Flap Fwd c.g.							
γ = 0	5.8	72.0	4.80	1.35	1.39	20.6	20.8
γ = -6	7.3	49.0		1.31	1.39	18.7	19.3

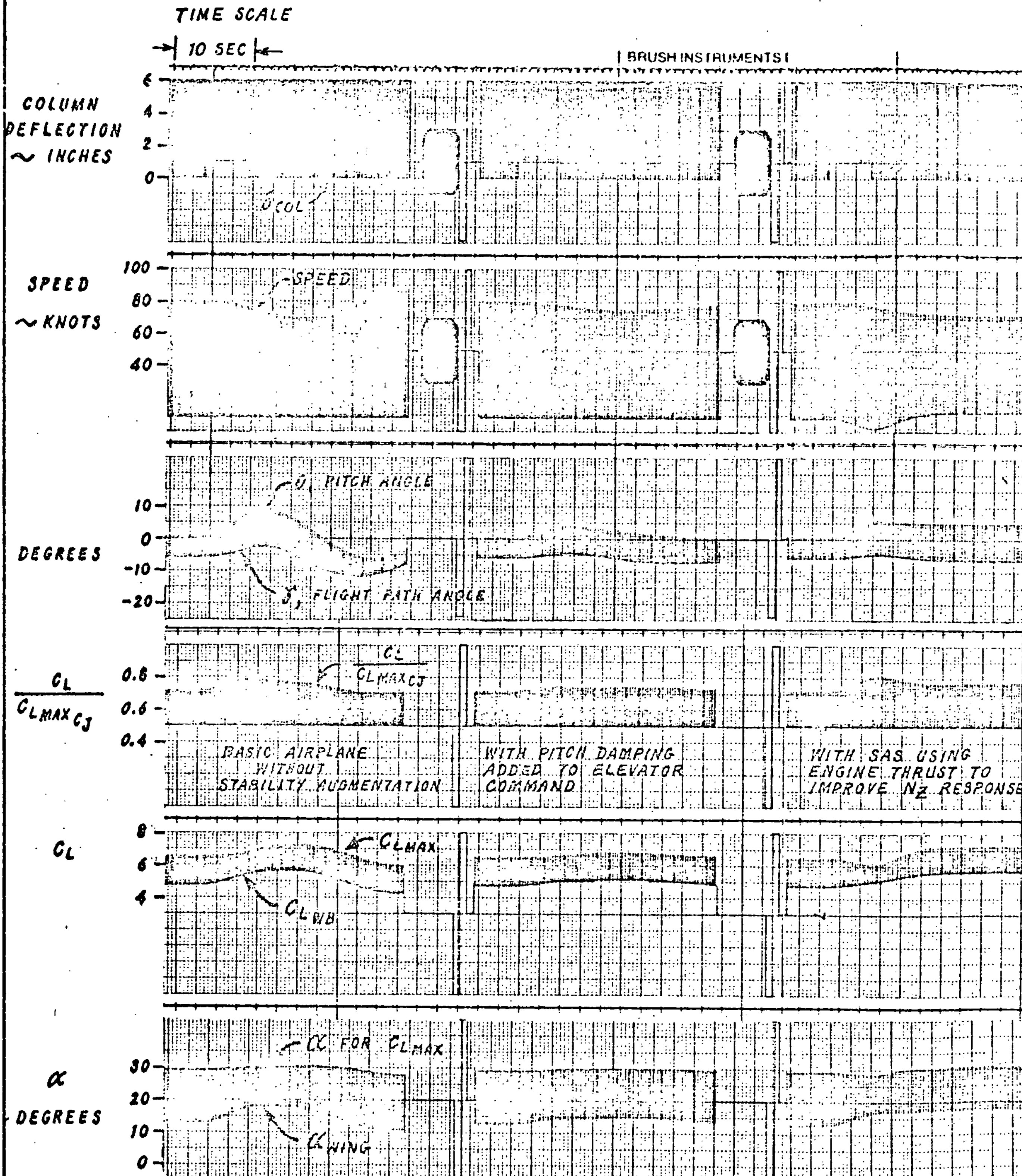
NOTE: The normal acceleration capabilities of the AW and EBF are far in excess of the minimum design value (1.45, see Section 9.0) because engine size was dictated by engine-out performance requirements. The IBF capability was lower than the design value due to a late change in aerodynamic data inputs. This latter design was not recycled for engine size and flap design.

Figure 3.2-1

3.3.2 The Effects of Stability Augmentation

Figure 3.2-2 shows typical variations of these margins (with and without Stability Augmentation Systems (SAS) of various types) in response to a step input of the column. In the basic airplane response, note that the similar relationship between C_L and speed, and C_J and speed (at constant power setting), allows the available $C_{L_{MAXCJ}}$ to grow at the same rate as the required C_L thus giving an approximately constant C_L margin. With a stability augmentation system that utilizes engine thrust (to help improve the flight path response to column inputs), the thrust changes which occur during a similar maneuver vary both the   margins and the C_L margins, and it can clearly be seen that control of such variations will necessarily become part of the design process for stability systems of this type.

In one series of tests designed to evaluate handling qualities for various SAS configurations, the results shown on Figure 3.2-3 were generated. Here the pilots were using similar evaluation maneuvers on each of several runs for each type of augmentation system. These evaluations consisted of "vertical - S" maneuvers (involving flight path variations about the nominal glide slope of ± 2 dots) and speed variations along the nominal glide slope of ± 10 knots. The figure shows the range of margin variations during evaluation runs made for each SAS; the basic airplane SAS off; a SAS involving no feed forward from the pilot's column to the throttle (denoted by MIN SAS); and a full de-coupling SAS with complete interconnects. These tests were completed for the AW and EBF airplanes only. The data show an increased variation in the lift margin at full power when the "full SAS" is used. Note, however, that this augmentation system improved the variation in the flare and gave much tighter control of angle of attack during the maneuvers.



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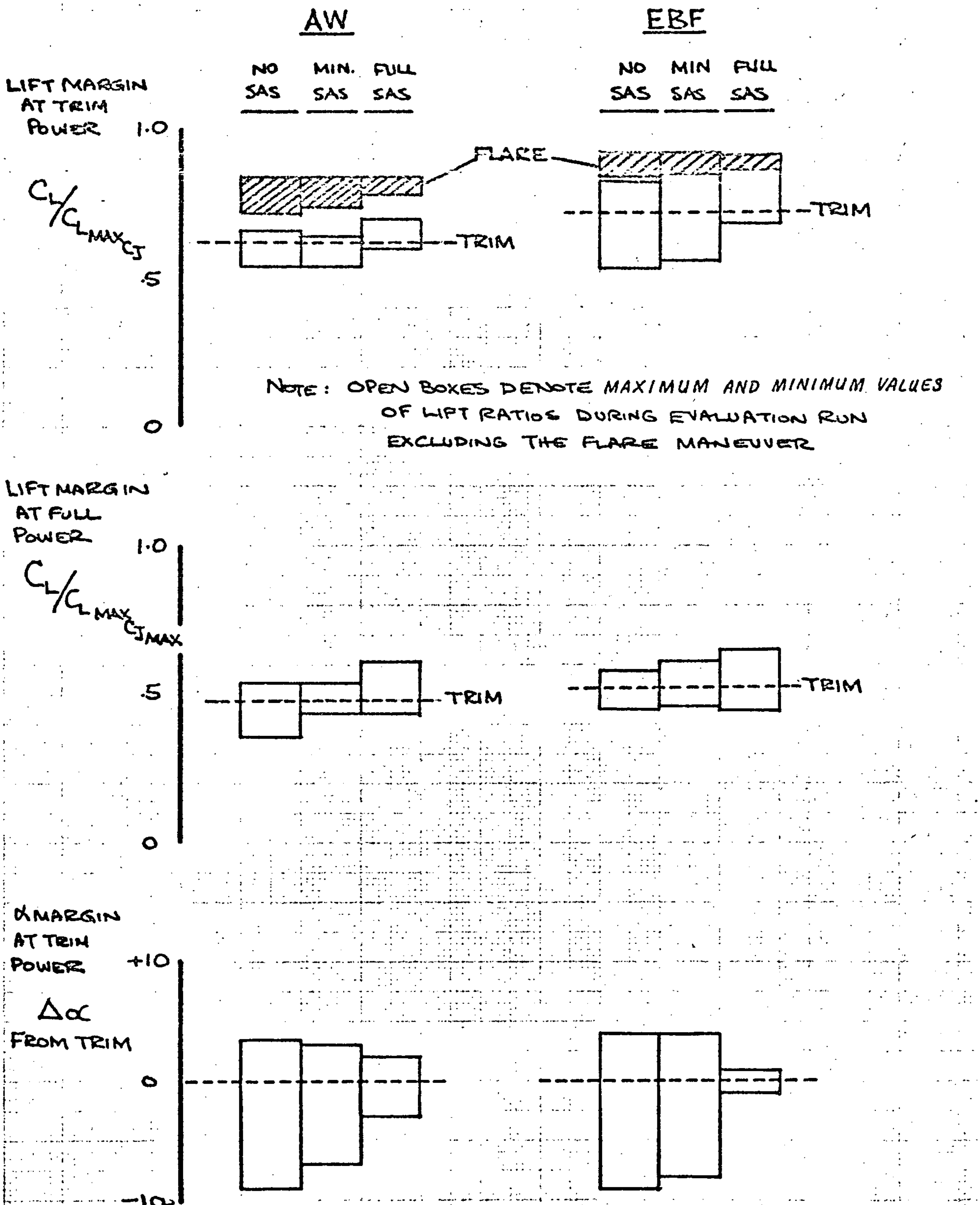

PILOT TASKS : VERTICAL "S" MANUEVERS AND SPEED CHANGES

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EFFECT OF STABILITY AUGMENTATION SYSTEMS ON MARGINS FOR THE EBF AND AW

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FIG 3.2-3

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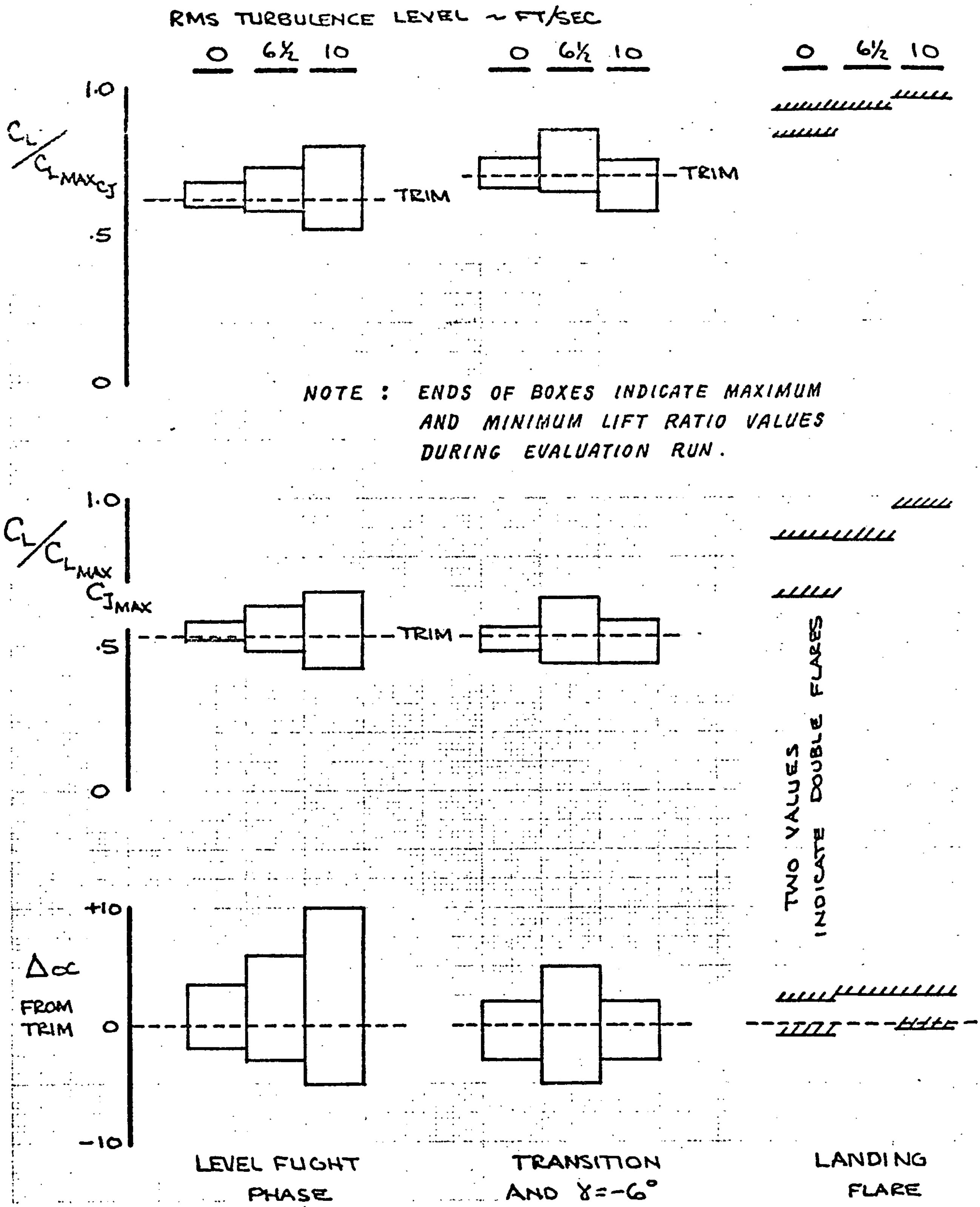
There also appears to be a small improvement in the $C_L/C_{L_{MAX}C_J}$ margin variations for the full SAS. The large difference in variation of this margin between the EBF and the augmentor wing airplane is attributed to the different rates of change of lift with power near the trim point, and the fact that the EBF design is closer to $C_{L_{MAX}}$ to begin with. The figure shows that the critical maneuver from the margins point of view is the flare. This will be dealt with more fully in Section 3.2.4.

3.2.3 The Effect of Turbulence

Figure 3.2-4 shows the effect of turbulence on the margin variations for an EBF airplane (with the minimum SAS involving no feed forward from the column to the engine). The evaluation was made in calm air and in simulated turbulence with R.M.S. gust intensities of 6-1/2 ft/sec and 10 ft/sec. Data are shown for three phases of the approach; a level flight segment approaching the glide slope; the transition phase from level flight to the 6° glide slope (including capture of the glide slope); and the final flare maneuver for landing. The effect of increased turbulence is obvious, especially during the level flight phase which was longer than the descent phase during these evaluation runs.

3.2.4 The Effect of Landing Flare and Ground Effects

The measured lift ratios included the ground effects on $C_{L_{MAX}}$, and this is the main reason why large increments of $C_L/C_{L_{MAX}}$ were detected in the flare. The increased values of this lift ratio at $C_{J_{MAX}}$ (Figure 3.2-4) illustrate the fact that the flare occurs at very close to full thrust. This powered flare technique was adopted by the pilots in order to give a reasonably flat attitude at touchdown, as required in commercial operations for passenger comfort, and also to provide good visual contact with the runway.



EBF AIRPLANE WITH MINIMUM SAS MODE

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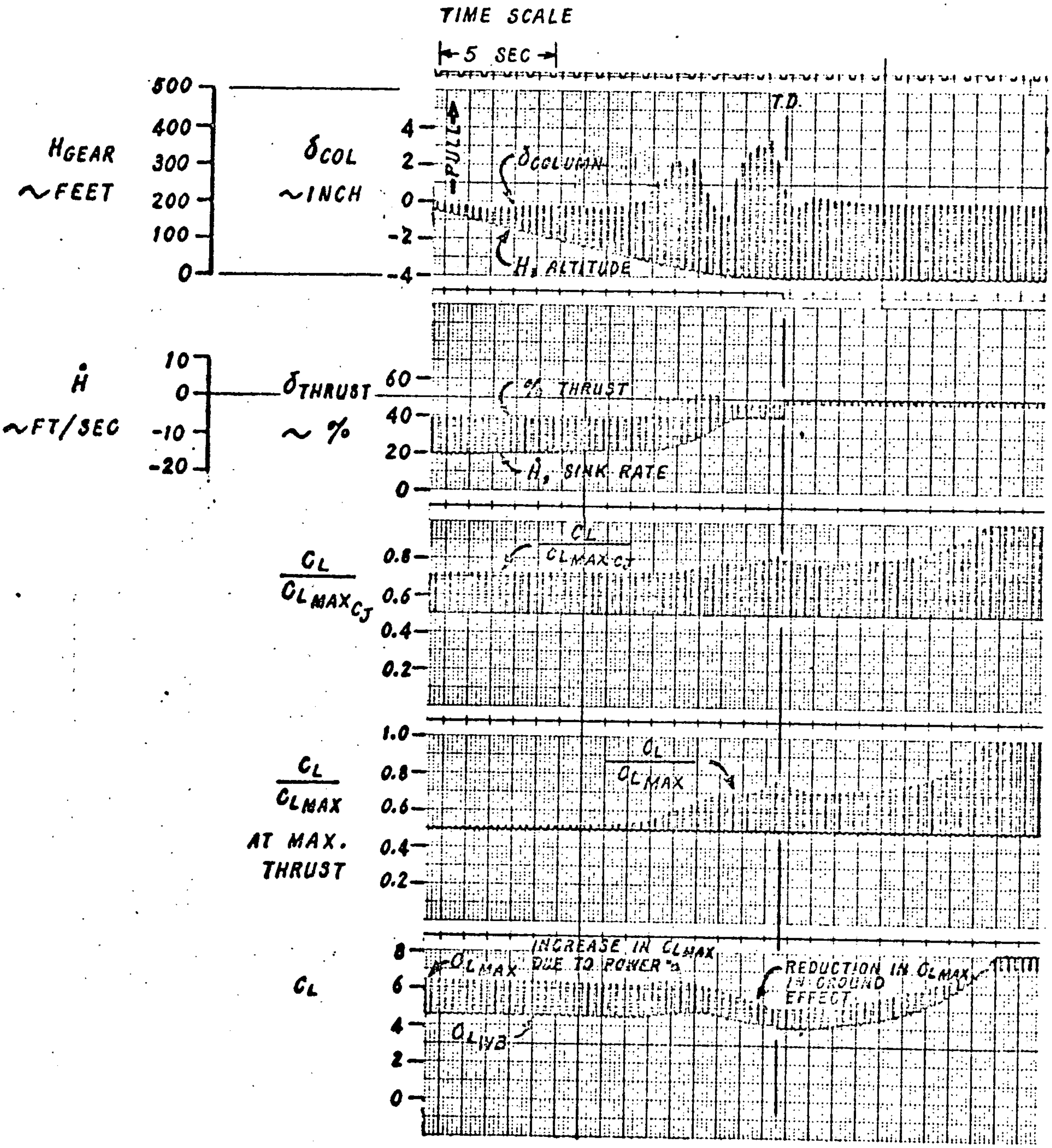
A time history of a typical flare maneuver for the EBF airplane is shown in Figure 3.2-5. The use of thrust for flare is clear, as also is the large reduction in $C_{L_{MAX}}$ in ground effect.

The loss of lift and change in $C_{L_{MAX}}$ in ground effect for the AW and EBF configurations is shown on Figure 3.2-6. Both suffer a 5-10% loss in C_L at the approach power and trim angle of attack, and about a 25% loss in $C_{L_{MAX}}$ at approach power.

In ground effects at the trim thrust setting for approach the EBF airplane has a $C_{L_{MAX}}$ only 10% larger than the trim C_L for approach, and the AW has a $C_{L_{MAX}}$ about 20% bigger than $C_{L_{TRIM}}$. It was therefore very obviously necessary to use power to flare the EBF airplane, and indeed both airplanes were flared by using power for other reasons. The increased power gave a much larger available maximum lift; about 60% greater than the trim lift coefficient for the augmentor wing, and about 40% for the EBF.

To conduct the tests described in 5.4.2.1, to determine the normal acceleration capability required for flare control in full ground effect, further limits were put on the $C_{L_{MAX}}$ available. The effect of these reduced $C_{L_{MAX}}$ values on the complete flare while entering ground effect from free air was not checked. However, it is logically expected that airplanes which meet the total requirements of 5.4.2.1 (both in free air and in ground effect) will be acceptable.

For reference purposes a summary of the average thrust values used in the flare is given on Figure 3.2-7. There are three equivalent parameter values given to the vertical axis; actual % of maximum thrust; the equivalent steady state flight path angle that this thrust level would give at the reference approach speed; and the equivalent sink rate for this steady state



NOV. 17 COND 068

CALC			REVISED	DATE	MARGIN VARIATIONS IN THE FLARE MANEUVER EBF
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APPD					
APPD					THE BOEING COMPANY

D6-40409

FIG. 3.2-5

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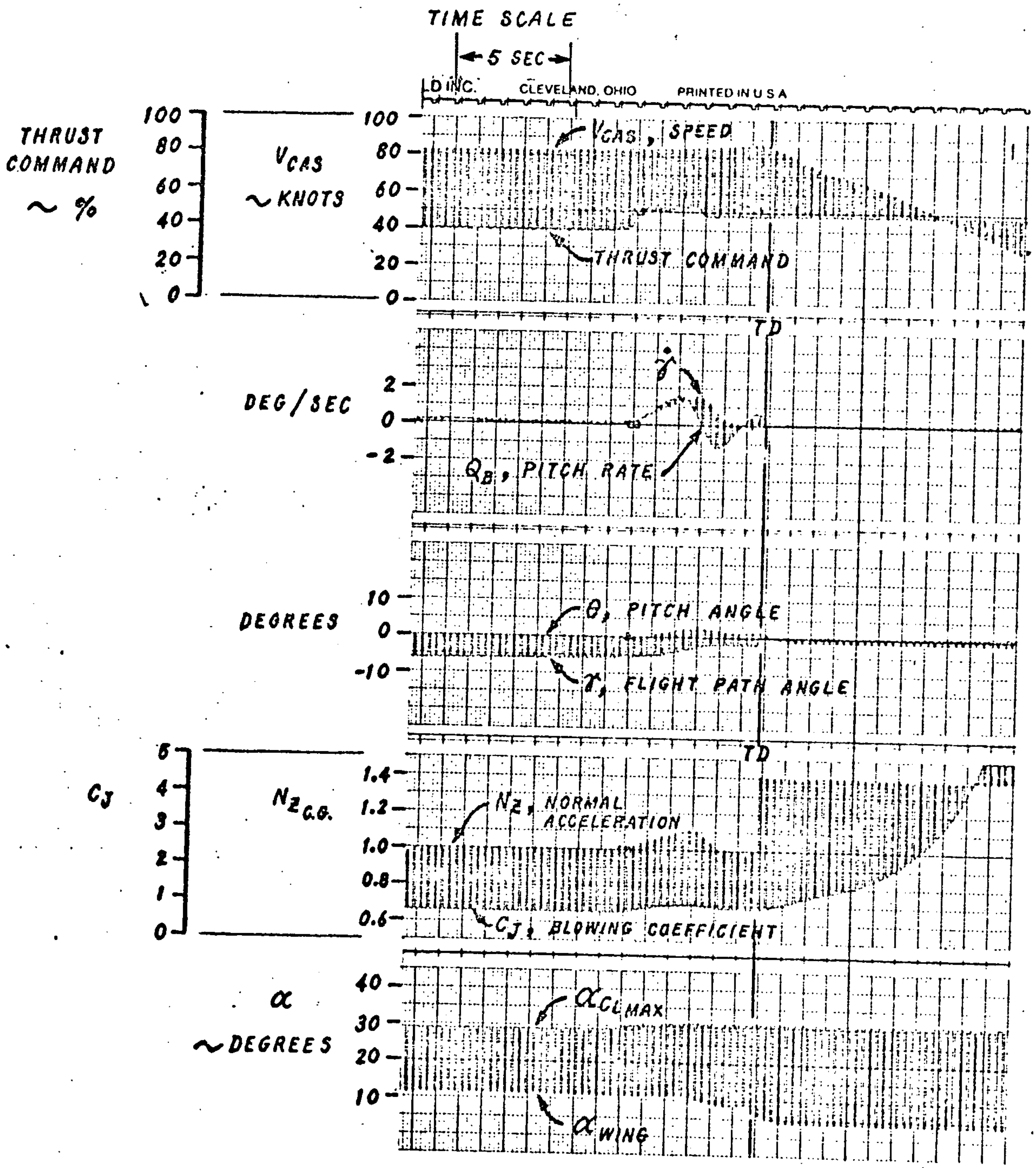

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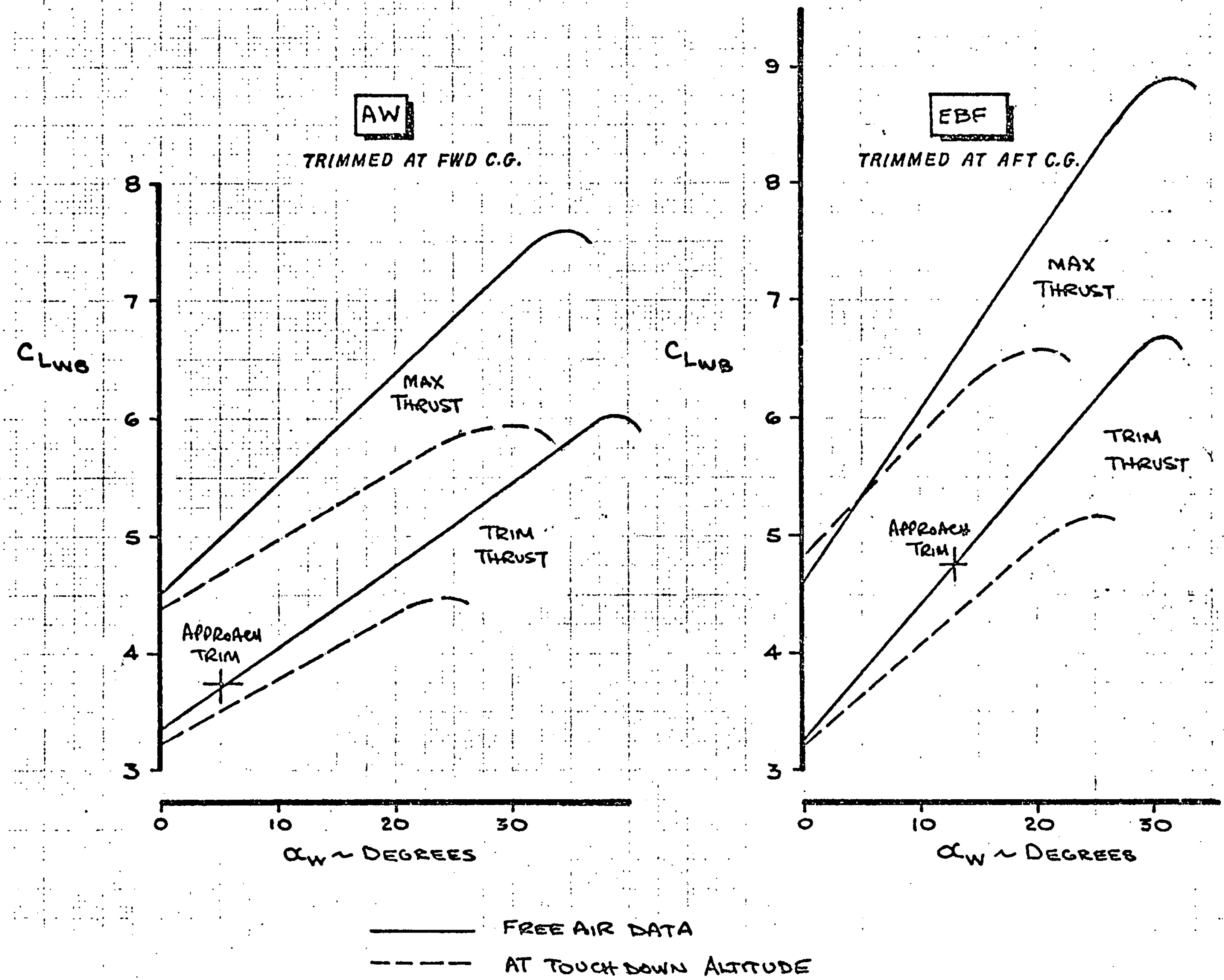
CALC			REVISED	DATE	MARGIN VARIATIONS IN THE FLARE MANEUVER EBF	D6-40409
CHECK						FIG. 3.2-5 (CONTINUED)
APPD						
APPD					THE BOEING COMPANY	PAGE 29



GROUND EFFECT ON LIFT AW & EBF

FIG 3.2-6

THE BOEING COMPANY

From the present study, and from previous work at Ames Research Center and at Boeing, it appears that the use of power early in the flare aids the pilot in producing more consistent touchdown sink rates from steep approach paths. A thrust increase at 40 to 50 feet altitude causes the approach sink rate to reduce and the pilot can more easily control the airplane to the ground and perform a more consistently judged flare with the elevator. This so-called "drift-down" flare appears to take place at about a 3° flight path angle, or 7-8 ft/sec., according to Figure 3.2-7. The beneficial effect on lift margin during the flare is obvious from Figure 3.2-6 and these data can be used as a guide for checking flare margins on other proposed STOL designs.

3.2.5 The Effect of Turns and Maneuvers

Margin variations in smooth turning flight were small for operational bank angles. Figure 3.2-8 shows a condition where the power increase required to maintain speed in the turn caused the $C_{L_{MAX}}$ to increase at about the same rate as the lift required to maintain the turn, and the maneuver was completed at constant margin. This behavior was typical of both the augmentor wing and EBF types at small bank angles. To investigate this effect at larger bank angles, typical flight test maneuvers were simulated including wind-up turns and stalls. Figure 3.2-9 shows a wind-up turn to a maximum of 45° bank angle, with only a slight loss of margin at the high angles. Analysis of a series of approaches involving positioning maneuvers using steadily increasing bank angles yielded the data in the upper plot on Figure 3.2-10. With the wings level the normal acceleration margin from stall was .63 g at trim thrust, and .82 g at maximum power. As increasingly larger bank angle maneuvers were completed, the speed, angle-of-attack, and engine thrust variations cut into the available margin as shown by the shaded area on the figure. In perfectly smooth flight in a conventional airplane the loss in margin would follow the solid line labeled "no powered lift" and equal the acceleration level used to maintain the given bank angle. The STOL airplane does considerably better than this, at least up to the bank angle which requires full power to maintain speed in level flight.

The margin available at $\phi = 45°$ is still of the order of .4 g, where the conventional airplane would be reduced to .2 g. The margin from stall at maximum power, of course, shows exactly the same loss in the maneuver as the CTOL airplane would.

The lower plot on Figure 3.2-10 shows similar results calculated for steady maneuvers on the augmentor wing airplane. Since the trim thrust value was a higher percentage for this airplane, full power would be reached at a lower bank angle than for the EBF. Nevertheless, a considerable improvement in maneuver margin over the conventional airplane is shown.

If these charts are used to size STOL airplane margins so that they may maneuver up to 1.5 g before stall (same as CTOL), then a margin of about .35 g <u>at the trimmed power level</u> would suffice for STOL. This value would vary, depending on how close to full thrust the trimmed flight condition turned out to be.

3.2.6 Margins at Low Power Settings

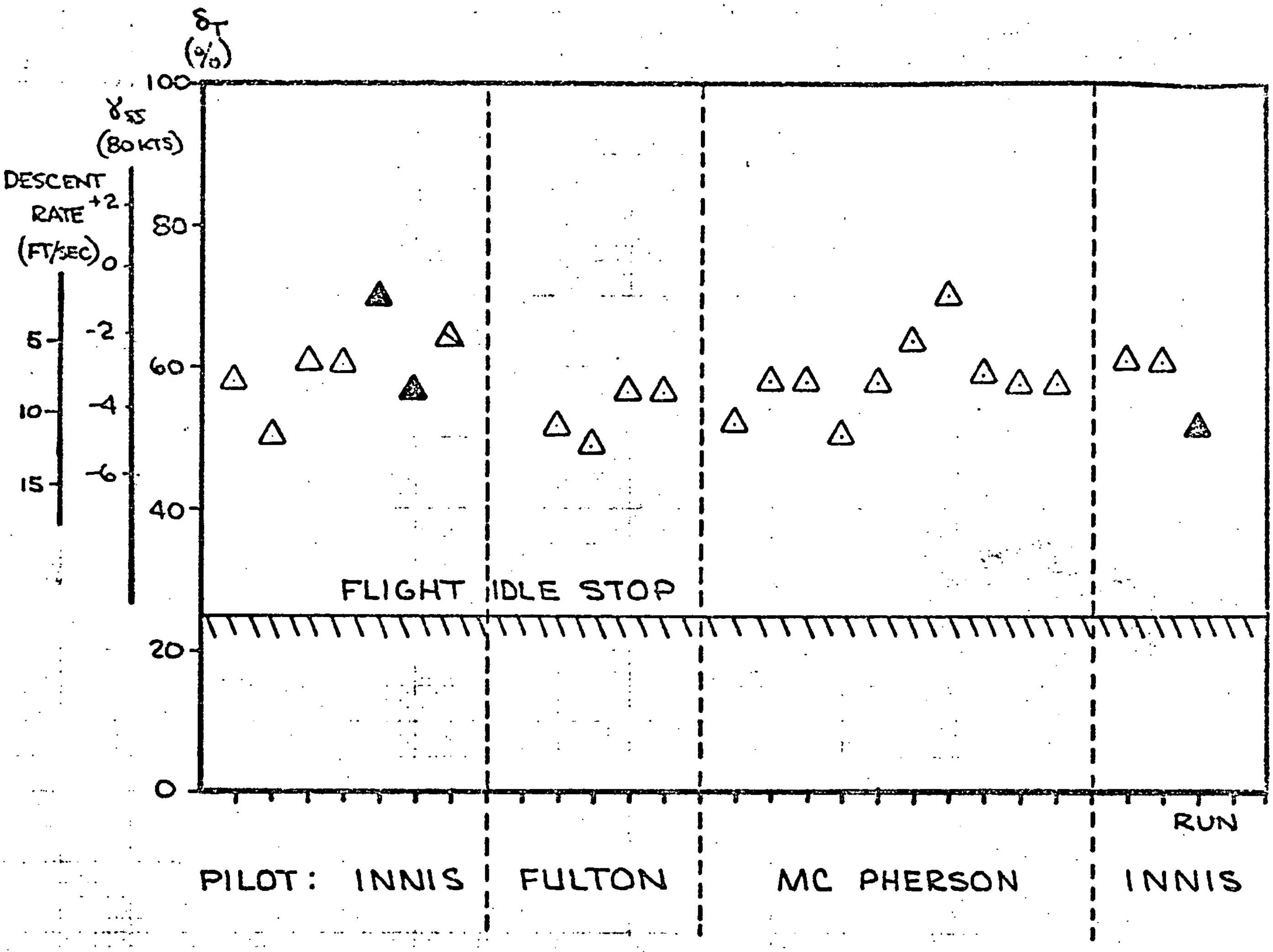
In general, the transition maneuver from level flight to the 6° glide slope (and other maneuvers requiring momentarily low power settings) were accomplished without excessive reduction in the available lift margin. However, if rapid thrust reductions are made during maneuvers the gust margins can be greatly reduced and an inadvertent stalled condition may result.

During the present study an evaluation was made of such maneuvers and

Figure 3.2-11 illustrates a particular flight in which there was no lower limit (or "flight-idle stop") on the engine settings that the pilot could select. Having been briefed to make rapid and precise flight path changes using thrust, the pilot pulled the throttle back to a very low setting producing an inadvertent stall condition. Although recovery from the "stall" was immediate upon thrust increase and the landing was successfully completed, this can only be attributed to the docile stall characteristics "engineered" into the simulation. Real life problems, such as asymmetrical wing stall, buffeting, or loss of lateral control would be unacceptable, and such inadvertent events must be guarded against with positive "stops" in the engine controller. Figure 3.2-12 shows a repeat condition with such a "stop" simulated. The setting of these in-flight engine limits so that they do not interfere with required performance capability is dealt with in Section 3.3. Similar "stalled conditions" can be encountered by engine thrust reductions during wind-up turn maneuvers as shown in Figure 3.2-13. Again, the ease of recovery from this condition is due mainly to the simulation of docile stall characteristics.

A more realistic rapid transition maneuver from level flight to the 6° glide slope is shown on Figure 3.2-14. This is an example of a late transition which required power reduction to the flight idle stop and is included here as being representative of the worst design condition. At the worst point in the maneuver the equivalent load factor increment available before stall was .22 g ($C_L/C_{L_{Max}}$ = .82). On the steady glide slope the margin was restored to .42 g. Consequently, there is a need for at least a .2 g maneuver margin at trim thrust on the glide slope just in order to make a rapid transition without stalling. To provide margin from stall warning will probably require a further .1 g. The total margin from stall at approach thrust must therefore be about .30 g.

AUGMENTOR WING
LONGITUDINAL FULL SAS
FLARE



SYMBOL	RMS TURBULENCE ~ ft/sec.
△	STILL AIR
▲	6.5
◬	10.0

AVAGNINA 12/13/71

MAXIMUM THRUST
FOR FLARE MANEUVER

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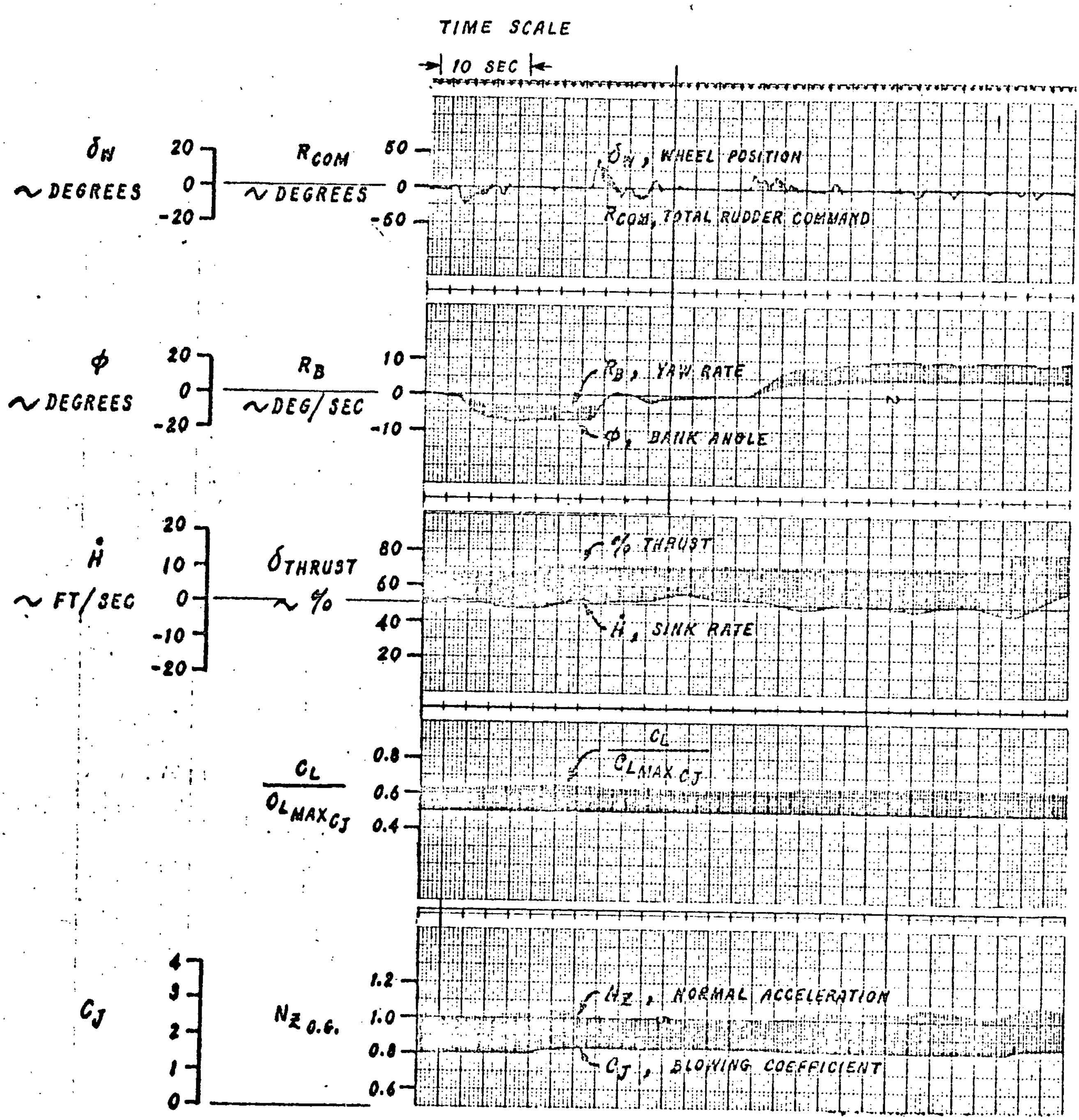
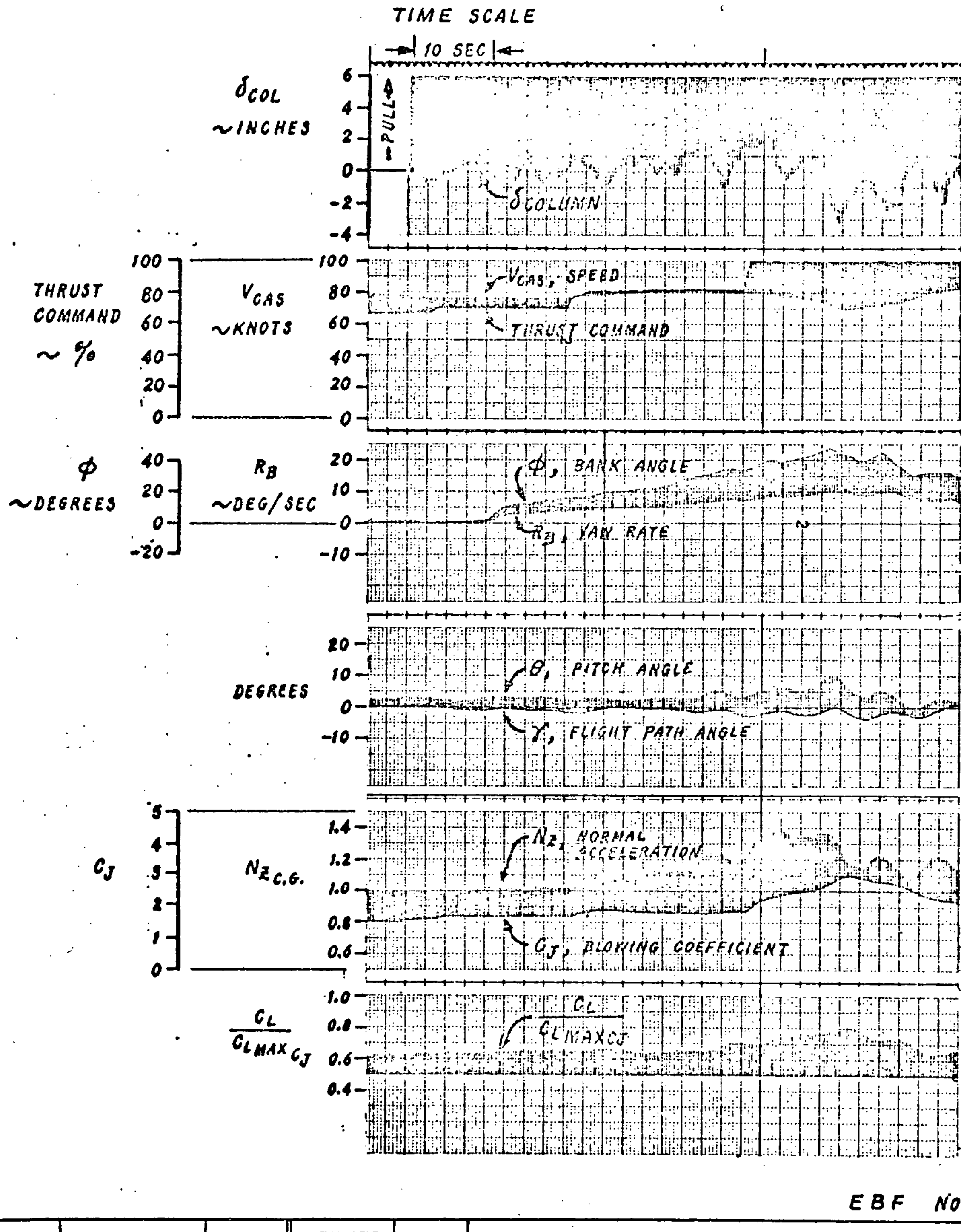
FIG 3.2-7



EBF NOV. 16, COND. 041

CALC			REVISED	DATE	MARGIN VARIATIONS IN TURNS	D6-40409
CHECK					ϕ = 15° AND 20°	FIG. 3.2-8
APPD						
APPD					THE BOEING COMPANY	PAGE 35



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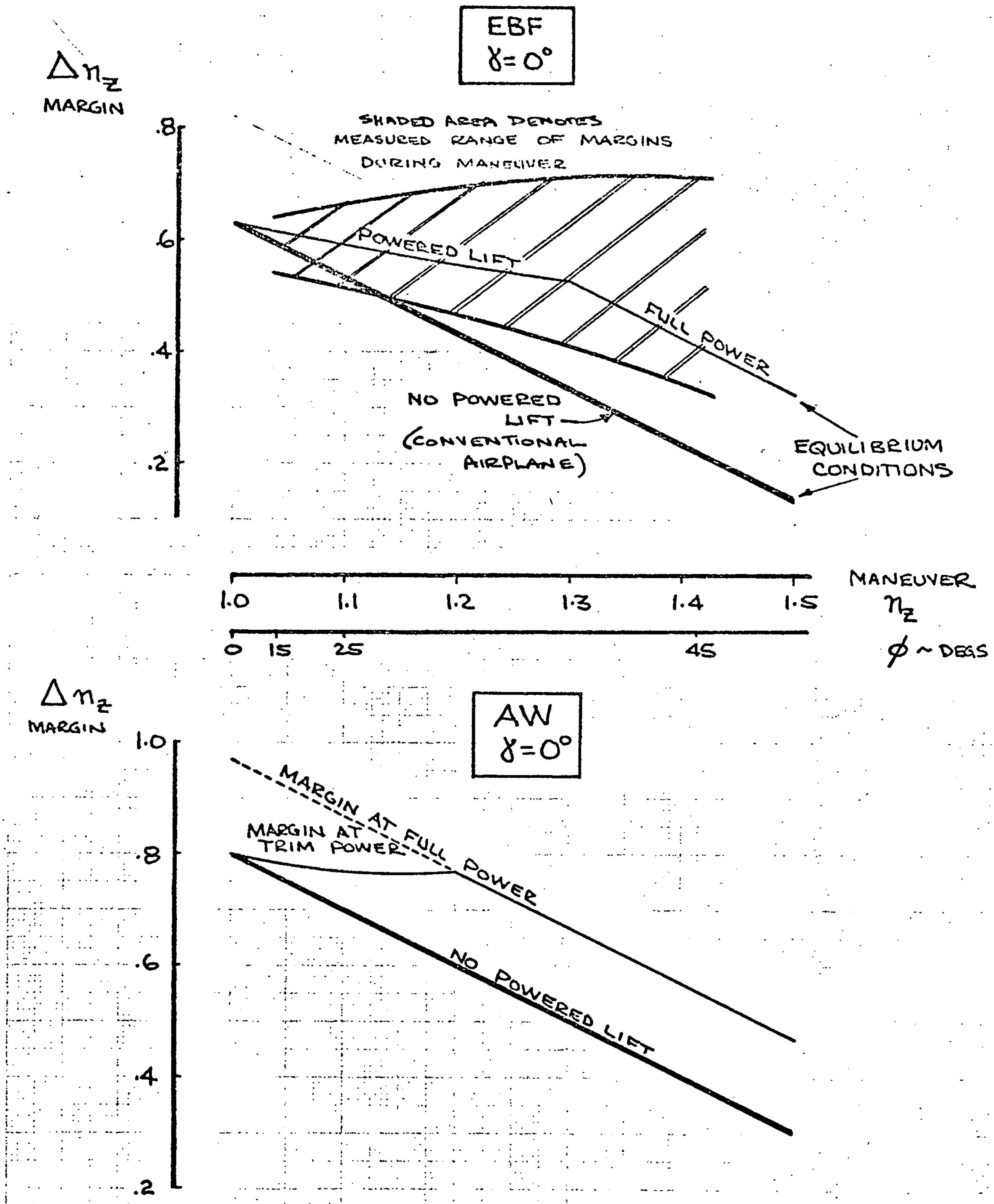
CALC			REVISED	DATE	MARGIN VARIATIONS IN A WIND-UP TURN MANEUVER $\phi = 45°$	D6-40409
CHECK						FIG. 3.2-9
APPD					THE BOEING COMPANY	PAGE 36
APPD						



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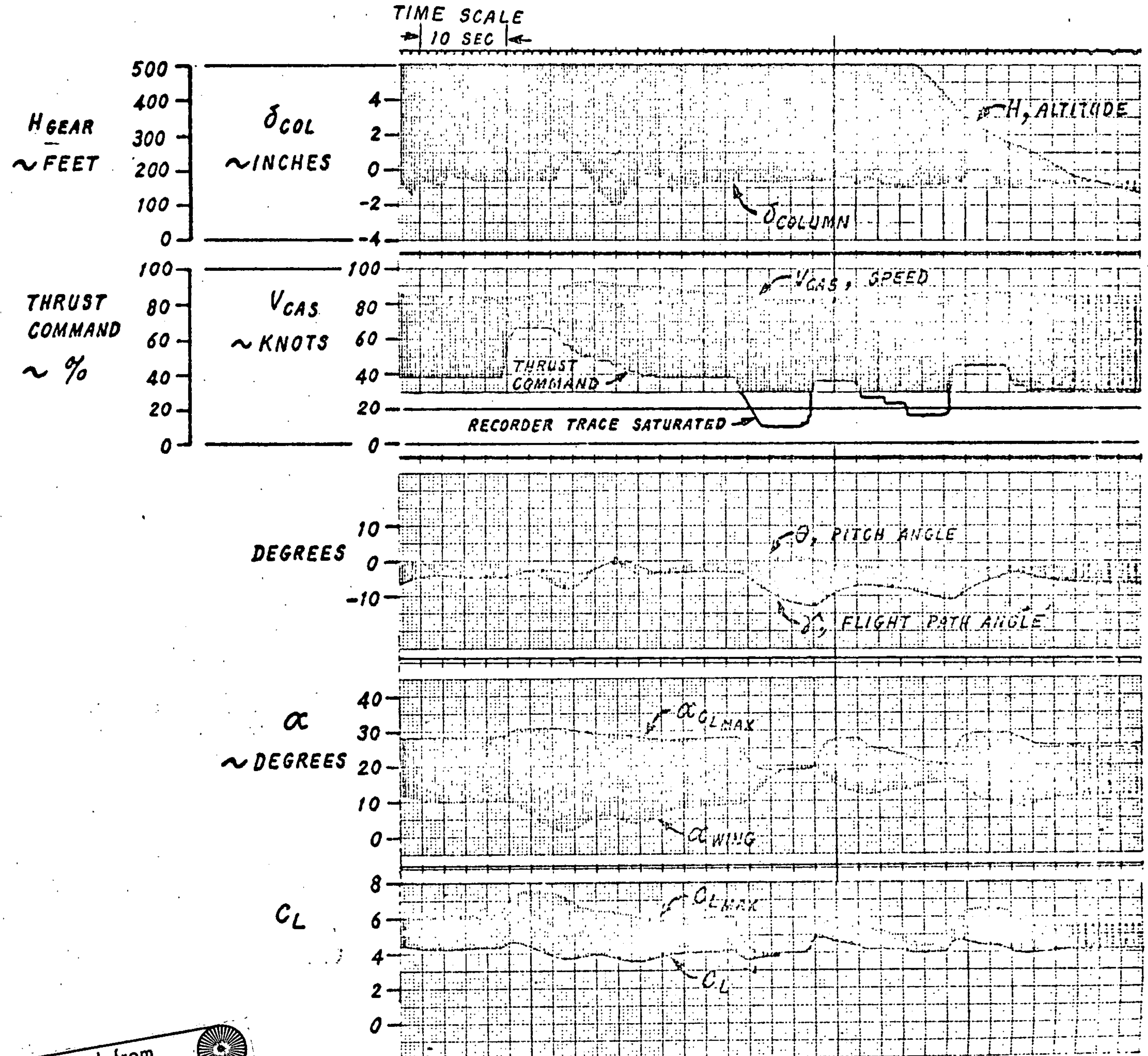
EFFECT OF TURNING FLIGHT ON LIFT MARGINS

D6-40409

FIG 3.2-10

THE BOEING COMPANY



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"STALL" CONDITION CAUSED BY POWER REDUCTION TO IDLE

FIG. 3.2-11

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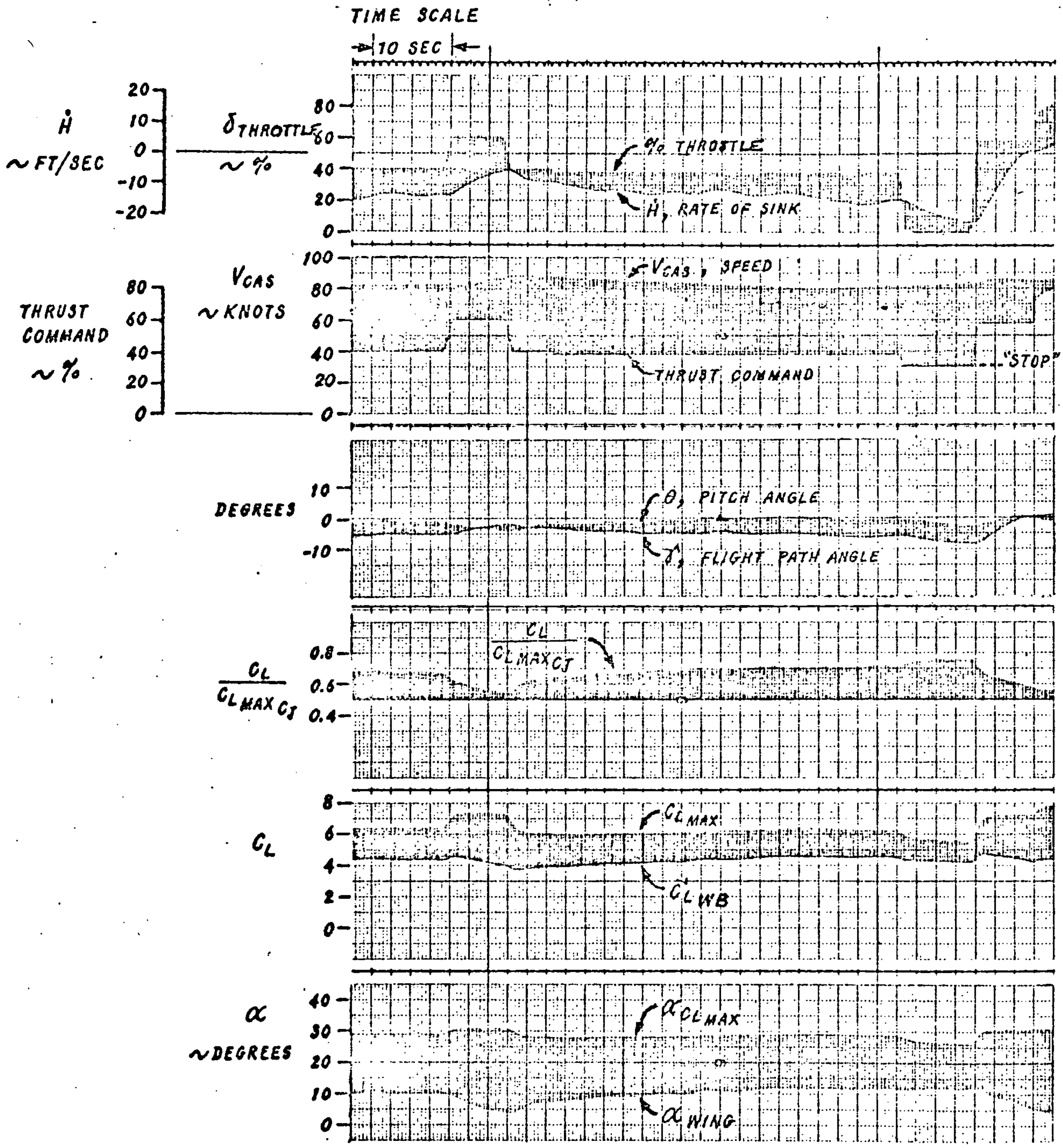
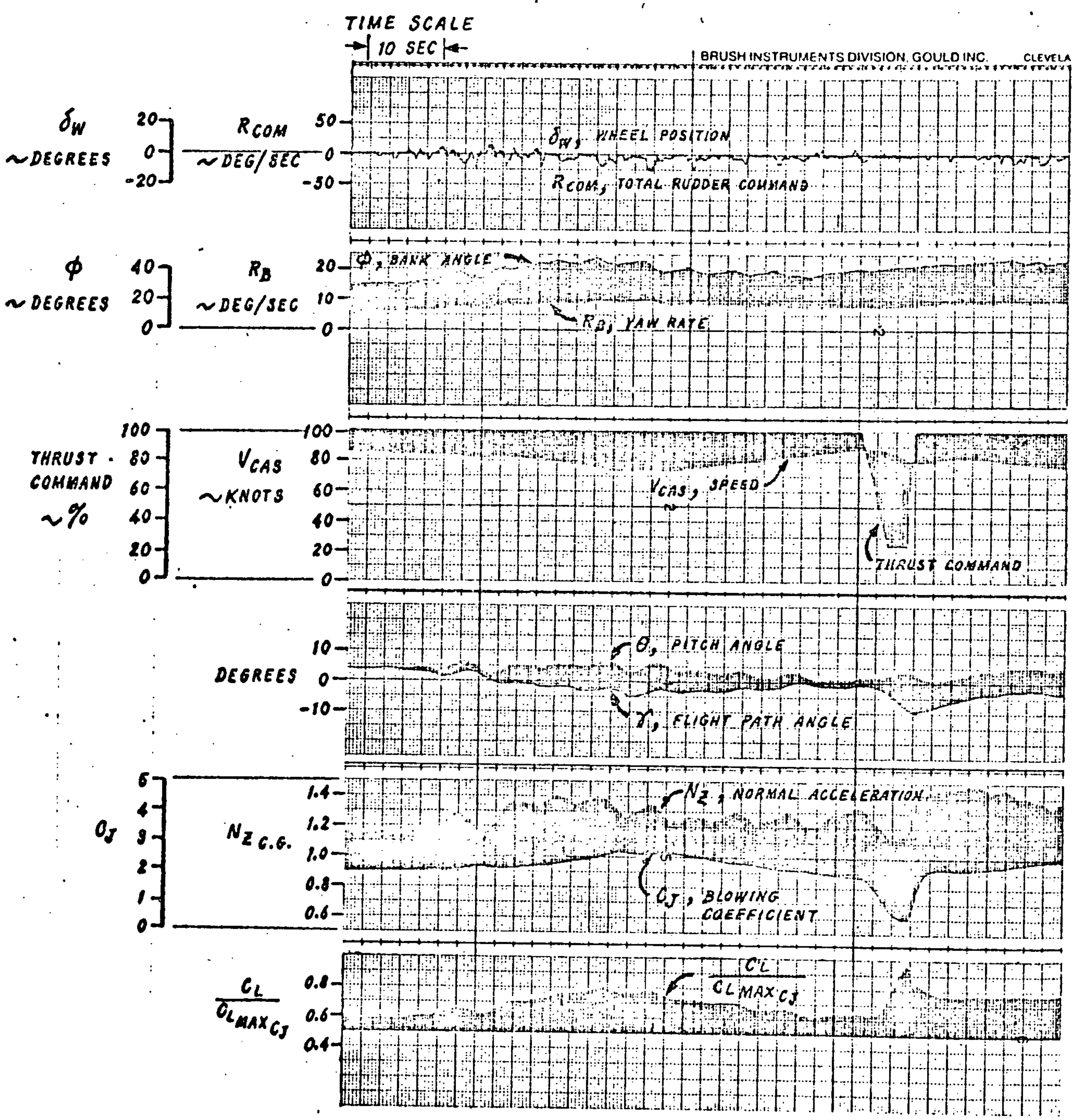


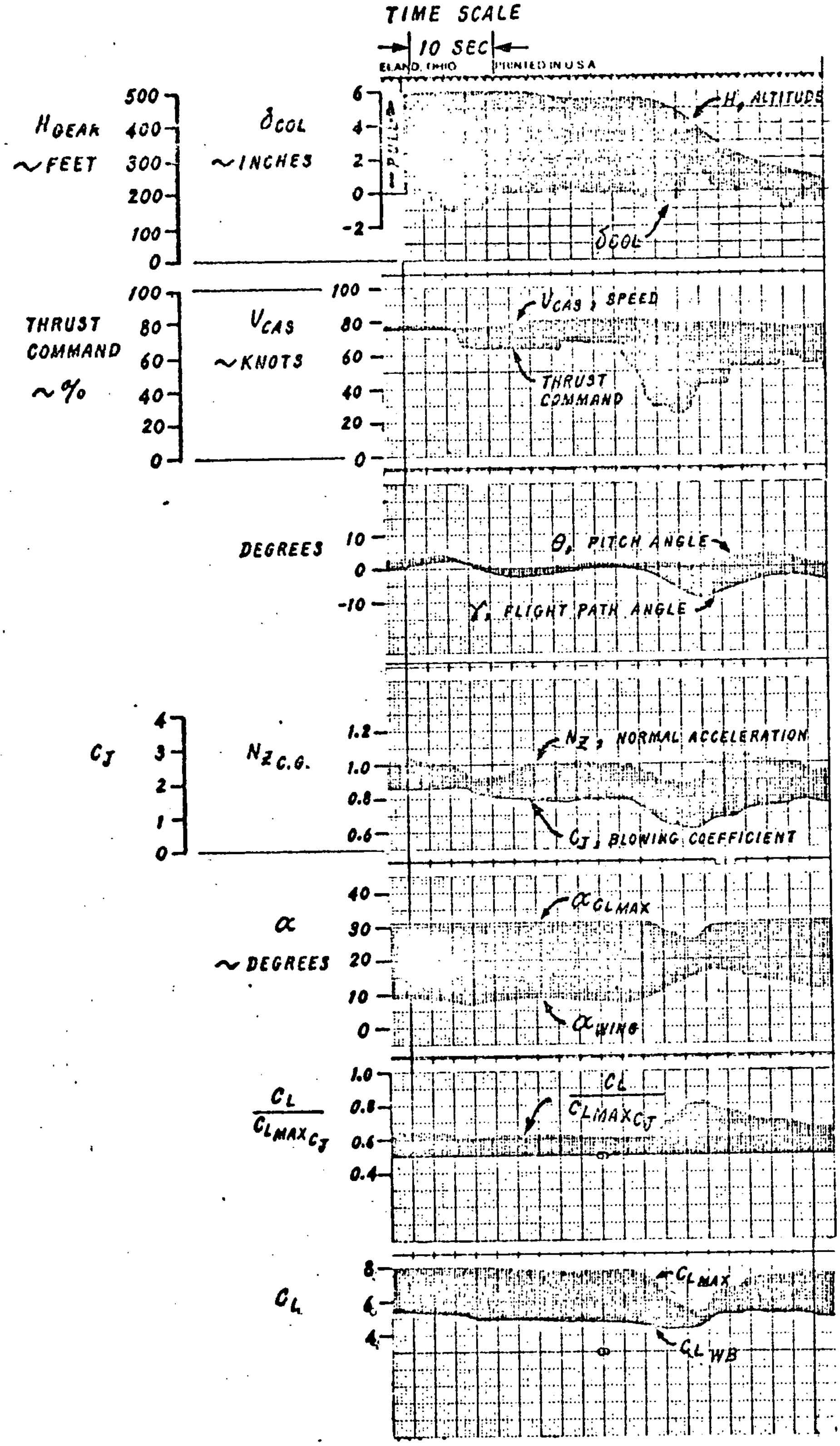
EBF NOV. 10, COND. 025

CALC			REVISED	DATE	PROTECTION FROM INADVERTANT STALL WITH FLIGHT IDLE STOP	D6-40409
CHECK						FIG. 3.2-12
APPD						
APPD					THE BOEING COMPANY	PAGE 39



EBF NOV. 16, COND. 044

CALC			REVISED	DATE	EFFECT OF POWER REDUCTIONS DURING MANEUVER TO $\phi = 45°$	D6-40409
CHECK						FIG. 3.2-13
APPD						
APPD					THE BOEING COMPANY	PAGE 40



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CALC			REVISED	DATE	MARGIN VARIATIONS DURING A RAPID TRANSITION MANEUVER	D6-40409
CHECK						FIG. 3.2-14
APPD						
APPD					THE BOEING COMPANY	PAGE 41

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3.2.7 Choice of Lift Margins for Maneuvering

In the following discussion the term "margin" is used to relate the wing-body lift coefficient in the flight condition or maneuver under study to the maximum wing-body lift coefficient available at the power setting for which the margin is specified. This lift ratio is quoted in terms of an effective normal acceleration margin. In actual flight the normal acceleration capability to stall would differ from this calculated value due to lift from the tail and the dynamics of the maneuver.

Summarizing the examples of margin variations discussed in Sections 3.2.1 through 3.2.6, the most critical flight maneuvers appear to be the landing flare and the transition from level flight to the glide slope, or, a capture of the ILS beam from a flight condition above the beam and high in speed. The flare maneuver is considered to be a normal requirement and the specified margin is that the acceleration requirement of Section 5.4.2.1 is available without running into stall warning. Guidance on design rules for this case are given in Section 3.2.4. The capture of the glide slope, and the setting of the flight-idle engine stop, are related to the flight path control requirement of Section 5.4.1.2. This is also a normal maneuver and must be free of stall warning and retain a level of protection from gusts. Guidelines for this are discussed in Section 3.2.6.

The selection of generalized margins to $C_{L_{MAX}}$ for the purpose of gross maneuvers and collision avoidance is still difficult, because this background material has shown that the required margins are to some extent a function of airplane configuration, trim conditions, and handling qualities. Stability augmentation system design must take into account the variations in lift margins caused by the control of flap angle and thrust. Figure 3.2-15 shows an example of how a properly designed SAS can achieve large speed and flight path variations at constant lift margin.

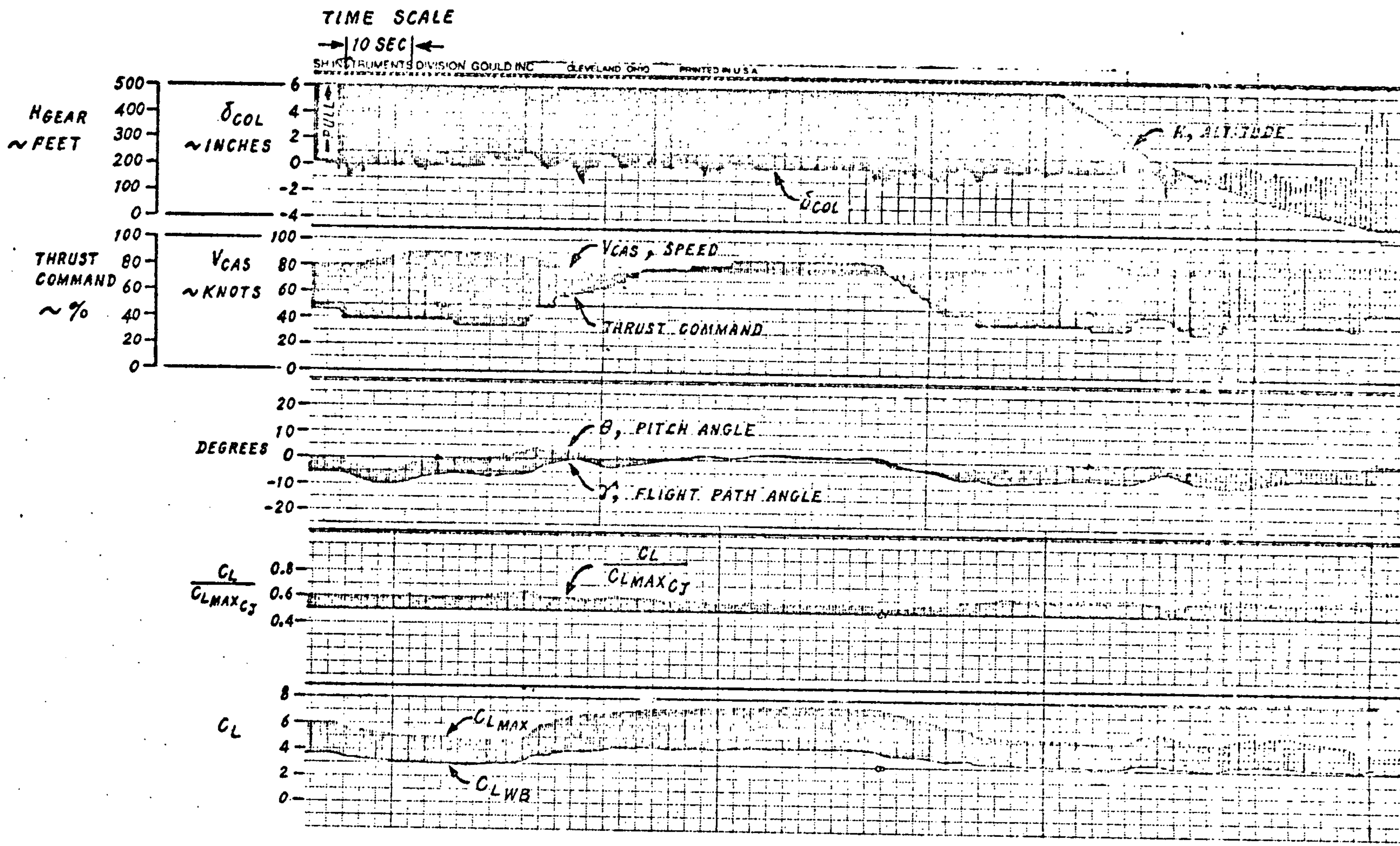
Also, the question of whether margins to $C_{L_{MAX}}$ should be tied to trim power or be referred to the lift available at maximum power is not completely resolved. Figure 3.2-10 shows that margins referred to trim thrust should take into account the beneficial effect of added power in the turn. Also relevant to this discussion is the fact that conventional handling qualities were only achieved in these airplanes by the use of a stability augmentation system (more correctly a command and stability augmentation system) which linked engine thrust to the column. Thus increased power was available to the pilot in all maneuvers. The discussion in Section 3.2.5 results in a requirement for a .35 g margin at the trim thrust, or .50 g margin at full thrust, to match current airplanes margins.

The margin at trim thrust to cover the transition maneuver was .3 g, as discussed in Section 3.2.6, and is therefore not limiting.

3.2.8 Protection from Gusts

For the purpose of comparing STOL gust protection requirements with the capability of current airplanes, a step gust input will be used. This is simpler to analyze and the results of simulated flight in turbulence will be used to modify the discussion to the continuous gust spectrum case.

Figure 3.2-16 relates the step vertical gust level to the equivalent angle-of-attack increase as a function of approach speed. The shaded area on the figure in the region of 125-135 knots represents the stalling angles of attack for a series of commercial jet transports. To match this same gust protection (about 20 knots) the STOL airplane requires a larger margin between the trim angle-of-attack and the angle-of-attack for $C_{L_{MAX}}$.



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A MANEUVERING APPROACH WITH SMALL MARGIN VARIATIONS

FIG. 3.2-15

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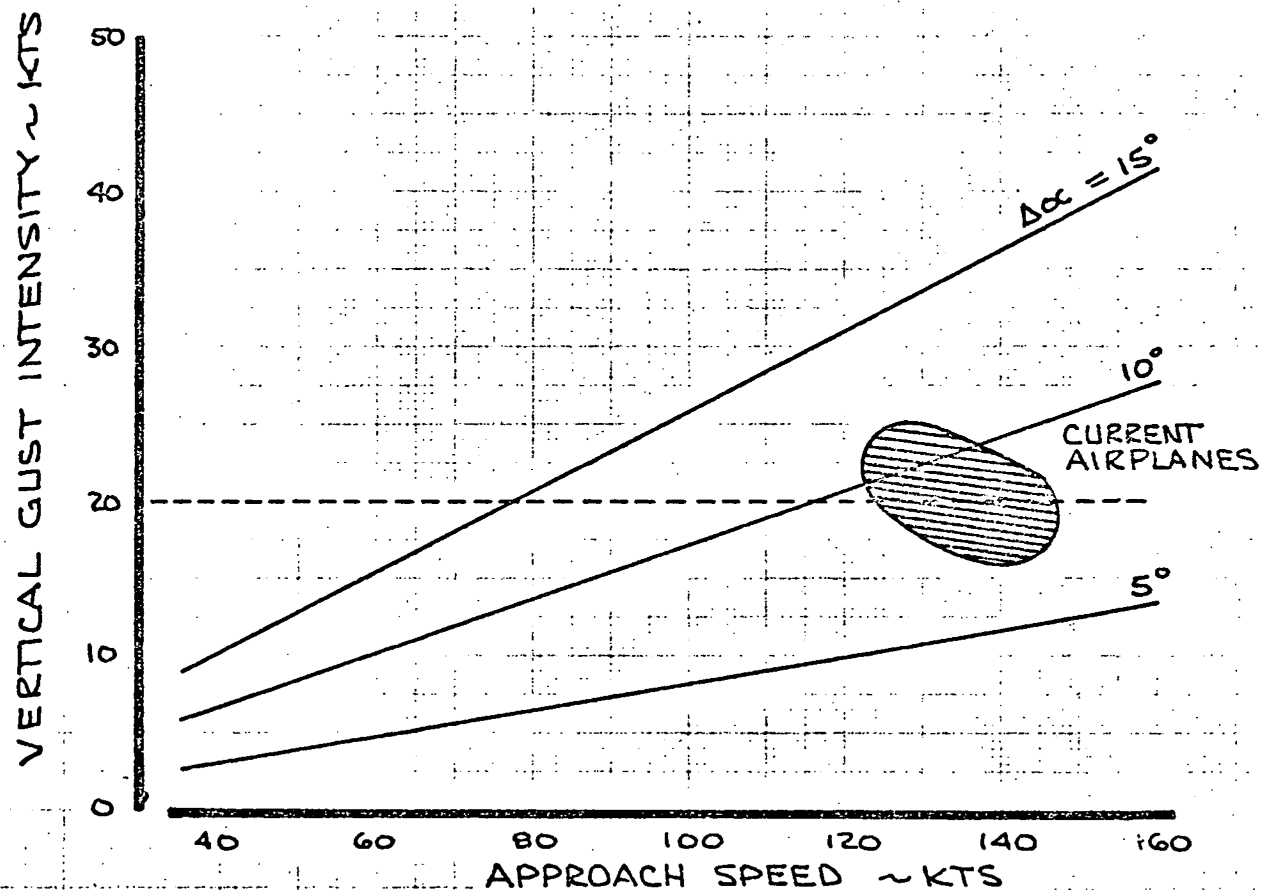
STEP GUST



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APR					
APR					THE BOEING COMPANY

D6-40409

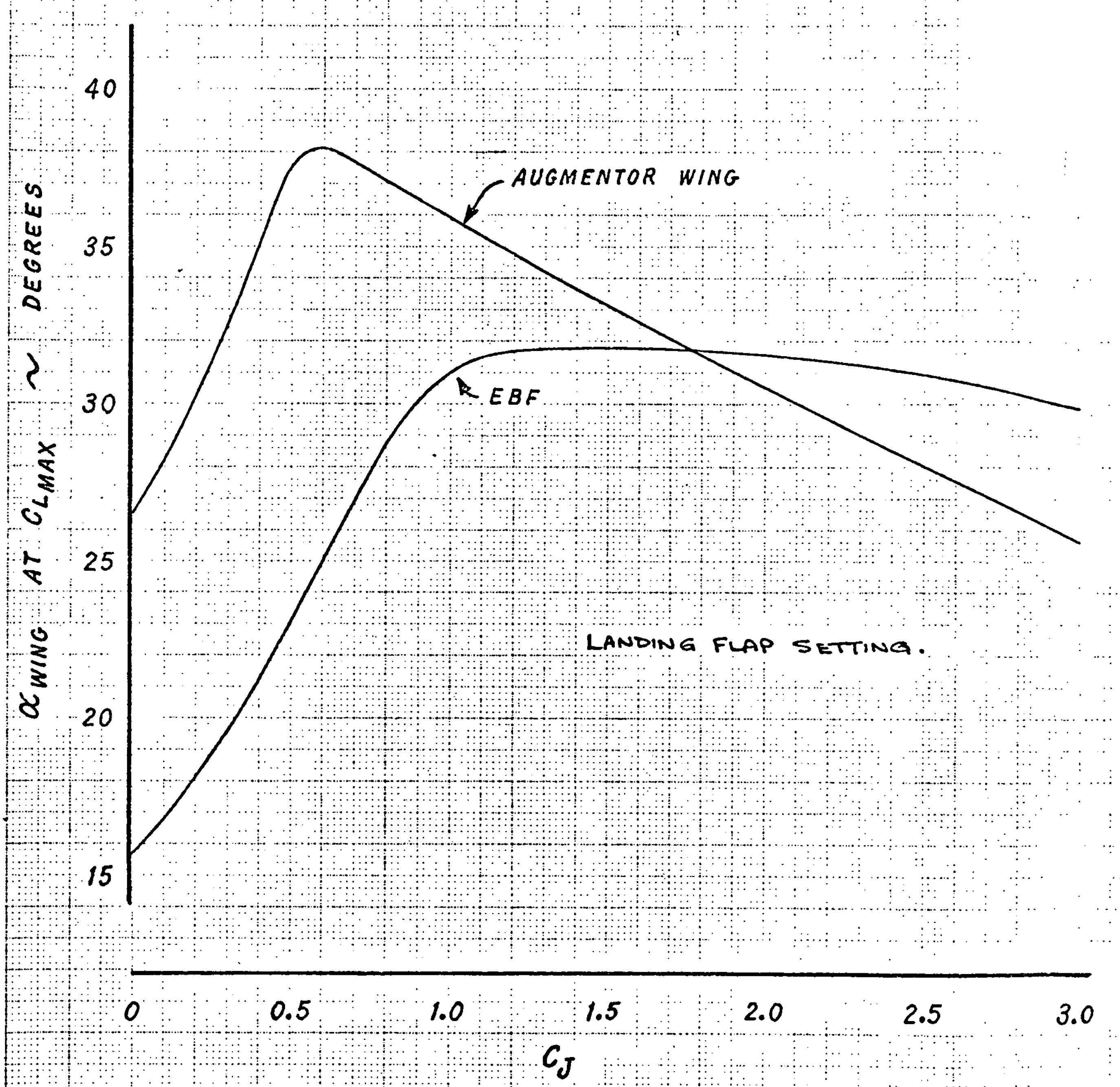
FIG. 3.2-16

If the STOL airplane lift characteristics at stall are different from conventional airplanes then the requirement for protection may possibly be changed. If very small lift losses are sustained after stall then α margins could possibly be reduced. The real requirement here is that no dangerous loss of lift should occur, and as operational experience with STOL grows it should be expected that airplanes with little or no loss of lift after $C_{L_{MAX}}$ may well be able to reduce this margin.

In the present study α variations during maneuvers and flight in turbulence were continually monitored and compared with the angle-of-attack at which maximum lift would occur at the power setting in use, $\alpha_{C_{L_{MAX}}}$. This latter variable was affected strongly by the power setting as shown in Figure 3.2-17. An example of the effect of this was a condition in which a pilot was asked to maneuver through ±5° flight path changes during an approach by using the throttle. Figure 3.2-18 shows the resulting +7°, -11° changes in α_{MARGIN} due to the thrust changes needed for this maneuver. Variations in angle-of-attack in simulated turbulence were relatively small, the most important margin changes again being due to the thrust variations required to maneuver. Even in the flare, (see Figure 3.2-5), angle-of-attack margins were not critical. Flare techniques developed by the pilots included the use of thrust with only 3° to 4° of attitude change, thus minimizing the α changes in the flare. Based on these data the margin in angle-of-attack has been set as equivalent to a 20 knot step gust (15° α at 80 knots), in order to "match" current jet transport capability.

3.2.9 General

It is recognized that the choice of actual operational margins will still depend on actual simulation work on specific airplanes which analyze maneuvers similar to those used in this study. Sufficient work has not yet been done to decide the extent to which these margins are dependent upon specific configuration details. However, the two types of high lift systems studied here did show similar margin requirements, and the results have been used to define a standard for the evaluation of competing concepts.



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VARIATION OF α_{WING} AT C_{LMAX} WITH BLOWING COEFFICIENT

THE BOEING COMPANY

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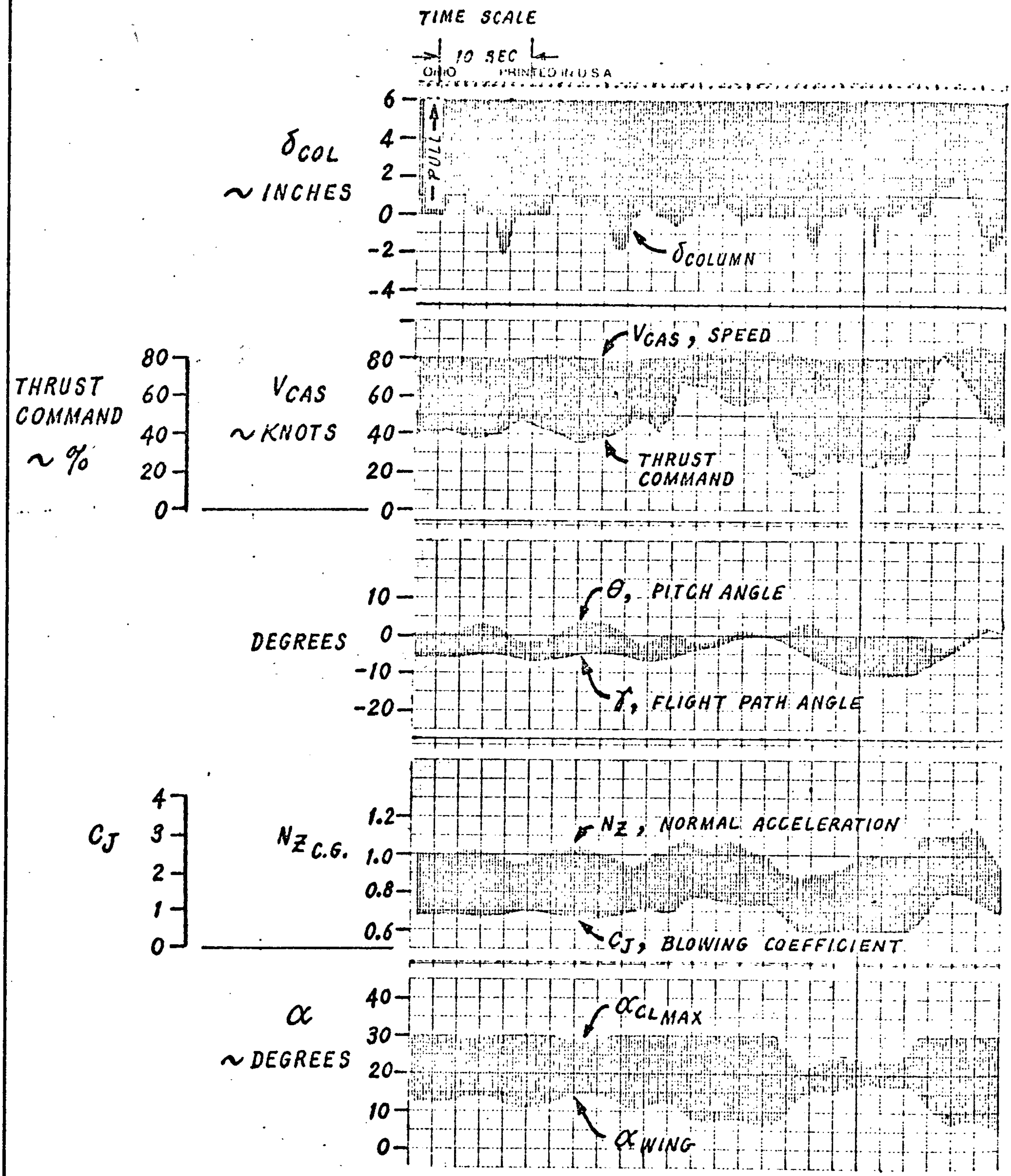
FIG. 3.2-17



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CALC			REVISED	DATE	VARIATIONS IN ANGLE OF ATTACK MARGINS IN MANEUVERING FLIGHT	D6-40409
CHECK						FIG. 3.2-18
APPD						
APPD					THE BOEING COMPANY	PAGE 49

3.3 Flight Path Margins

Criteria:

1. The primary flight path control system must be capable of achieving
   a. a descent angle of 2° steeper than the nominal glide slope angle at a speed 10 knots above the reference approach speed.
   b. a descent angle of 2° less than the nominal glide slope angle at a speed of 10 knots below the reference approach speed and in the design headwind.
   c. an incremental flight path angle above the nominal of

$$\Delta\gamma = 20 \times \left(\frac{\partial\gamma}{\partial V}\right)_{\text{TRIM FOR APPROACH}} \quad - \begin{cases} \gamma \text{ - degrees} \\ V \text{ - knots} \end{cases}$$

2. The maximum climb rate available at full power at the reference approach speed with one engine failed must be at least 250 ft/min in a 3°/sec steady turn maneuver. A configuration change is allowable to meet this requirement as long as it is easily made with the same motion that is required to increase thrust, and provided no noticeable loss of lift or appreciable change in control forces results.
3. The maximum rate of sink below 1000 feet altitude shall be less than 1000 ft/min.

Discussion:

1. The control requirements for the primary flight path control system are discussed in detail in Section 5.4.1.2 and were justified by particular tests conducted in the present study. It is worth noting that the first requirement listed effectively picks the value of the flight-idle "stop" referred to in Section 3.2.6.
2. The maximum climb rate requirement evolved from the engine-out

controllability and go-around tests that were conducted on the EBF airplane. Due to the large rolling moments generated by engine failure for this type of airplane, the design engine-out rolling moment is a direct function of the maximum installed T/W. Thus a series of tests were conducted to determine the minimum engine-out climb gradient (and hence the minimum T/W) felt to be acceptable to the pilot. These tests included clearing turns during the go-around climb-out, for which a turn rate of 3°/sec was considered to be adequate. In this clearing turn it was found that the variations in climb rate due to pilot technique and airplane dynamics resulted in portions of the turn occurring with the airplane descending rather than climbing unless the average climb rate capability in the turn was greater than 250 ft/min. It was therefore considered that this was a minimum value of acceptable one-engine climb rate.

3. Many times during evaluations of flight path control requirements pilots commented on the excessive rates of descent needed to capture the glide slope from an initial condition above the reference speed and above the ILS beam. The nominal glide slope for these tests was at -6° and the reference approach speed was 80 knots. Nominal rate of descent was therefore 850 ft/min. This increases to 955 ft/min at 10 knots above reference, and 1275 ft/min at increased speed and on a 2° steeper glide path. There are many documented tests suggesting a limiting value on acceptable rate of descent near the ground, the most often quoted value being 1000 ft/min. The present tests amplified the need for such a limitation and discussions with the evaluating pilots brought forward the following guidelines for limitations for manual flight:

a. The nominal stabilized glide slope should represent a rate of descent of between 800 and 1000 ft/min.

b. Below 1000 feet, the maximum rate of descent needed in maneuvering should be less than 1000 ft/min if at all possible.

c. Above 1000 feet, rates of descent as high as 1500-2000 ft/min are acceptable.

d. The airplane should be stabilized on the final descent slope before reaching 400' altitude or at least 45 seconds from touchdown.

e. In the descent, the body attitude should be within two degrees of the touchdown attitude and power levels must be set high enough to give good engine acceleration characteristics.

Since some of these ideas are not fully tested against the STOL approach requirements, the criteria chosen was the simple restriction on maximum sink rate.

3.4 Speed Margins

Criteria: The choice of reference speed and approach configuration shall provide a margin of 15% in speed from the demonstrated V_{MIN} at the approach power setting.

V_{MIN} is defined as the lowest obtainable speed in 1 g flight that does not suffer uncontrollable roll, yaw or pitch motions, intolerable buffet, or exceed the maximum allowable angle-of-attack.

Discussion: This requirement is comparable to the familiar speed margin of conventional jet airplanes, and is taken from references 5 & 6 since no specific tests to determine the necessity of this margin were completed in the present study. It is not expected to be more limiting than the normal acceleration requirement at approach power of Section 3.2 which provides a lift margin for the critical maneuver cases.

It is worth noting that the requirements on minimum control speed, V_{MCA}, in Section 7.1.6 can also set a speed margin requirement for the approach reference speed if the airplane is rudder power or lateral control power limited.

Other speed margins suggested by various investigators have been keyed to minimum speeds defined by the capability to maintain the reference flight path angle at full power. The purpose of such a margin would appear to be to ensure the capability to return to the original glide slope after a disturbance due to gusts or wind shear or some inadvertent speed change. The flight path margins of Section 3.3. cover this since the requirement for $\Delta\gamma \geqslant 20 \cdot \frac{d\gamma}{dV}$ was determined to cover just this very case. The speed margin from $\gamma = 0$ at full thrust for the airplanes used in this study was between 25 and 30 knots all engines operating, and less than 5 knots with one engine failed.

No requirement has been generated for a speed margin from the power-off V_{MIN}. Other discussions in this chapter have expressed the need for a flight-idle stop on the engines of powered-lift vehicles with the express intent of avoiding inadvertent buffet or stall due to idle power selection by the pilot. The trimmed flight path angle at 80 knots and 25% power is about 10° - 11° giving a descent rate of 1500 ft/min, well above the usable values quoted in Section 3.3. Once the flight-idle stop is installed, of course, the question still arises in the form "should there be a speed margin from the V_{MIN} at flight-idle power". No requirement has been found for such a margin in this study, but the question deserves more study. A flight-idle stop set to the flight path control requirements of Sections 5.4.1.2 and 3.3 proved to be satisfactory in the present study for all the transition maneuvers accomplished, even in gusts up to r.m.s. levels of 6.5 ft/sec. Calculation of the speed margin available for the AW and EBF airplanes at this stop setting shows it to be of the order of 5 to 10 knots.

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3.5 Touchdown Dispersions

No criteria were determined in this area.

Discussion:

A statistical analysis of touchdown point and sink rate at touchdown has been made for the following configurations:

Aircraft	SAS Configuration	Piloting Technique
AW	Full SAS, poor n_z response (τ_{n_z} = 2.0 Sec.)	Flare with column
AW	Full SAS, improved n_z response (τ_{n_z} = 0.6 sec.)	Flare with column
AW	Minimum SAS	Flare with throttle
EBF	Minimum SAS	Flare with throttle

Horizontal error and sink rate error with respect to the glide slope at an altitude of 50 feet were also analyzed for the first two AW configurations.

The target sink rate at touchdown was 6 ft/sec. The target touchdown point was 600 ft. from the threshold on a simulated 2250 ft. runway.

Two experimental test pilots flew the runs analyzed over a period of several days. The pilots were not initially familiar with the configurations flown. Therefore, the pilots' learning curves influence these results and statistical methods must be used for evaluation. Most of the runs were primarily intended to evaluate aircraft handling qualities in the landing approach and flare. Precision landings were considered to be of secondary importance. Vertical "S" maneuvers or speed changes during the approach were used in the majority of runs. A few runs in cross winds and turbulence are included. Most of the runs were performed in still air with unlimited visibility. No engine failures are included. The pilots did perform go-arounds if it appeared that a reasonable touchdown could not be made.

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The maneuvers during the approach were larger and of a different type than a pilot would normally use. However, they are similar to tasks such as capturing the glide slope or flying in a gusty environment. These evaluation maneuvers were terminated before the flare. All of the landings were visual although several runs simulated breaking out of an overcast at various altitudes. Altitude and raw ILS data were available in the simulator cockpit. It is possible that performance could be improved by the use of more elaborate landing aids.

For the reasons stated, it is felt that the data are not adequate to determine design field lengths for the aircraft configurations studied. The data are valuable for the following uses:

- o They give a qualitative evaluation of the performance of the various configurations in the flare and touchdown.
- o They will serve as a basis to design experiments to determine field lengths rationally.
- o They suggest relationships between easily measured flight path parameters and touchdown dispersions which can be used to define approach and landing aids and to set flight path boundaries for satisfactory landings.

3.5.1 Statistical Analysis

Some simple statistical analyses have been caried out on the available data in order to determine the effects of the various configurations on landing performance. The mean values and the standard deviations of touchdown distance, sink rate at touchdown and the aircraft relationship to the glide path at 50 ft. are shown in Table 3.5-1.

The standard deviation is a measure of the amount of scatter of the data about the mean. Statistical tests can be used to determine if the difference between two means or standard deviations are significant in a statistical sense.

With a large number of available data points, the mean value of the touchdown point and the distribution of the touchdown distances about the mean could be precisely determined. As the number of data points is reduced the effect of a single point becomes more important in determining the mean and the standard deviation of the set.

With the knowledge of the number of data points in a set of data, an estimate of the accuracy with which the mean and the standard deviation is known can be made. It can then be determined if there is sufficient data to give statistically significant differences between calculated mean values and standard deviations.

Tests of significance have been made for the following combinations and the results are shown in Table 3.5-2.

- AW aircraft, full SAS, effect of load factor response
- AW aircraft, full SAS and simple SAS, effect of piloting technique
- Simple SAS, AW and EBF aircraft, effect of airplane configuration.

The statistically significant results are further analyzed in the subsequent sections of this chapter.

Touchdown distances and sink rates at touchdown for the configurations tested are shown in Figure 3.5-1. Data from Reference 13 for the C-141 aircraft with an all-weather landing system are shown for comparison. The C-141 data were obtained for a conventional glide slope approach.

REV SYM

STATISTICAL PERFORMANCE SUMMARY

PARAMETER	Mean, $\bar{x} = \frac{1}{n}\sum_{i=1}^{n} x_i$				Standard Deviation: $\Delta_x = \sqrt{\left\{\frac{n\sum x_i^2 - (\sum x_i)^2}{n(n-1)}\right\}}$			
	AW Aircraft Full SAS		Minimum SAS*		AW Aircraft Full SAS		Minimum SAS*	
	n_z = 2.0 Sec.	n_z = 0.6 Sec.	AW Aircraft	EBF Aircraft	n_z = 2.0 Sec.	n_z = 0.6 Sec.	AW Aircraft	EBF Aircraft
1) Touchdown distance (from threshold), X_{TD}, ft.	680	695	630	536	316	165	262.5	193.5
2) Sink rate @ touchdown, $\dot{H}_{TD}$, ft/sec	-6.7	-4.9	-7.45	-7.6	2.37	1.97	2.95	2.67
3) Horizontal glide-slope error @ 50', $\Delta X_{50'}$, ft.	-66	0.8	-	-	84	109	-	-
4) Sink rate error @ 50', $\Delta \dot{H}_{50'}$, ft/sec	1.6	2.75	-	-	1.66	1.88	-	-

*Note: The minimum SAS required an unconventional piloting technique.

TABLE 3.5-1

REV SYM

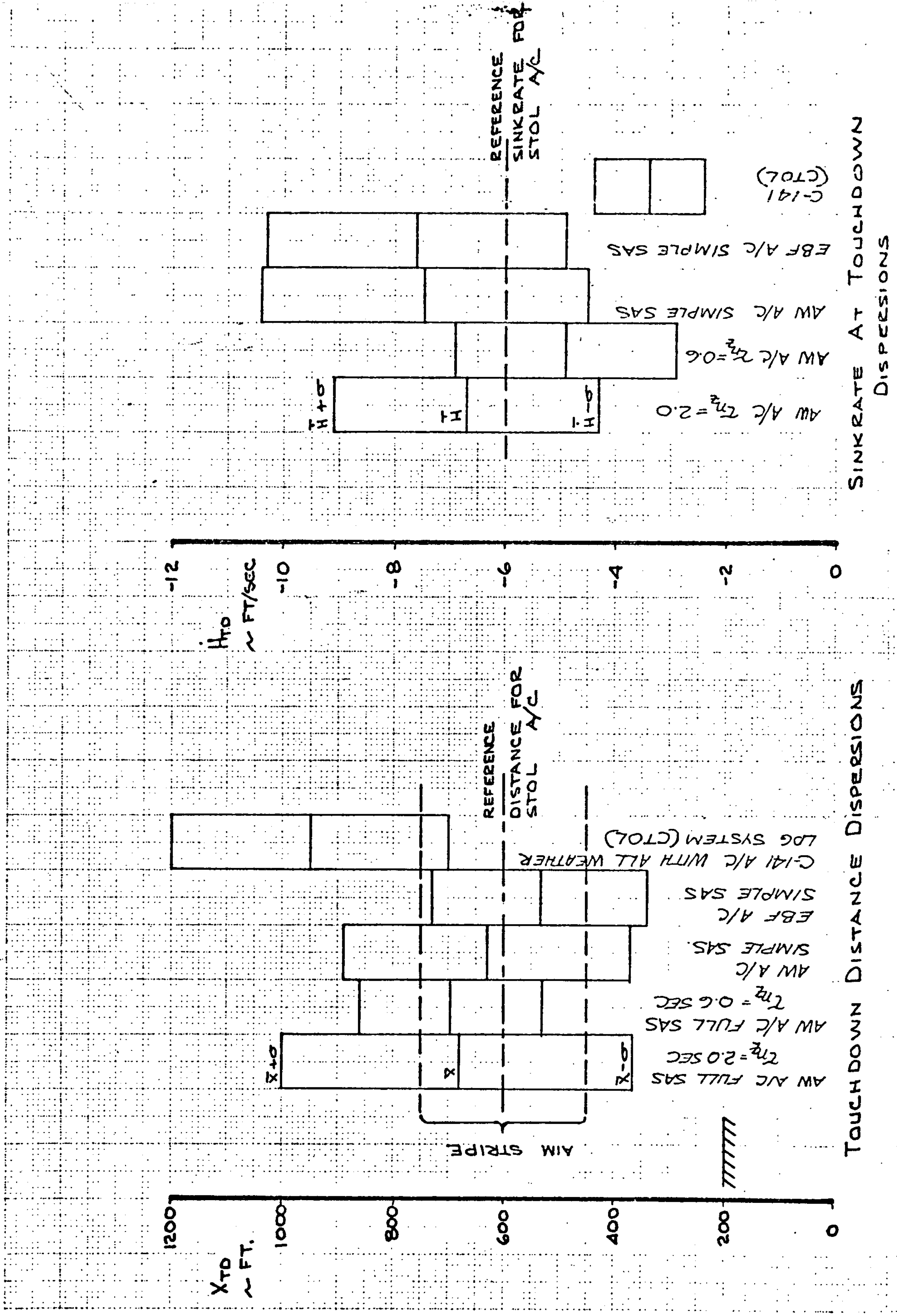
TESTS OF SIGNIFICANCE

PARAMETER		CONFIGURATION: AW A/C, Full SAS τ_{n_z} = 2. Sec. vs. τ_{n_z} = 0.6 sec.	CONFIGURATION: Minimum SAS, AW vs. EBF	CONFIGURATION: AW A/C Full SAS (τ_{n_z}=0.6 Sec) Vs. Minimum SAS
X_{TD}:	Mean	O	✓	O
	Std Deviation	✓	O	✓
$\dot{H}_{TD}$:	Mean	✓	O	✓
	Std Deviation	O	O	✓
$\Delta X_{50'}$:	Mean	✓		
	Std Deviation	O		
$\Delta \dot{H}_{50'}$:	Mean	✓		
	Std Deviation	O		

TABLE 3.5-2

O - Difference not statistically significant.

✓ - Difference is significant



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DISPERSIONS OF TOUCHDOWN DISTANCES AND SINK RATE AT TOUCHDOWN

D6-40409

FIG 3.5-1

THE BOEING COMPANY

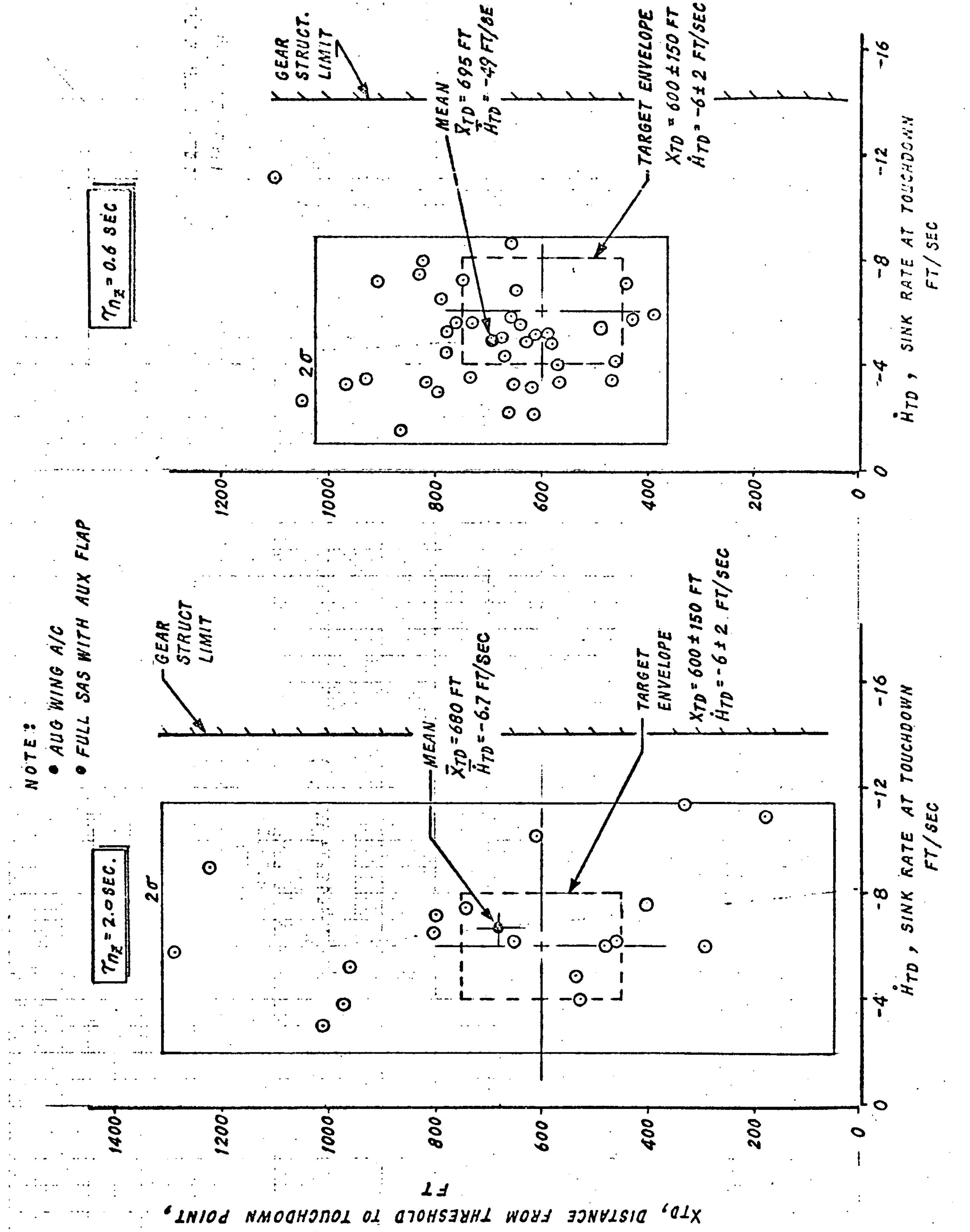
3.5.2 Effect of Load Factor Response on Touchdown Characteristics

The AW aircraft configuration with full SAS was used to evaluate the effects of varying load factor response. Conventional piloting techniques, (flare with column) were used. SAS gains were varied to obtain equivalent time constants for the load factor response, τ_{n_z}, of 2 sec. and 0.6 sec. These equivalent time constants were measured from responses to a column pulse, see Section 5.4.1. Data are presented in Figures 3.5-1 and 3.5-2, and Table 3.5-1 to show the effects of load factor response characteristics.

The data show a significant reduction in the touchdown dispersion (as measured by the standard deviation) and the mean sink rate at touchdown for the more responsive (τ_{n_z} = 0.6 sec.) aircraft. The differences between the mean touchdown distances and the sink rate dispersions for the two configurations were not statistically significant.

There is no significant linear correlation between the touchdown distance, X_{TD}, and the sink rate at touchdown, $\dot{H}_{TD}$, for either configuration. None of the sink rates at touchdown exceeded the assumed gear limits of -14 ft/sec. The mean sink rates of -6.7 and -4.9 ft/sec., for τ_{n_z} = 2 and 0.6 sec., respectively, were satisfactory. Also the mean touchdown distances of 680 ft. and 695 ft. are not statistically different although the dispersions about the mean for the less responsive aircraft are significantly greater. This suggests that the fine scale controllability of both aircraft was sufficient, but the long term control of flight path and precision control of the flare was more difficult with the τ_{n_z} = 2 sec. configuration. This configuration was more sensitive to flare initiation height and the initial conditions when the flare was begun. This suggests that this configuration would also be more affected by gusts and turbulence.

The responsive (τ_{n_z} = 0.6 sec.) system gave the pilot sufficient control during approach and flare to touch down well past the threshold, yet choose the point consistently enough to stop the aircraft within the required distance. With the stopping characteristics used in the simulation, the aircraft could stop in about 1100 feet from the touchdown point.



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TOUCHDOWN DISPERSION COMPARISON WITH SINK RATE AT TOUCHDOWN, τ_{n_z} = 0.6 & 2.0 SEC

D6-40409

FIG 3.5-2

3.5.3 Effect of Piloting Technique on Touchdown Characteristics

The AW aircraft was also tested with a less complex longitudinal SAS. This "minimum" SAS did not use an auxiliary flap control to aid in decoupling thrust and lift and required a different piloting technique. The pilot controlled speed by changing aircraft attitude with column. He controlled rate of sink with the throttles. This technique is significantly different from that used in large jet aircraft and required several runs for the pilots to feel sure of themselves. The SAS was used to damp the pitch response and to quicken the engine characteristics to give good load factor response to throttle commands.

Accurate and repeatable flares were difficult using this unconventional technique as demonstrated by the higher sink rates at touchdown shown in Figures 3.5-1 and 3.5-3. Because of the learning process involved it is difficult to say if the larger dispersions in touchdown distances compared to the more responsive, conventional aircraft, are meaningful. It does appear that the pilot tended to touchdown harder with a larger dispersion with the simple SAS using the unconventional technique than he did with the conventional technique using the more responsive SAS.

With further training and improved flight path information in the cockpit it would be possible to improve landing performance and to produce good, consistent landings using this technique.

3.5.4 Comparison of EBF & AW Configurations

Figure 3.5-3 also compares the AW and EBF configurations both flown with the minimum SAS mode. Both configurations had essentially the same mean sink rates at touchdown and the same dispersion about the mean. The lower touchdown distance dispersions obtained with the EBF aircraft may be due to pilot experience. By the time the pilots flew the EBF configuration they had gained experience flying the AW in the unconventional mode for two days.

3.5.5 Comparison of Touchdown Dispersions With Flight Path Parameters at 50 Feet

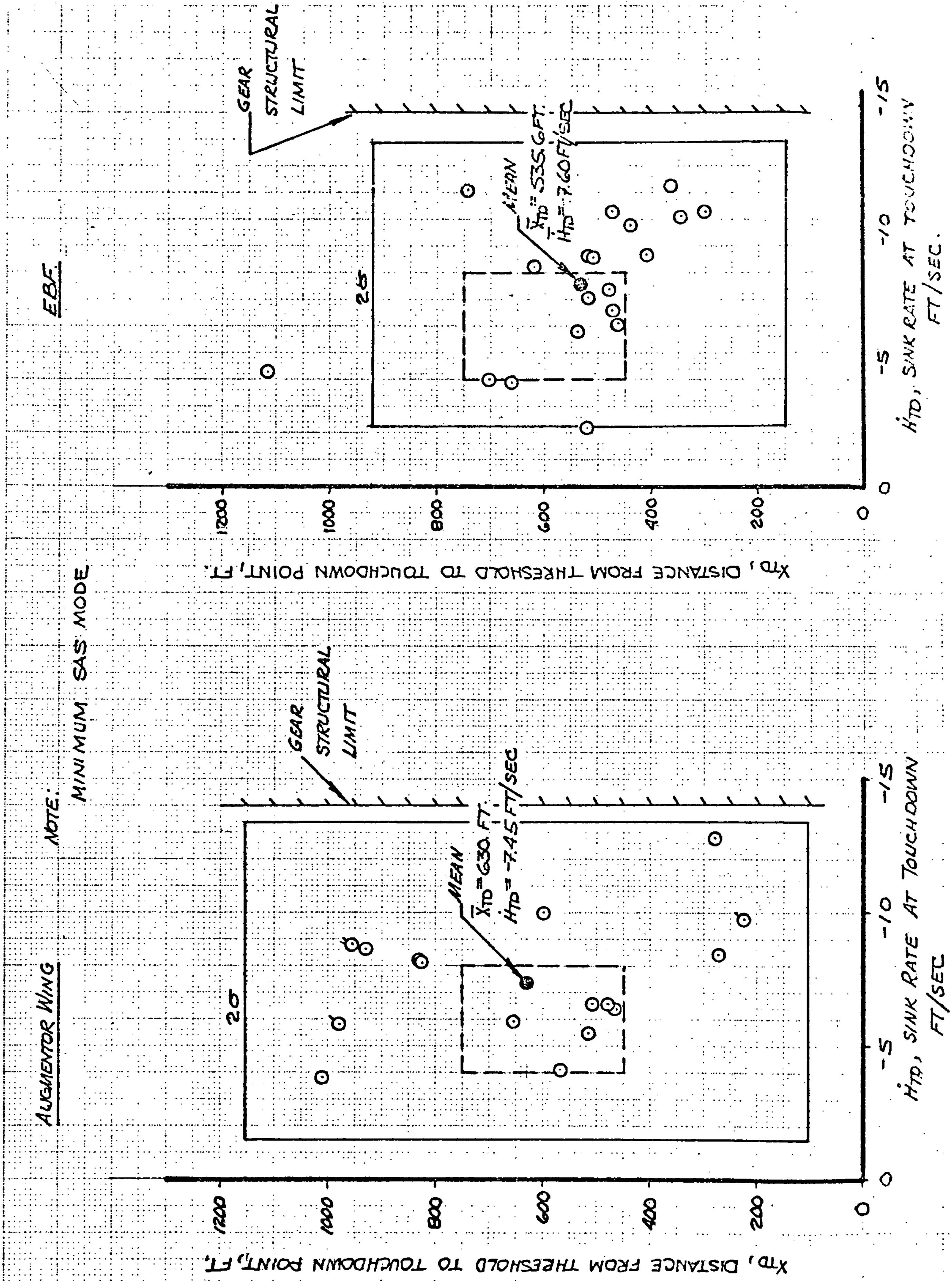
Figures 3.5-4 through 3.5-7 compare the distance of the touchdown point from the aim point (ΔX_{TD}, ft.) with sink rate error ($\Delta \dot{H}_{50'}$, ft/sec) and horizontal flight path error ($\Delta X_{50'}$, ft.) measured with respect to the glide path when the aircraft passed through 50 feet altitude.

The computer used in the simulation stored various flight path parameters when the aircraft passed through 50 feet altitude during the approach. These parameters were recorded and used to calculate the relation of the aircraft to the glide path at this point. The flare was generally initiated below 50 feet altitude for the more responsive aircraft and at about 50 feet for the aircraft with τ_{n_z} = 2.0 sec. Analysis (see Table 3.5-2) shows that the differences in the dispersions of $\Delta X_{50'}$ and $\Delta \dot{H}_{50'}$ (see Figures 3.5-4 through -7 for definitions of these terms) for τ_{n_z} = 2.0 sec and τ_{n_z} = 0.6 sec. is not significant. This implies that the pilots could position the aircraft relative to the glide path prior to the flare equally well for either configuration. The significant difference between the mean values of $\Delta X_{50'}$ and $\Delta \dot{H}_{50'}$ for the two configurations indicates that the pilots learned that these initial conditions helped compensate for shortcomings in the flare capability of the aircraft. Figure 3.5-4 shows that the pilots tended to be right on the glide path at 50 feet when flying the more responsive aircraft. With the less responsive aircraft they tended to be below the glide path as shown in Figure 3.5-5.

The large difference in touchdown dispersions for the two configurations is significant. It indicates that although the pilots could set up initial conditions to their liking, the aircraft with the sluggish n_z response still had large touchdown dispersions. Further investigation of Figures 3.5-4 and 3.5-5 shows that the pilot could cut the touchdown dispersions somewhat if he kept the aircraft within ±2 dots using the raw ILS data with the τ_{n_z} = 0.6sec. configuration. It does not appear that the touchdown dispersions of the sluggish configuration would be much improved if this criterion were met.

Statistical analysis indicates that there is significant linear correlation between $\Delta X_{50'}$ and ΔX_{TD} for the τ_{n_z} = 0.6 sec. configuration (99.6% confidence level). The linear correlation between $\Delta X_{50'}$ and ΔX_{TD} is less (95.5% confidence level) for the τ_{n_z} = 2. sec. aircraft. It appears that about 17% of the touchdown distance error may be attributed to the horizontal glide slope error in both cases.

There does not appear to be any correlation between sink rate error at 50 feet and touchdown distance, as shown in Figures 3.5-6 and 3.5-7. This implies that as long as the aircraft has sufficient flare capability the sink rate at the beginning of the flare is unimportant. However, the aircraft location at the beginning of the flare still affects the touchdown point. Better landing aids would enable the pilot to position the aircraft more precisely and improve touchdown dispersions.



CALC	MACK	JUNE 1972	REVISED	DATE	TOUCHDOWN DISPERSION COMPARISON WITH SINK RATE AT TOUCHDOWN, EBF & AUG. WING A/C MINIMUM SAS	D6-40409 FIG. 3.5-3
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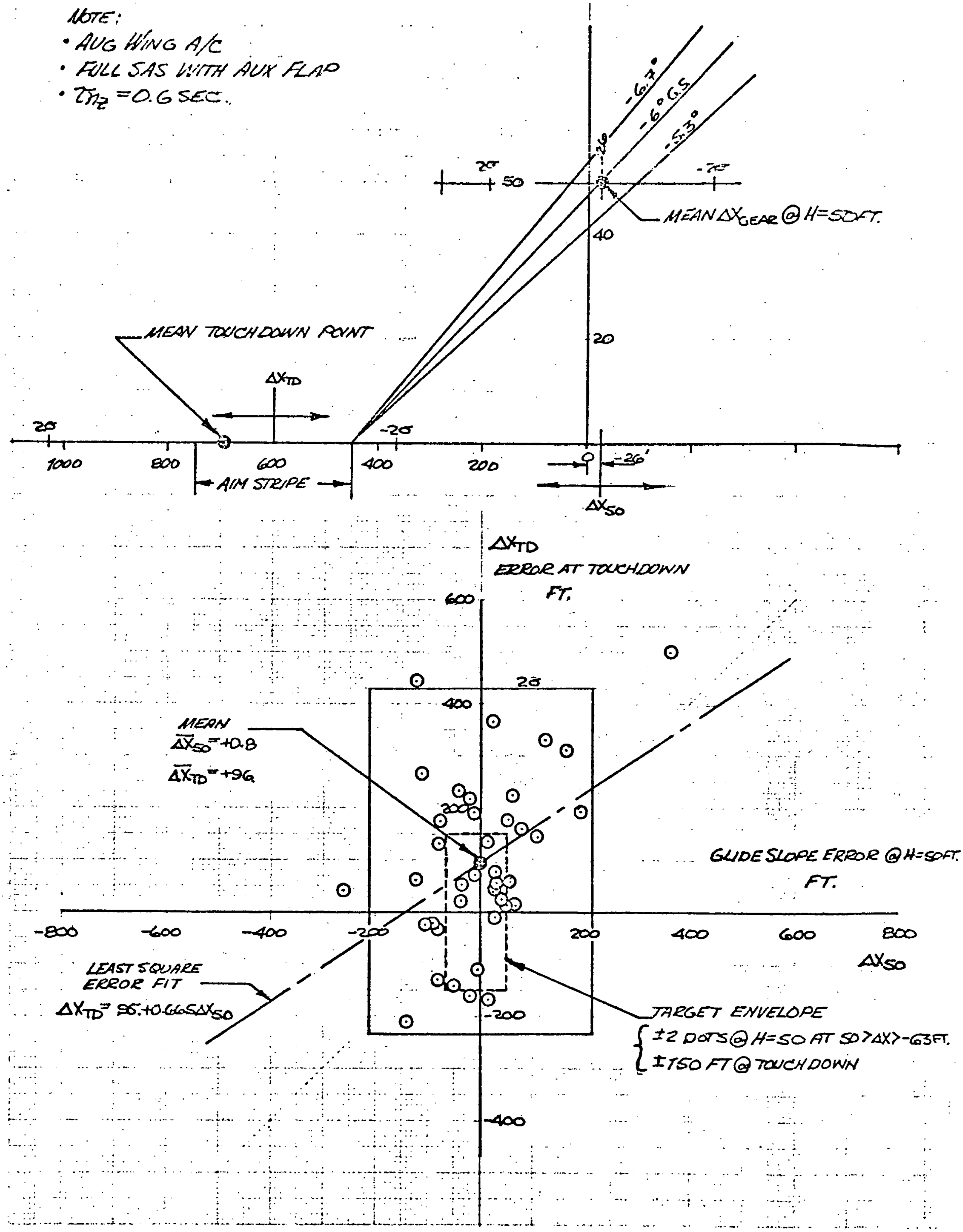
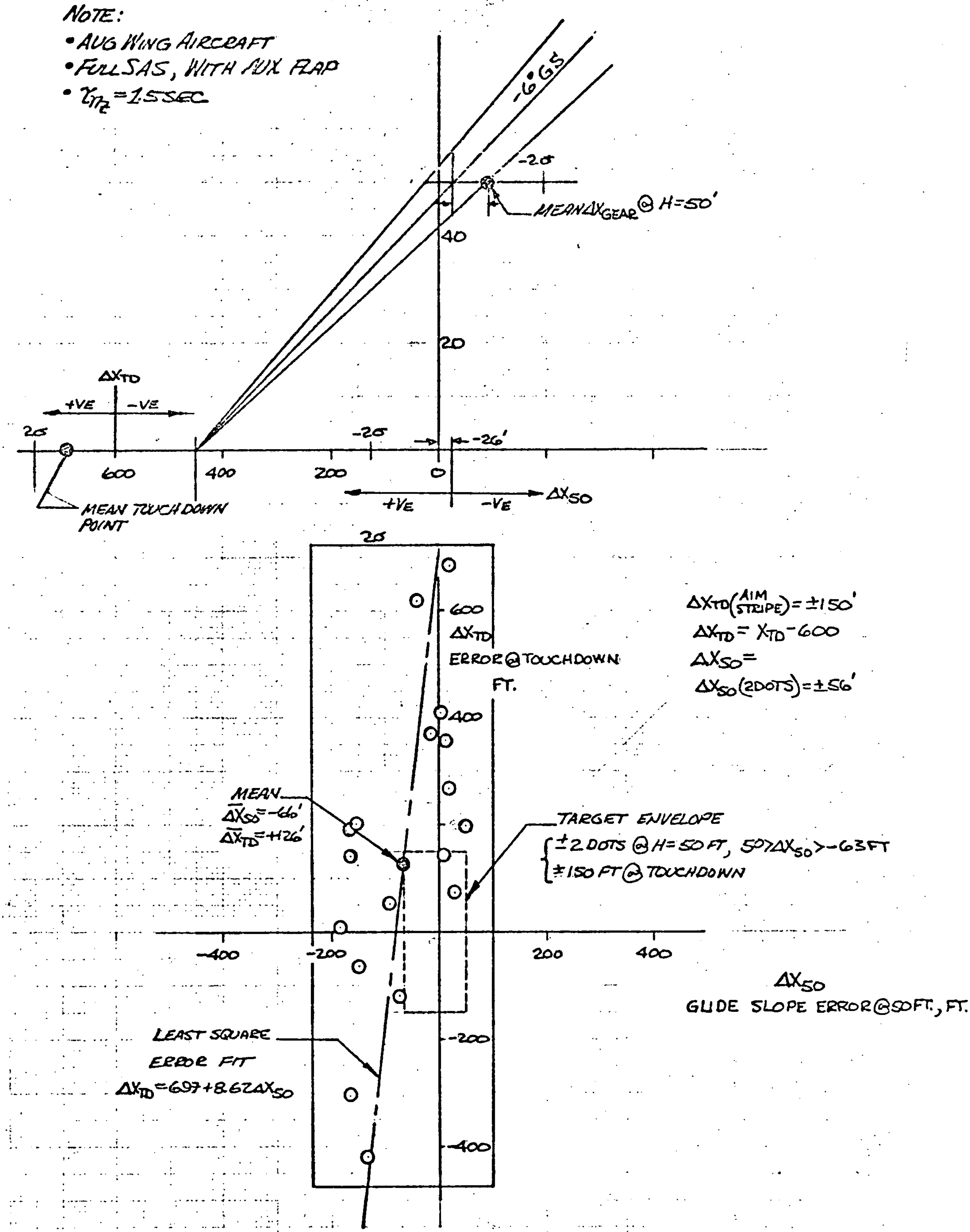
APR					THE BOEING COMPANY	67



CALC	MACK 8 DEC '71	REVISED	DATE	TOUCHDOWN DISPERSION COMPARISON WITH GLIDE SLOPE ERROR @ 50 FT. ALTITUDE τ_{η_z} = 0.6 SEC	D6-40409
CHECK					FIG 3.5-4
APR					
APR				THE BOEING COMPANY	



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TOUCHDOWN DISPERSION COMPARISON WITH GLIDE SLOPE ERROR @ 50FT. ALTITUDE
τ_{η_z} = 2.0 SEC

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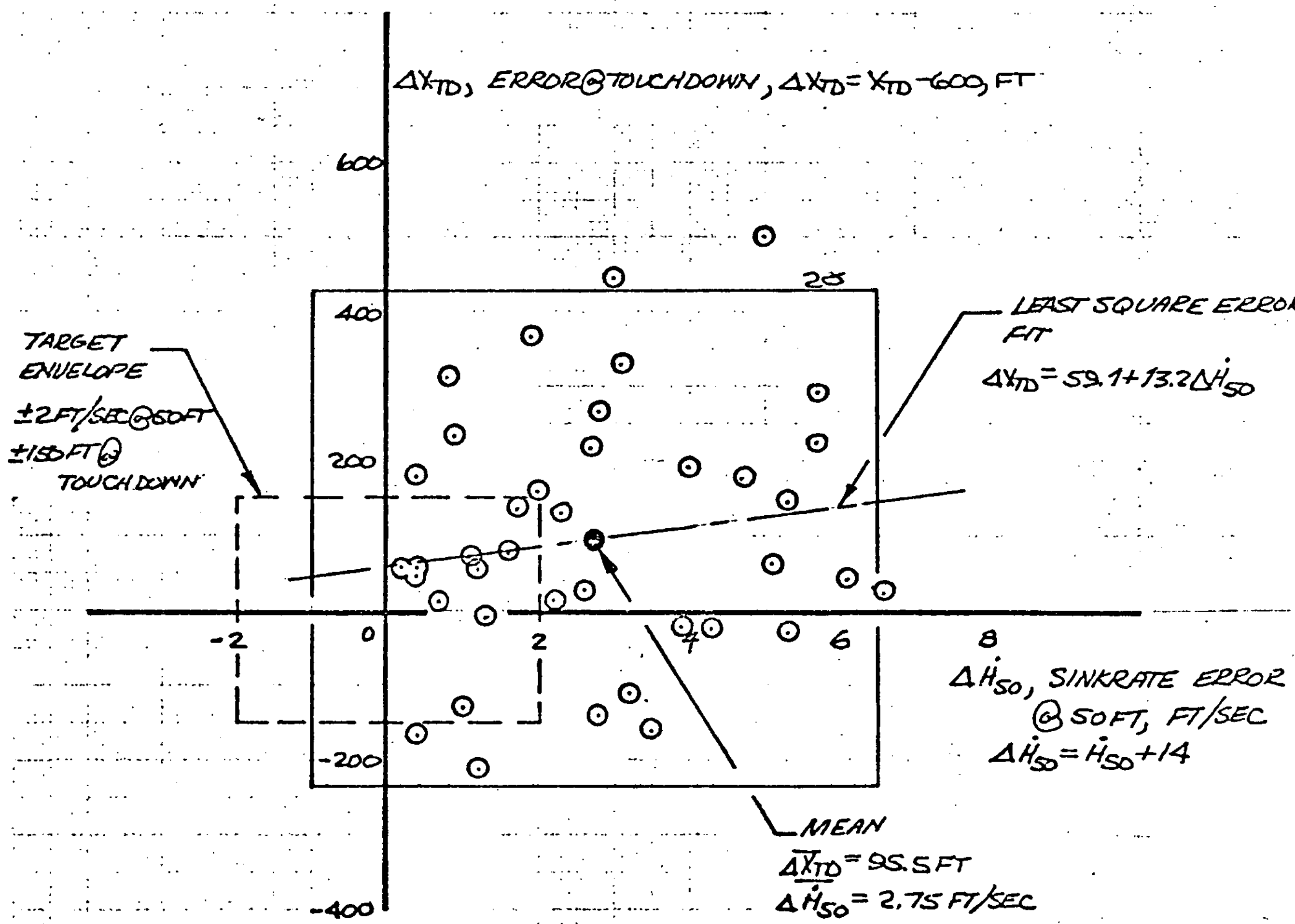
FIG 3.5-5

NOTE:
- AUG WING A/C
- FULL SAS WITH AUX FLAP
- τ_{n_z} = 0.6 SEC



CALC	MACK 9 DEC '71		DATE	TOUCHDOWN DISPERSION COMPARISON WITH SINK RATE ERROR @ 50 FT τ_{n_z} = 0.6 SEC	D6-40409
CHECK					FIG 3.5-6
APR				THE BOEING COMPANY	
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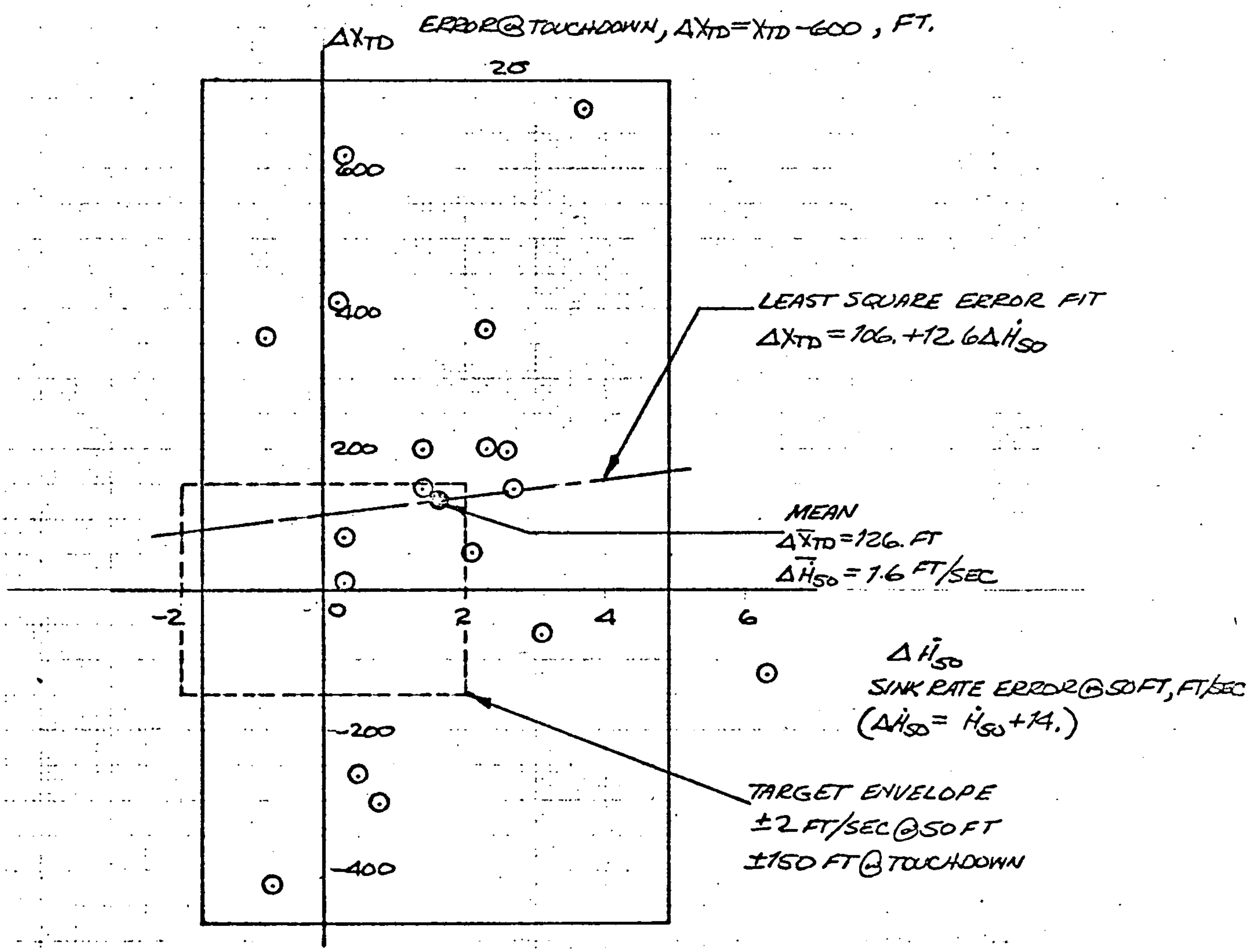
NOTE:
- AUG WING A/B
- FULL SAS WITH AUX FLAP
- $\tau_{n_z} = 1.5$ SEC.



CALC Mack 9 DEC '71	TOUCHDOWN DISPERSION COMPARISON WITH SINK RATE ERROR AT 50 FT. $\tau_{n_z} = 2.0$ SEC	D6-40409
CHECK		FIG 3.5-7
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3.6 Landing Distances

Criteria

The landing field length for the STOL airplane shall be determined in a rational manner to include touchdown dispersions and braking distance dispersions which account for the following events:

(1) Turbulence up to the maximum design value

(2) Wind shears up to the maximum design value

(3) Engine failure at the worst altitude during the approach

(4) Crosswinds up to the maximum design value.

Sufficient hard runway surface will be provided for underruns and overruns to cater for the worst combinations of the above cases.

Discussion

In the present study, insufficient data were generated to define the individual effects of environmental conditions or engine failures on landing dispersions. However, the results which were taken give a general idea of the sensitivity of field length to airplane handling qualities. A calculation of the braking sequence after touchdown was included in the simulation and this produced a realistic visual scene of the derotation, a good aural simulation of the thrust reversers, and some impressive acceleration cues from the motion system which simulated sustained longitudinal accelerations with cab pitch angle. The calculated distances to stop were a function of touchdown speed, the braking assumptions being $\mu_{MAX} = .2$ plus about 30-40% thrust reversing.

The resulting roll-out distances were nominally 1100 feet, varying about ±5% due to touchdown speed variations. Adding these to the touchdown dispersions discussed in Section 3.5 gives the total ground run variations for individual airplanes and for various augmentation system designs. Figure 3.6-1 compares the 3σ dispersions of touchdown distance and stopping distance for the two SAS configurations tested on the augmentor wing airplane. The interpretation of the 3σ data must be approached with care since the data points are not distributed normally and the number of points is fairly small. However, for use in determining SAS effects on airplane field length capability, the 3σ data lends some scale to the comparison. The less responsive airplane is obviously going to need a larger runway.

Interpreting these data in terms of field lengths may be misleading as far as absolute numbers go, but Figure 3.6-2 shows the comparison in order to point out the obvious relationship between the aiming point to threshold geometry and the touchdown dispersion capability of the airplane. Also apparent from this figure is the need for more such data to define whether the 2000 foot field is really a possibility.

Figure 3.6-3 shows the frequency distribution and cumulative distribution of the 39 landings accomplished with the responsive SAS on the augmentor wing airplane. A normal distribution line is shown for comparison. A good fit to the experimental points appears to be a logarithmic normal distribution

$$N = \frac{1}{\sqrt{2\pi}\sigma} \exp\left\{ \frac{-(\ln X_{TD} - 600)^2}{2\sigma^2} \right\}$$

where σ = standard deviation of the touchdown pts.

Figure 3.6-4 compares this cumulative frequency distribution with data taken for the 727 airplane landing on 5000 foot fields and data taken by the FAA for large 4-engined jet transports landing on 10,000 foot runways.

In conclusion, the data obtained during the NASA Ames simulation are not adequate to define field length requirements for the configurations tested. However, they do give a basis of comparison for further tests and for analytical evaluation of the effects of configuration changes. These data can be used as a basis to design experiments to determine rational field lengths.

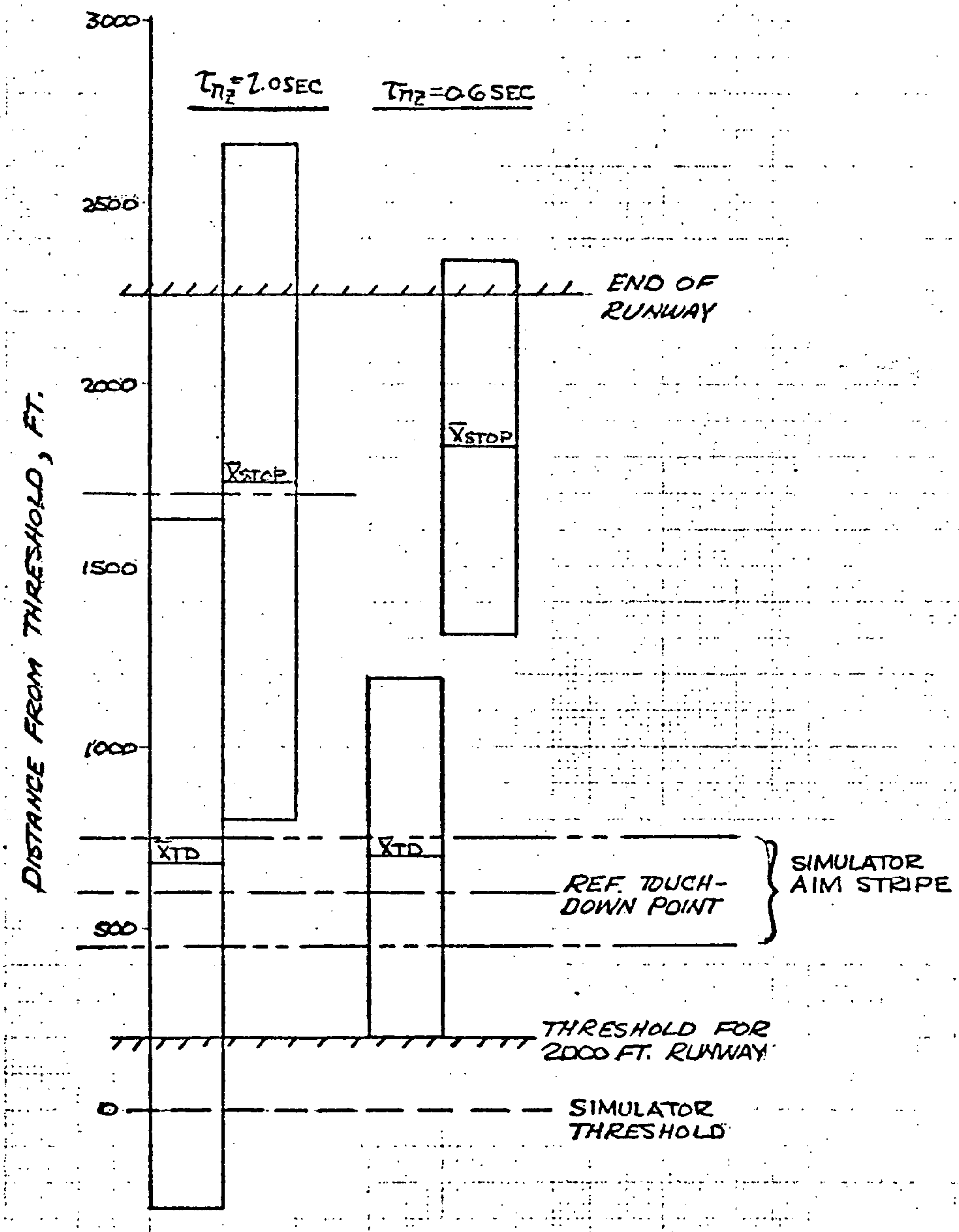
The data show that:

(1) The load factor response characteristics will influence touchdown dispersions and gear design.

(2) Touchdown characteristics can probably be improved with better landing aids.

(3) The simulator can be used to determine field length parameters if the experiments are carefully designed for this purpose.

NOTE:
- NASA AMES FSAA MOVING BASE SIMULATOR
- AUGMENTOR WING AIRCRAFT (751-100)
- FULL SAS WITH AUX. FLAP
- AUTOMATIC STOPPING ROUTINE
- $\gamma_{APP} = -6°$, $V_{APP} = 80$ KNOTS
- 3σ DISTRIBUTIONS ABOUT MEAN
- NO ENGINE FAILURES INCLUDED
- $\mu = 0.2$



MACK 7 DEC. '71	DATE	EFFECT OF HANDLING QUALITIES ON TOUCHDOWN AND ROLL-OUT DISTANCE DISPERSIONS.	D6-40409
CHECK			FIG. 3.6-1
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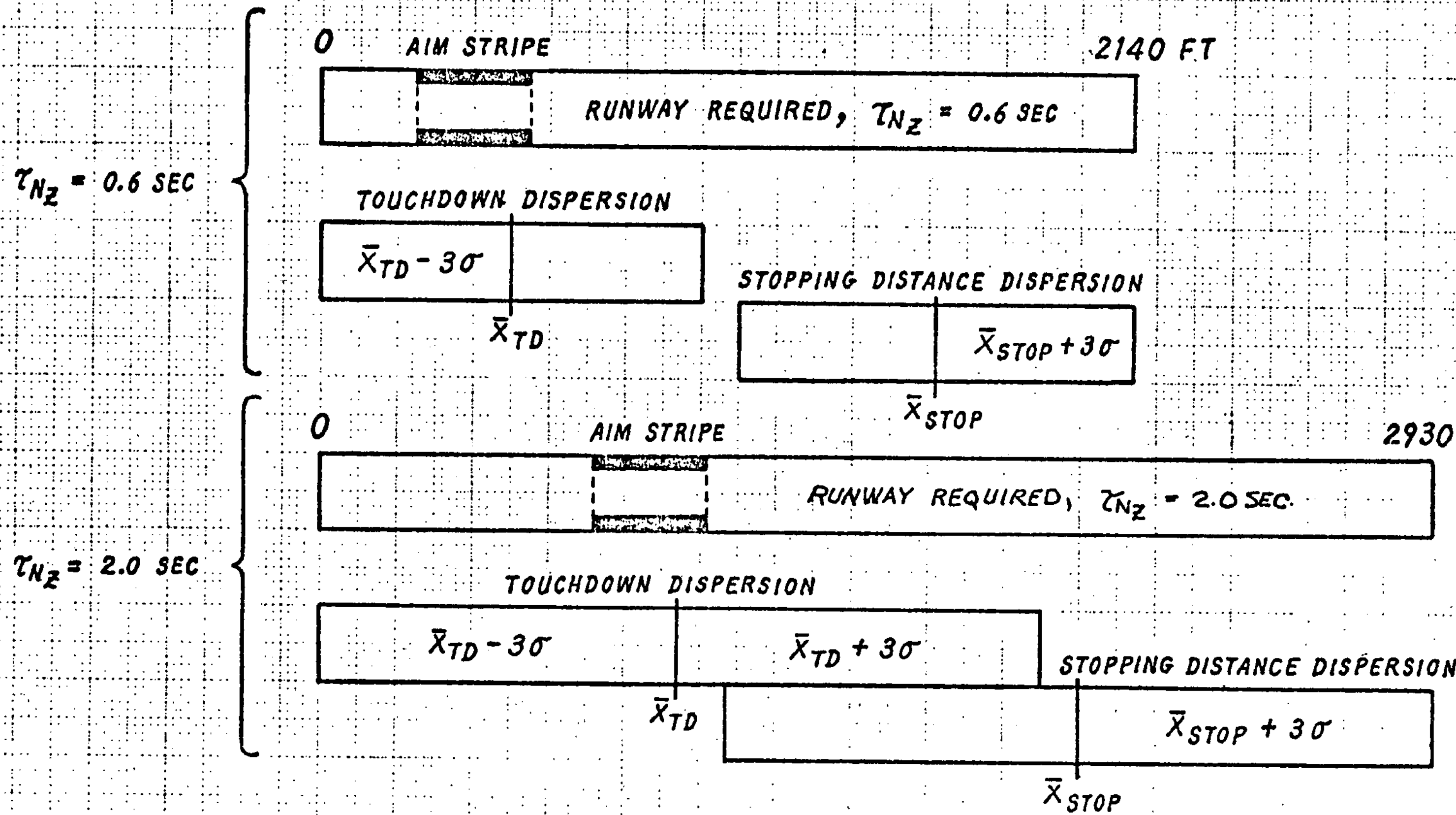
APR		THE BOEING COMPANY	



AUGMENTOR WING AIRCRAFT, FULL SAS

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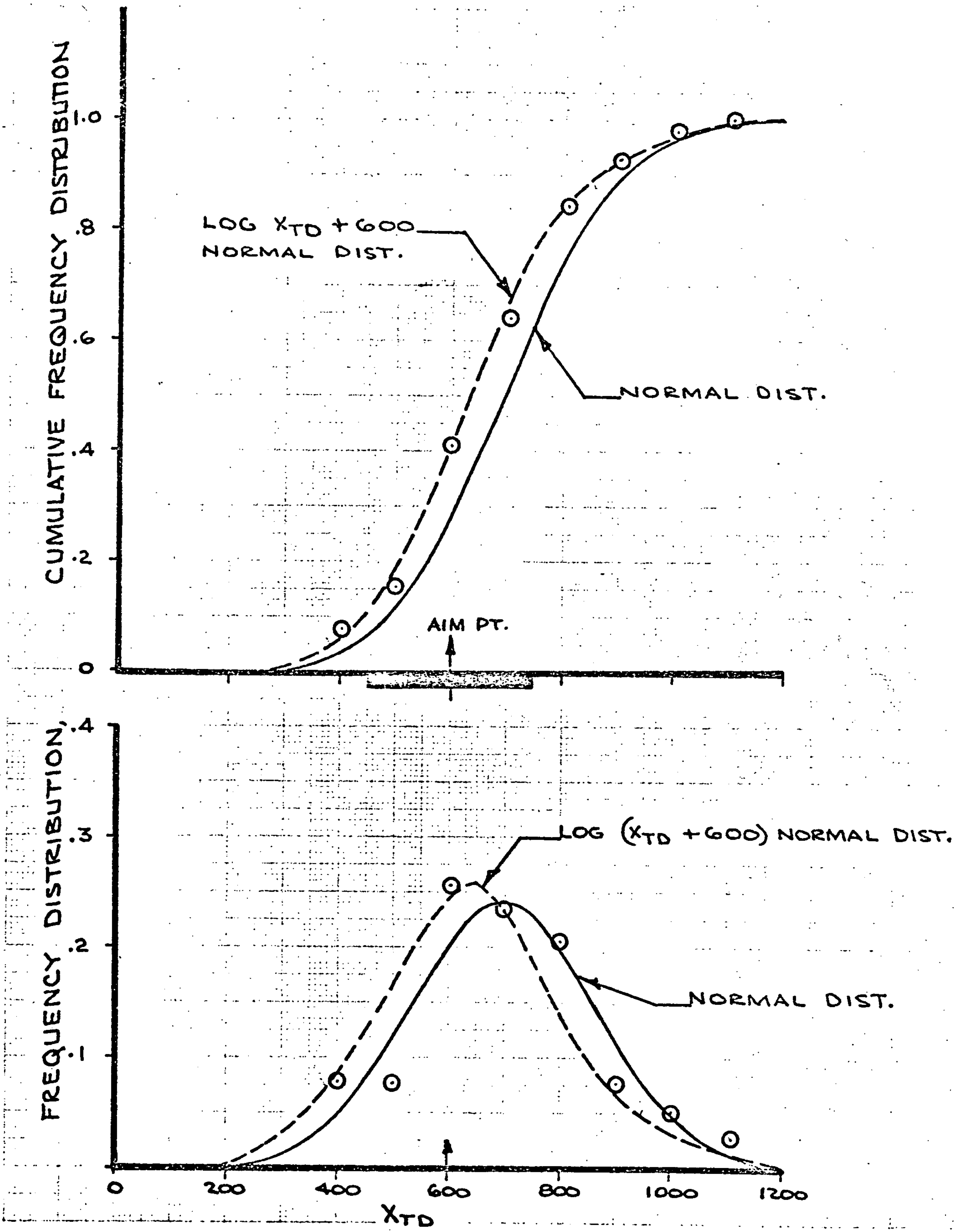
REQUIRED RUNWAY LENGTH
EFFECT OF LOAD FACTOR
RESPONSE CHARACTERISTICS

THE BOEING COMPANY

D6-40409

FIG. 3.6-2



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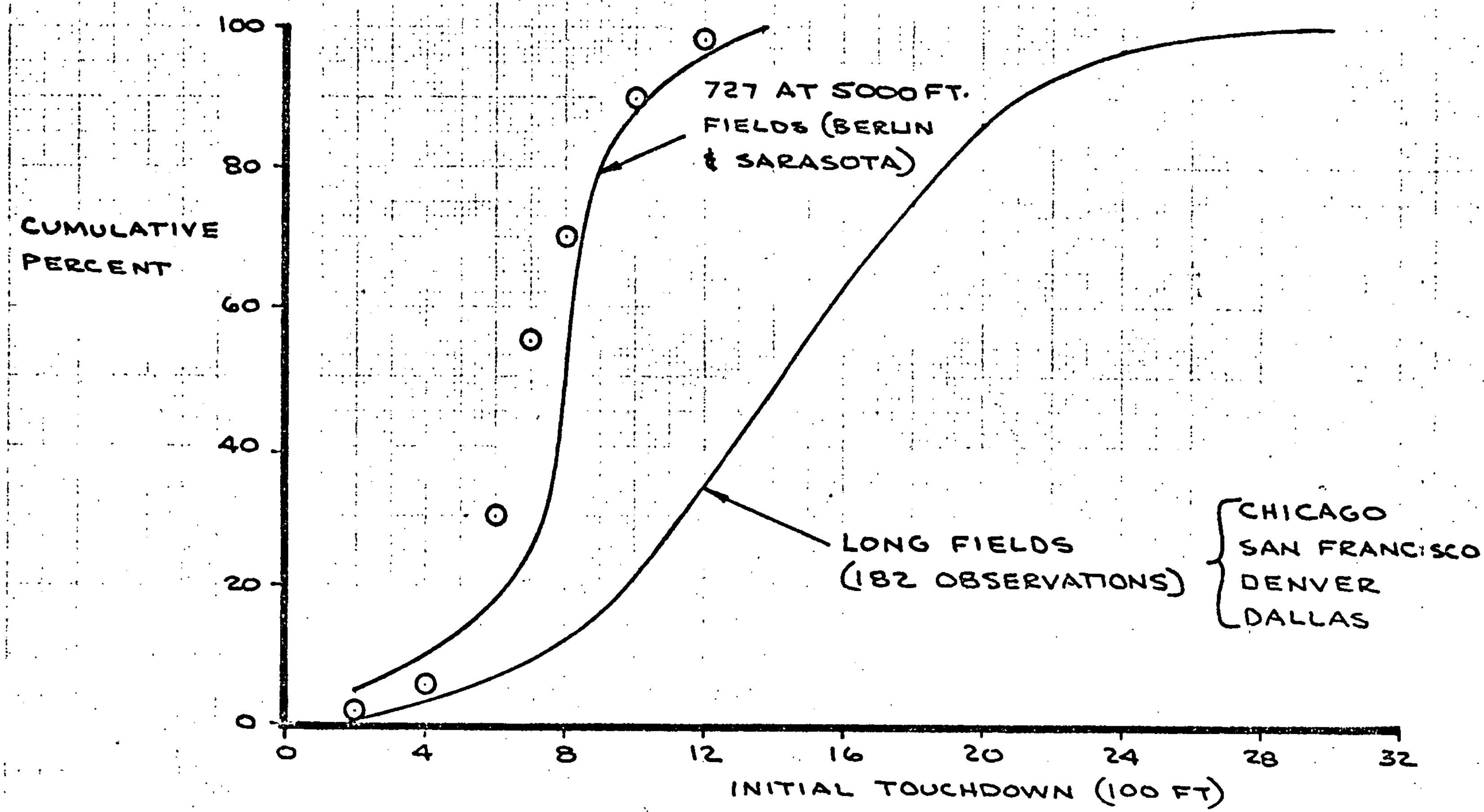
AUGMENTOR WING
TOUCHDOWN DISPERSION DATA
$\tau_{NZ} = 0.6$

THE BOEING COMPANY

D6-40409

FIG 3.6-3



⊙ DATA POINTS FROM FIG. 3.6-3

REF. D6-25214 BOEING

DISPERSION DATA

STOL vs. CTOL

FIG 3.6-4

Page

4.0 CONTROL SYSTEM CHARACTERISTICS

4.1 General

This section deals with those aspects of the flight control system that influence the pilot's impression of vehicle handling qualities. The paragraph headings are conventional, but the customary distinction between primary and secondary controls has been dropped. For STOL aircraft, items such as throttles, flaps, and trim systems tend to bear close relationship to vehicle handling qualities and safety. Segregation of these functions is no longer recommended.

4.2 Cockpit Control Travels

Criteria: - The cockpit controllers shall have the following travels:

Longitudinal	±4.0 to ±6.5 inches
Lateral*	±3.0 to ±6.5 inches
Directional	±2.5 to ±4.5 inches

*If a wheel is used the maximum travel should not exceed ±60 degrees.

Discussion: These criteria are taken directly from Reference 7. The present study used ±6 inch column; ±60 degree wheel; and ±3.5 inch pedal. These travels were satisfactory, but parametric variations were not run.

4.3 Control Centering and Breakout

Criteria: The cockpit controls shall have positive centering in flight at all normal trim settings. Absolute centering is not required but controller positioning should be sufficiently precise to provide ease of stabilization and must be compatible with augmentation system sensor requirements. Centering forces should not interfere with tracking tasks.

Pitch	0.5 to 3.0 pounds
Roll	0.5 to 3.0 pounds
Yaw	1.0 to 10.0 pounds

Throttle 1.0 to 3.0 pounds

Discussion: These criteria are taken directly from Reference 7. In the present study the nominal breakout forces were 2.0 pounds column; 2.0 pounds wheel; and 6.0 pounds pedal. These levels were satisfactory for one-hand operation. Parametric variations were not run.

4.4 Feel System

Criteria: The slope of the force versus displacement curves beyond the breakout region shall fall within the following ranges:

Pitch control	2 to 5 pounds per inch
Roll control	1 to 3 pounds per inch
Yaw control*	10 to 35 pounds per inch
Throttle	0

The increase in force produced by a one-inch travel from trim shall be greater than the breakout force.

Discussion: The force gradients are based on Reference 7. The maximum force levels associated with these gradients and associated breakout forces are compatible with the maximum force level requirements of References 5 and 8 except as noted for the pedals. The column gradient range of 2 to 5 pounds/inch has been retained for compatibility with the requirements for other axes, although Boeing transport experience has shown that the optimum level can be expected to lie near the upper end of this range.

The requirement relating to the force buildup in the first inch of travel and the breakout is intended to provide good centering, allow precise positioning for small travels, and to protect against pilot induced oscillations, PIO. It should be noted that for the roll axis the incremental force for one inch

*The maximum pedal force shall be less than 100 pounds. If a high initial gradient is chosen, a nonlinear shape must be provided to limit the maximum force level.

of travel (1.6 pounds) used in the present study is slightly less than the breakout force (2 pounds). This was considered to be satisfactory and would suggest that the gradient-breakout criterion is approximate.

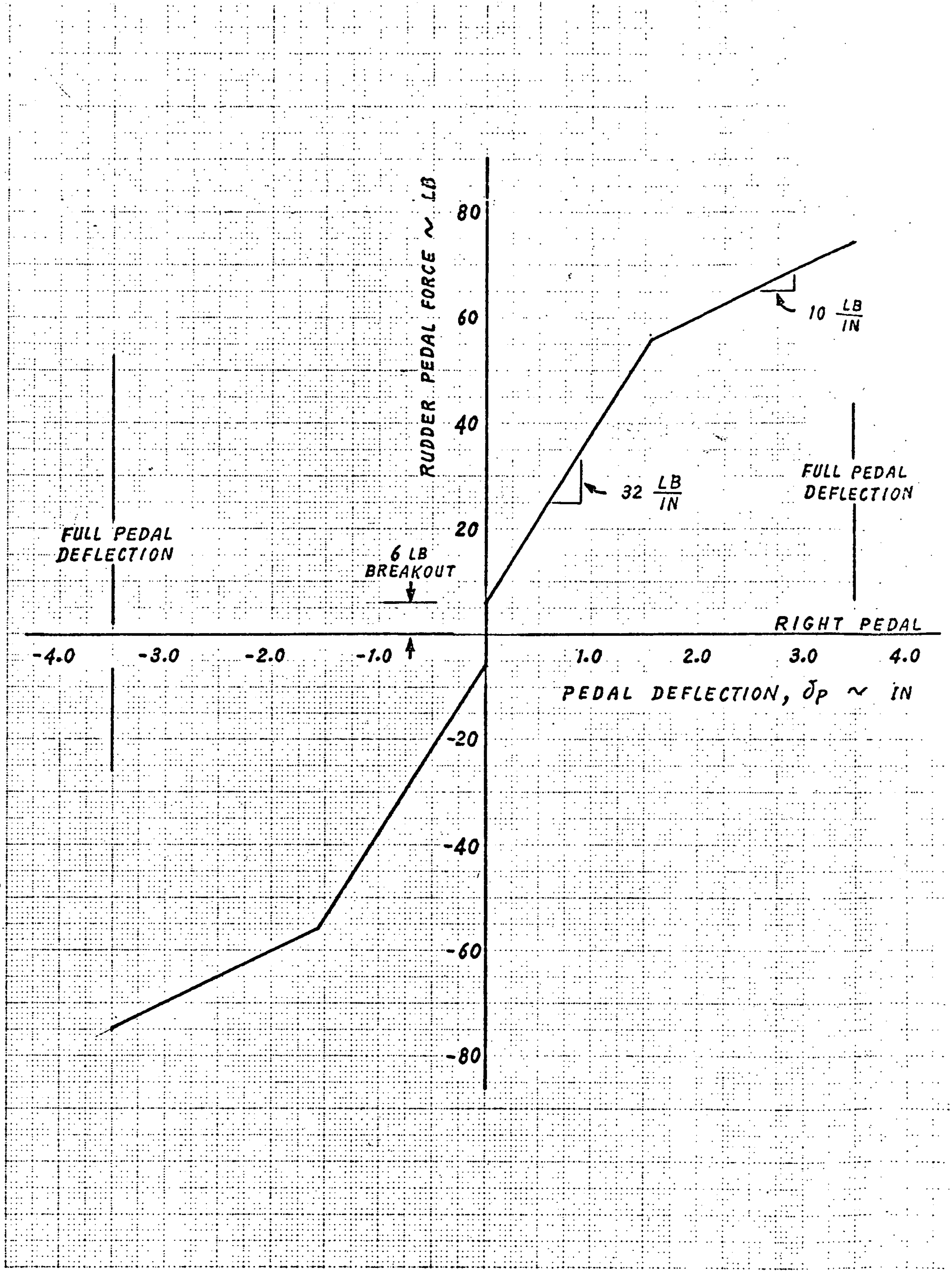
The present study was run with nominal linear gradients of 5 pound per inch (50 #/g) on the column and 1/6 pound per degree (1.6 pound per inch) on the wheel. The pedals used a nonlinear gearing as shown in Figure 4.4-1. These sensitivities, in combination with the breakout levels of the Section 4.3 discussion, were satisfactory for one-hand operation.

A limited study of column force gradient and gearing (inch/g) was conducted. The results are summarized in Table 4.4-1. These data suggest that the optimum longitudinal force gradient is a function of the longitudinal gearing (Runs A, B & D). The pilot selected a column sensitivity of 50 pounds per g as near-optimum. The accompanying lateral gradient was 1/6 pound per degree and the lateral sensitivity was 0.134 $\frac{\text{rad/sec}^2}{\text{inch}}$. Other lateral sensitivities were not examined. These data also show that the choice of longitudinal force gradient is influenced by control forces developed in flare (Compare runs B and D).

The choice of column gradient must be based on harmony, inner loop dynamic response, sensitivity (Fs/g), and trim force considerations. The force gradient criteria of this section should, however, provide a useful guide for preliminary control system designs.

TABLE 4.4-1

RUN	FORCE GRADIENT LATERAL	FORCE GRADIENT LONGITUDINAL	LONG. GEARING IN/g	STICK FORCE PER g	PILOT COMMENTS
A	1/6#/deg (1.5 #/in)	5#/in	15 in/g	75#/g	Lateral forces good but lat-long. harmony poor
B	1/6#/deg	5#/in	10 in/g	50#/g	Harmony good; forces good
C	1/6#/deg	5#/in	5 in/g	25#/g	Forces and damping decreased slightly
D	1/6#/deg	10#/in	5 in/g	50#/g	Forces undesirable in flare. Pitch sens. O.K.
E	1/6#/deg	7.5#/in	10 in/g	75#/g	Deterioration



CALC	MACK	9/3/71	REVISED	DATE	DIRECTIONAL - CONTROL FORCE CHARACTERISTICS	D6-40409
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APR					THE BOEING COMPANY	PAGE 83

4.5 Control Harmony

Criteria: Longitudinal and lateral breakout forces, force gradients, travels, and sensitivities, shall be compatible so that intentional inputs to one control axis will not cause inadvertent inputs to the other.

Discussion: Control harmony requires careful selection of many control parameters and specific criteria have not been formulated. In general, final selection of critical control system parameters must be based on piloted simulator studies. The control parameters used in the present study are found in Sections 4.2, 4.3 and 4.4. Control harmony was judged to be adequate.

4.6 Control System Free Play

Criteria: Free play in any of the cockpit controls shall not result in objectionable flight characteristics. Particular attention should be given to small amplitude control inputs in critical tracking situations. The analysis of stability augmentation and automatic control systems shall include appropriate deadband estimates.

Discussion: Free play produces an amplitude-sensitive phase lag in the control system response that can degrade pilot performance in precision tracking maneuvers and can result in PIO. Free play can also cause limit cycle problems in automatic control system designs.

4.7 Powered Control Systems

Criteria: The ability of the aircraft to satisfy the stability, control, and handling qualities criteria shall not be limited by any powered control system component. Powered controls shall be capable of producing the defelections required for all operational maneuvers. The surface rate capabilities shall be adequate to perform critical combined axis tasks such as approach and landing in a heavy turbulence environment. The effect of engine speed on hydraulic flow rates shall be included. Actuator hysteresis shall be compatible with the requirements of Paragraph 4.6.

Discussion: Certain components in powered control systems typically require careful design attention. The ability of control surface actuators to produce the required deflections requires that actuators be sized for blow-down with the appropriate combinations of hydraulic system failures. It is further necessary to provide overtravel in actuator command paths so that actuator compliance will not limit the deflection under load.

Simultaneous surface activity that occurs in critical flight conditions such as approach and landing with heavy turbulence and pilot in the loop can place severe demands on hydraulic flow capability. Oil temperature effects and the effect of engine power setting on pumping capacity must be included in this analysis. The effect of command servo rate limiting must also be included. A piloted simulation in which the hydro-mechanical systems and components are accurately modeled and atmospheric turbulence is represented by continuous power spectra is desirable.

4.8 Control System Dynamic Characteristics

Criteria: Control system dynamic characteristics, in combination with airframe dynamics shall provide satisfactory aircraft handling qualities. The frequency and damping of the cockpit controllers shall be selected so as to provide good pilot performance in tracking maneuvers. The cockpit control deflection shall never lead the control force for pitch, roll, yaw and thrust controllers.

Control system oscillations shall not degrade aircraft handling qualities or safety. Sensor and direct component coupling through structural modes shall be investigated. Control system response shall be quick enough to provide the overall vehicle responses required in Sections 5.3, 5.4, 7.1 and 7.2.

Discussion: These criteria are based on References 7 and 8. The criteria for feel system lags and overall control-vehicle frequency response of

Reference 3 have not been used due to lack of supporting data for STOL configurations.

4.9 Augmentation Systems

Criteria: The stability augmentation or command augmentation system shall be designed so that when functioning normally no adverse air or ground handling characteristics are produced. Particular attention shall be given to operation in turbulent air, on-the-ground operation where control surfaces might be driven hard over, and maneuvers where component authority limits or saturation could occur.

If aircraft handling characteristics change significantly with SAS failures, redundancy levels shall be compatible with handling qualities requirements for a given level of aircraft flight status.

Discussion: This paragraph is redundant with that proper compliance with other criteria should guarantee acceptable augmentation system design. Experience has shown, however, that some items tend to be overlooked in the design process and must be given good visibility if a timely design is to be achieved. These criteria are included for this purpose.

These criteria are based on References 7 and 8. They are intended to direct proper design attention to component authorities, saturation limits, bandwidths, and reliabilities; system control laws; sensor locations; system redundancy levels, etc.

4.10 Trim Systems

Criteria: For all flight conditions where continuous operation is required, the trim devices shall be capable of reducing the control forces to zero. Operation must be smooth and free from mechanical cross coupling effects. The trim controls shall be conveniently located and must provide the pilot with an indication of the amount of control remaining.

Powered trim systems shall be drift-free and shall have runaway protection. Rate capabilities shall be sufficiently high to prevent excessive force buildup, but not so rapid that overcontrol results. In determining rates and authorities both low and high speed flight conditions shall be considered.

Discussion: These criteria are based on References 7 and 8. The requirement for position indication is satisfied directly for parallel systems where controller zero-force position is varied. For series trim systems, or control systems utilizing an integrator in the forward path, a trim position indication must be provided. For optimum operation in approach and landing it is felt that it must be possible to trim the steady pitch and roll forces to zero following a critical engine failure.

4.11 Auxiliary Controls

Criteria: The design of the auxiliary controls shall be compatible with their specific relationship to vehicle handling qualities and safety. Cockpit controllers shall be conveniently located and functionally compatible with pitch, roll, and yaw controllers so that harmonious operation shall exist for all flight modes and operations.

Control functions such as thrust vector, throttle system, flap system, drag and lift devices (other than flaps) shall be drift free and shall have adequate rate capabilities to insure proper operation for critical flight maneuvers such as approach and landing in heavy turbulence. Combined axis requirements shall be investigated. Special attention shall be given to minimizing system time lags.

If automatic control of these devices is used to artificially produce a given mode of control (for example, conventional aircraft flight path control mode on approach), consideration shall be given to change in control technique resulting from failures as discussed in Section 4.12.

Discussion: These criteria are intended to focus design attention on controller and control functions that have traditionally been considerd secondary. At present the configuration of STOL control systems is largely unknown. It is possible that conventional cockpit controllers will be retained and that complex interconnections between auxiliary controls will be used to produce conventional airplane response. It is also possible that new piloting modes will be developed and that new cockpit controller configurations will be used.

STOL aircraft that have high powered lift levels present unique control system design problems. A critical area is flight path control on approach and landing which is discussed in detail in Section 5.0. The natural characteristics of the powered lift STOL vehicle require a mode of operation in which flight path is tracked with throttle and speed is controlled with column. This technique is foreign to fixed-wing airline pilots with conventional training. It is possible to change the characteristics of the powered lift STOL, through automatic control of auxiliary control devices, so that conventional fixed-wing piloting techniques can be used, but this results in a more complex control system design.

Regardless of the approach that is ultimately chosen, it is apparent that systems such as throttles, flaps, thrust vector control, and so forth, will have more stringent response and reliability requirements than they have traditionally had. In the present study only conventional cockpit controllers (column, wheel, pedal, throttles) were used. For system configurations where auxiliary control functions were used, (for example, trailing edge flap modulation), these motions were commanded by augmentation system and conventional controller commands. Additional controllers were not introduced into the cockpit. Boeing simulator experience has shown that a pilot will degrade his rating of a configuration where his hands must manage more than two primary control tasks simultaneously.

4.12 Failures

Criteria: In selecting the normal aircraft flight control mode, careful consideration shall be given to the flight control system complexity required to develop this mode, and to the changes in aircraft piloting technique that can result from probable flight control system failures. Emergency piloting modes that require special pilot training are dangerous and unacceptable unless it can be guaranteed that every pilot will have, and retain, the special skill level required.

In designing the flight control system, redundancy levels and positive warning devices shall be provided as required to match the skill levels of all possible crew members.

Discussion: This document deals with normal handling qualities and these failure criteria are therefore concerned with those considerations that influence the choice of normal control mode and the related configuration of the flight control system.

In the Section 4.11 discussion, it was pointed out that the natural characteristics of powered lift STOL vehicles lead to a piloting mode that is foreign to pilots with conventional fixed-wing training. A more conventional piloting mode can be produced by designing a flight control system that introduces coupling between basic and auxiliary controls. The danger that exists is that if a flight control system design is selected that provides a conventional piloting mode, there are potential system failures that can drastically change the piloting technique required for continued flight. The designer has the choice of specifying special training for the flight crew, of providing system redundancy levels that will preclude dangerous system failures, or of accepting the operational penalty of diverting from the destination STOLport to a conventional length runway after a failure.

5.0 LONGITUDINAL CONTROL

5.1 General Discussion

The longitudinal control criteria have been expressed in terms of the control power, control sensitivity, and dynamic response requirements, for control of:

- o Pitch attitude
- o Flight path
- o Speed

Satisfactory pitch attitude control is treated as being necessary, but not sufficient for satisfactory handling qualities; and the required attitude response depends on whether attitude is used primarily for flight path control or speed control. Two piloting techniques are defined to distinguish between these two approaches to attitude control, and the criteria are related to the piloting techniques.

Load factor response is used as the fundamental parameter in defining satisfactory flight path control. Speed control criteria are not well defined but tentative criteria are proposed as a starting point.

Flight path and speed control requirements for STOL were found to be similar to both CTOL and VTOL requirements, which suggests that it may eventually be possible to develop a common criteria in terms of those parameters. Differences appear to be related to the sink rate on final approach, the glide slope angle, and the distance between the glide slope transmitter and the desired touchdown point.

The current study was conducted with an 80 knot approach speed, a 6 degree glide slope, and a distance from glide slope transmitter to center of the aim zone of 150 feet (Reference 4).

For this configuration, it was found that the load factor response time had to be quicker than for CTOL, and the load factor authority must be more than required by VTOL criteria.

It was found that flight path response has a strong effect on the required field length. With a steep glide slope and a relatively short distance between the glide slope transmitter and the aim point, it is necessary to make an abrupt flare fairly close to the runway. If the flight path response is sluggish, the flare must be initiated at a higher altitude, which results in landing long. If the pilot delays the flare in an attempt to hit the aim point with the sluggish system, the result is likely to be a short hard landing, or an over-flare and float to a long landing.

The longitudinal control criteria apply only to configurations which require continuous manipulations of no more than two controllers in order to control attitude, airspeed, and flight path angle.

In the interest of verbal simplicity, these controllers have been called "column" and "throttle". The term "column" is used for the pitch controller, since columns are commonly used in transport aircraft. However, it is not intended to rule out the use of sticks or push-pull controllers. In conventional transports, the "throttles" are used to control engine thrust. However, within the context of the STOL control technique definitions, the "throttle" might actually be the controller for thrust vectoring or direct L/D control devices. Whatever device the "throttle" actually controls, the pilot should not be required to remove his hands from it except for momentary and infrequent actuation of a "trim control" or to make a discrete configuration change.

5.2 Control Techniques, Related Parameters, and Systems Tested

5.2.1 Control Techniques

Criteria: The column should always provide direct and rapid control of pitch attitude. Aft column should produce a nose-up response. Forward column should produce a nose-down response.

Short term response of flight path and speed should be compatible with one of the following piloting techniques:

Technique #1 - Pitch rate/attitude are used to control load factor/flight path angle. The throttle is used to control airspeed.

Technique #2 - Pitch attitude is used to control airspeed. The throttle is used to control load factor/flight path angle.

Subsequent longitudinal control criteria have been related to these two techniques.

Discussion: Before establishing criteria for controlling a vehicle, it is necessary to define:

- o The variables to be controlled
- o The controllers to be used
- o The control technique (cross-check logic) to be employed by the pilot

These definitions are sometimes omitted in the flying qualities literature for airplanes, because the piloting techniques and the response to contol inputs are generally well understood. Longitudinal handling qualities criteria for airplanes (e.g., Ref. 3) are written with the objective of providing satisfactory control of both airspeed and flight path (about a given trim point) using one controller only (the pitch controller). An exception is made in the case of a landing approach on the "back side of the drag curve", (i.e., $\frac{\partial \gamma}{\partial V} > 0$), where the criteria allow the use of thrust to control the speed divergence that results when pitch attitude is constrained (by the pilot) to the attitude required for tracking the glide slope.

A powered lift STOL aircraft, by definition, is not an airplane. The STOL flight envelope bridges the gap between VTOL and CTOL; and the STOL response to control may run the gamut from a VTOL-type response to a CTOL-type response, depending on the magnitude of the powered lift effects, As a specific example, advancing the throttles in an airplane produces primarily a forward acceleration; whereas in some STOL aircraft advancing throttles produces primarily an upward acceleration.

If the STOL response to thrust application is primarily a vertical acceleration, then control Technique #1 (conventional jet transport technique) simply will not work. However, control may be quite satisfactory if Technique #2 is used. In fact, some of the powered lift characteristics which make the use of Technique #1 difficult or impossible tend to enhance controllability with Technique #2.

Consequently, one of the first and most important decisions for the STOL aircraft control system designer is whether to:

1. Add the extra control devices (modulated flaps, vectored thrust, etc.) necessary to force the STOL aircraft to fly like an airplane; or
2. Accept a response compatible with the natural STOL characteristics, but possibly unfamiliar to airplane pilots.

The first approach may require a heavier and more expensive airframe; and has the potential disadvantage of forcing the pilot to switch to an unfamiliar control technique if the stability and control augmentation system fails. The second approach may require a faster engine response, and additional training time for pilot checkout.

It is expected that vehicles with satisfactory (though quite different) handling qualities can be developed using either approach. However, different piloting techniques will be required; and the criteria will be different in certain areas. In an attempt to accommodate both design approaches while

while retaining enough specific numerical criteria to be meaningful, two different piloting techniques have been defined. The longitudinal control criteria have then been related to these two techniques.

Both techniques require that pitch attitude be controlled with the column. This is consistent with existing VTOL and CTOL practices; and there is no apparent reason why STOL should be different. With attitude controlled by the column, attitude and throttles become the two controllers available to control speed and flight path. The difference in the two techniques is whether the cross-check logic associates a speed error with a need to make an attitude change or a throttle adjustment. It is recognized that with either technique the pilot must coordinate attitude and throttles as the situation warrants. However, the distinction between the techniques is one of emphasis.

The two piloting techniques defined by the criteria are sometimes referred to as the "front side" (of the drag curve) and the "back side" techniques, respectively. However, the need for using Technique #2 could arise from several factors other than being on the back side of the drag curve. These factors are discussed in Section 5.2.2.

Reference 9 discusses the two piloting techniques from a pilot's point of view; and makes the important observation that flight on the back side of the drag curve, while not difficult, requires continuous control applications with both hands -- one on the column and the other on the throttle. Since most unaugmented STOL aircraft will approach on the back side, the pilot should not be required to take his hands off the column or throttles during the approach. This requirement may be unduly restrictive for an experimental aircraft, but it is considered reasonable for routine airline operations.

Speed and flight path angle are considered to be the two primary controlled variables because they determine the touchdown point, landing impact, stopping distances, and (in combination) the stall margin. This is not meant to imply that the pilot must fly a reference approach speed or that the cockpit must contain a γ indicator. As pointed out in Reference 5, angle-of-attack is sometimes more useful than speed as an approach reference; and vertical speed indicators are normally used for flight path control. Certainly the full complement of instruments will be used in the cross-check. Whatever instruments are used, however, the net result must be precise control of the length and direction of the total velocity vector (i.e., speed and flight path angle).

5.2.2 Parameters Influencing Control Techniques

<u>Criteria</u>: - The choice of control technique and/or the requirement for augmentation is influenced by several parameters. Some of the more important ones are tabulated below. Parameter values should be within the range indicated:

TABLE 5.2-1

PARAMETER	LIMITS	
	TECHNIQUE #1	TECHNIQUE #2
n_z/α	$\geq$ .04 g/degree (See discussion)	≥ 0 (See discussion)
$\partial\gamma/\partial V$ at constant thrust	$\leq +$ .06 deg/knot*	Unknown $\frac{\partial\gamma}{\partial V} < +.2 \frac{deg}{knot}$ suggested
$\Delta\theta/\Delta\gamma$ at constant V	≥ 0.75	Unknown. Negative values undesirable, but allowable.
$\Delta\theta/\Delta V$ at constant	< 0 Desired	$-.6 \text{ deg/knot} < \frac{\Delta\theta}{\Delta V} < 0$
$\eta_T \sim$ effective thrust vector angle	Unknown $0 \rightarrow 45°$ Suggested	Unknown $13° \rightarrow 90°$ Suggested

*Also the difference in $\partial\gamma/\partial V$ between V_{MIN} and V_{Min}-5 knots shall not exceed .05 deg/knot.

Definitions of the parameters are as follows:

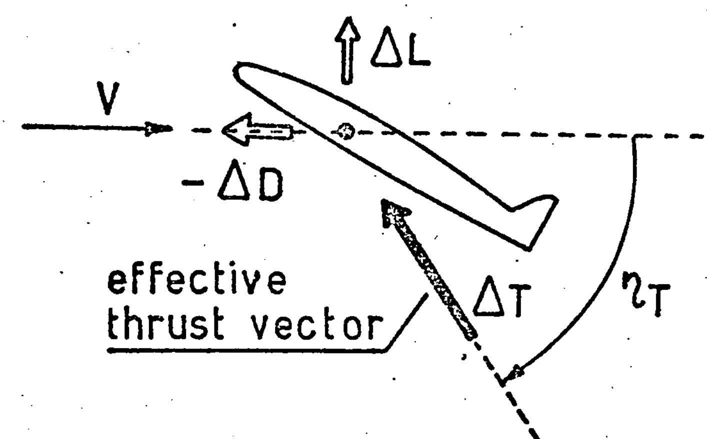
n_z/α = the steady state normal acceleration change per unit change in angle of attack, due to a column input at constant speed with throttles fixed.

$\partial\gamma/\partial V$ = slope of trimmed γ vs V for speed changes resulting from column inputs with throttles fixed

$\Delta\theta/\Delta\gamma$ = slope of trimmed θ vs γ when γ is changed at constant V

$\Delta\theta/\Delta V$ = slope of trimmed θ vs V when V is changed at constant γ

η_T = $\tan^{-1}\left[\dfrac{\Delta(L)}{\Delta(-D)}\right]_{\alpha = CONST}$ where L and D include all thrust effects. The effective thrust vector angle, η_T, gives the direction of the initial reaction to a thrust change, as indicated in the sketch below



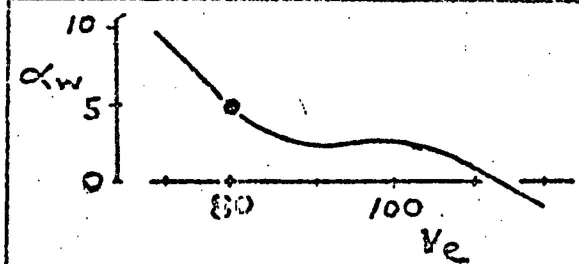
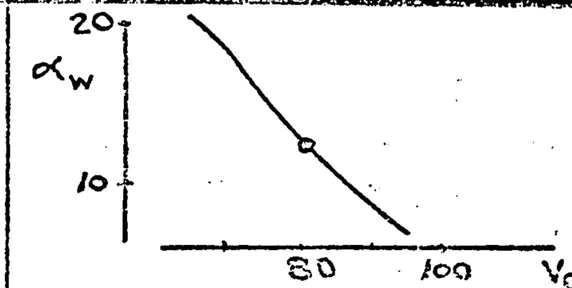
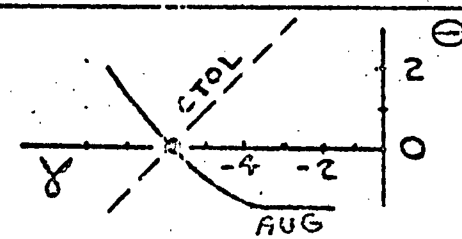
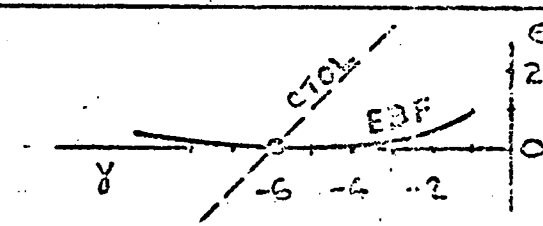
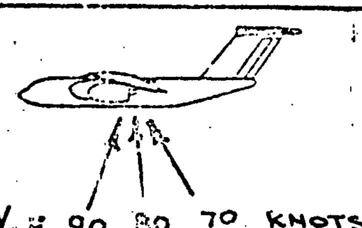
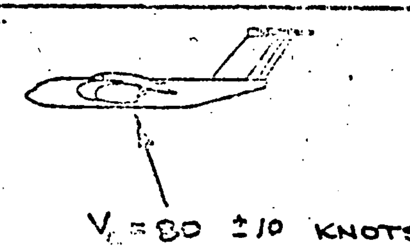
Discussion: Considerable insight regarding the controllability of an aircraft can be obtained from an examination of "static" trim and force data. The criteria presented here are intended to direct attention at several "static" parameters which have a strong effect on controllability and which are relatively easy to use in the preliminary design phases. Data for the configurations used in the current study are summarized on Figures 5.2-1 and Table 5.2-2.

The proposed limits are very tentative and should be the subject of additional studies. A discussion of each parameter follows:

A. n_z/α - Definition

The definition of n_z/α for STOL aircraft is modified from the definition given in Reference 8 in that "throttle fixed" has been added. While this presents a problem in testing for n_z/α at

COMPARISON OF LONGITUDINAL CHARACTERISTICS OF AMES SIMULATION MODELS

ITEM	AUG. WING	E B F
TRIM ANGLE OF ATTACK VARIATION WITH SPEED. $\gamma = -6^0$		
TRIM PITCH ATTITUDE VARIATION WITH FLIGHT PATH ANGLE. V = 80 KTS.		
EFFECTIVE FORCE VECTOR ORIENTATION FOR THRUST APPLICATION $\gamma = -6^0$		
SPEED, FLIGHT PATH AND FLARE CONTROL	• NO AUX. FLAP ~ EBF EASIER TO CONTROL. • WITH AUX. FLAP ~ NO DIFFERENCE BETWEEN EBF & AW <u>NOTE:</u> AUX-FLAP REQUIRED TO GIVE CONVENTIONAL θ / γ RELATIONSHIP AND THRUST VECTOR ORIENTATION.	

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COMPARISON OF LONGITUDINAL CHARACTERISTIC OF AUG. WING AND EBF MODELS

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FIG 5.2-1

TABLE 5.2-2
BASIC AIRFRAME (NO SAS) DATA FOR CONFIGURATIONS TESTED

	PARAMETER	AUG. WING	EBF
A	n_z/α g PER DEG.	0.019	0.026
B	$\partial\gamma/\partial V$ DEG / KNOTS	+0.17	+0.18
C	$\Delta\theta/\Delta\gamma$ DEG/ DEG	-1	0
D	$\Delta\theta/\Delta V$ DEG / KNOT	-5 NON-LINEAR WITH REVERSALS	-5 LINEAR
E	η_T	80° VARIES WITH SPEED	70° CONSTANT WITH SPEED

NOTE: 1. SLOPES MEASURED AT $\gamma = -6°$, V= 80 KNOTS TRIM POINT
2. BOTH CONFIGURATIONS UNFLYABLE WITH TECHNIQUE #1
3. EBF MARGINALLY SATISFACTORY WITH TECHNIQUE #2
4. AUG. WING UNACCEPTABLE WITH TECHNIQUE #2

ENGR.	ALLISON	FEB 72	REVISED	DATE	COMPARISON OF PARAMETERS RELATED TO CONTROL TECHNIQUE SELECTION	D6-40409
CHECK						TABLE 5.2-2
APR					BOEING	100
APR						

"constant speed", it is considered necessary due to the effects of blowing modulation. If thrust is added to maintain constant speed, the resultant load factor increment could give a much higher apparent value of n_z/α than if only a column input were made. Since n_z/α is one of the parameters used to determine if flight path can be controlled and the aircraft flared via attitude changes (Technique #1), the effects of throttle inputs should be excluded. However, automatic blowing modulation or other DLC devices that function with the throttles fixed may be used to augment n_z/α, and are not excluded by the definition.

A possible way to test for n_z/α is to begin a pullup from a shallow dive with throttles fixed, and measure the Δn_z and $\Delta\alpha$ at the instant speed returns to the trim condition. The throttles-fixed constraint is, of course, no problem for an analytical calculation of n_z/α.

n_z/α- Control Technique #1

When using control Technique #1, the aircraft is pitched by means of column inputs to develop load factor. In the absence of thrust or blowing modulation, or other DLC devices, lift changes must be produced by angle of attack changes. n_z/α is a very important handling qualities parameter for landing approach in that it influences:

- o the pitch acceleration ($\ddot{\theta}$) capability required to achieve a given vertical speed crossover time, as shown by Equation 5.3.1.3.

- the control anticipation parameter, $(\ddot{\theta}/\delta_{col}) \times (\delta_{col}/n_z)$ as discussed in Paragraph 5.3.2 and in Reference 3.
- the attitude change required to flare.

The $n_z/\alpha > .04$ g/deg limit corresponds to the value suggested in Reference 8 for determining the conversion speed between "airplanes" and "STOL aircraft". It is about midway between the MIL-F-8785, Paragraph 3.2.2.1.1 lower limits for land-based and carrier-based transports. Since control Technique #1 is an "airplane" (CTOL) control technique, "airplane" criteria should apply. If n_z/α of the unaugmented airframe does not meet the above requirements for Technique #1, then:

- n_z/α should be augmented by means of thrust or blowing modulation or other DLC devices; or
- Technique #2 should be used.

When blowing modulation or DLC is used, the angle of attack change required for an incremental increase in load factor can become zero or negative. Hence, n_z/α can become infinite or negative, respectively. In this case, the negative n_z/α would be acceptable even though the criteria prescribes positive values; provided the $\Delta\theta/\Delta\gamma$ requirement is met.

References 3 and 8 indicate the minimum n_z/α criteria are somewhat arbitrary, and should be the subject of future study.

<u>n_z/α - Control Technique #2</u>

When using control Technique #2, direct thrust reaction or blowing effects on the aerodynamics are used to develop load factor. Consequently, angle of attack changes are not required to initiate a flight path change.

Maximum and minimum n_z/α limits have not been determined for Technique #2. Since a totally non-aerodynamic VTOL vehicle such as a "flying bedstead" could be controlled with this technique, it is expected that $n_z/\alpha \approx 0$ is the lower limit. However, positive n_z/α is desirable to provide vertical damping (Z_w), to minimize the required throttle activity, and to facilitate flaring. If n_z/α becomes too large, excessive throttle compensation would be required to maintain the flight path when the aircraft is pitched to change speed. In this case, Technique #1 should be used; or, if this is impractical due to unfavorable values of the other parameters listed in Table 5.2-1, "negative DLC" should be used to reduce the effective n_z/α.

The n_z/α limits for this control technique should be determined in future studies.

B. $\partial\gamma/\partial V$ - Control Technique #1

The slope of the flight path angle vs speed curve, $\partial\gamma/\partial V$ is related to the slope of the trimmed drag vs speed curve. Positive values of $\partial\gamma/\partial V$ are associated with flight on the back side of the drag curve. The significance of $\partial\gamma/\partial V$ is thoroughly discussed in numerous reports, e.g., Reference 3. Briefly, when on the "back side", drag increases if the pilot slows down and changes attitude as required to maintain the flight path. With the pilot (or autopilot) attempting to hold a fixed flight path using pitch control, a speed divergence results from the drag instability. In a conventional airplane, the speed divergence can be stabilized with throttle (or autothrottle) inputs, provided the divergence rate is not excessive. Hence, there is a need for a limit on the allowable instability.

The $\partial\gamma/\partial V \leq .06$ deg/knot limit was taken from MIL-F-8785, which was based on "elevator only" control of flight path, i.e. control Technique #1. This limit was based on experience with conventional airplanes, for which the effective thrust vector is oriented forward along the horizontal axis. As discussed below, the thrust vector for STOL aircraft may have a significant vertical component. The effect of thrust vector orientation on the allowable $\partial\gamma/\partial V$ instability has not been determined.

$\partial\gamma/\partial V$ - Control Technique #2

Aircraft can be controlled with $\partial\gamma/\partial V$ more unstable than the limit specified for control Technique #1. However, as discussed in Reference 9, experienced pilots will tend to switch to control Technique #2 when very far on the back side of the drag curve. With this technique, attitude is varied to eliminate the speed divergence; but the throttles must be used to retain the desired flight path. Limits on $\partial\gamma/\partial V$ for this technique and the effects of thrust vector orientation on these limits, are unknown. In the current study, flight path control was marginally satisfactory for two unaugmented configurations having $\partial\gamma/\partial V \approx +.2$ deg/knot.

C. $\Delta\theta/\Delta\gamma$ - Control Technique #1

When a conventional airplane makes a constant airspeed climb or descent, α remains essentially constant, so $\Delta\theta/\Delta\gamma = 1$. This need not be the case for a STOL aircraft. Consider the Augmentor Wing data on Figures 5.2-2 and 5.2-3 as an example. The Figure 5.2-2 trim curves show that angle of attack must be reduced in order to maintain $n_z = 1$ g in a climb at constant speed. The angle of attack reduction is necessary to compensate for the increased blowing that results when thrust is advanced to climb. The change in θ can be calculated from $\theta = \gamma + \alpha$, and is shown on Figure 5.2-3. It is seen that the unaugmented aircraft must be rotated nose down as it climbs, which is opposite from conventional airplanes. Control Technique #1 would be virtually impossible to use in this situation. Typical pilot comments on familiarization runs were "something is backwards", "are you sure the simulation is working right", etc.; even though the pilots had prior STOL experience. These comments were not heard during evaluations of the EBF, which Figure 5.2-4 shows could change γ at constant θ.

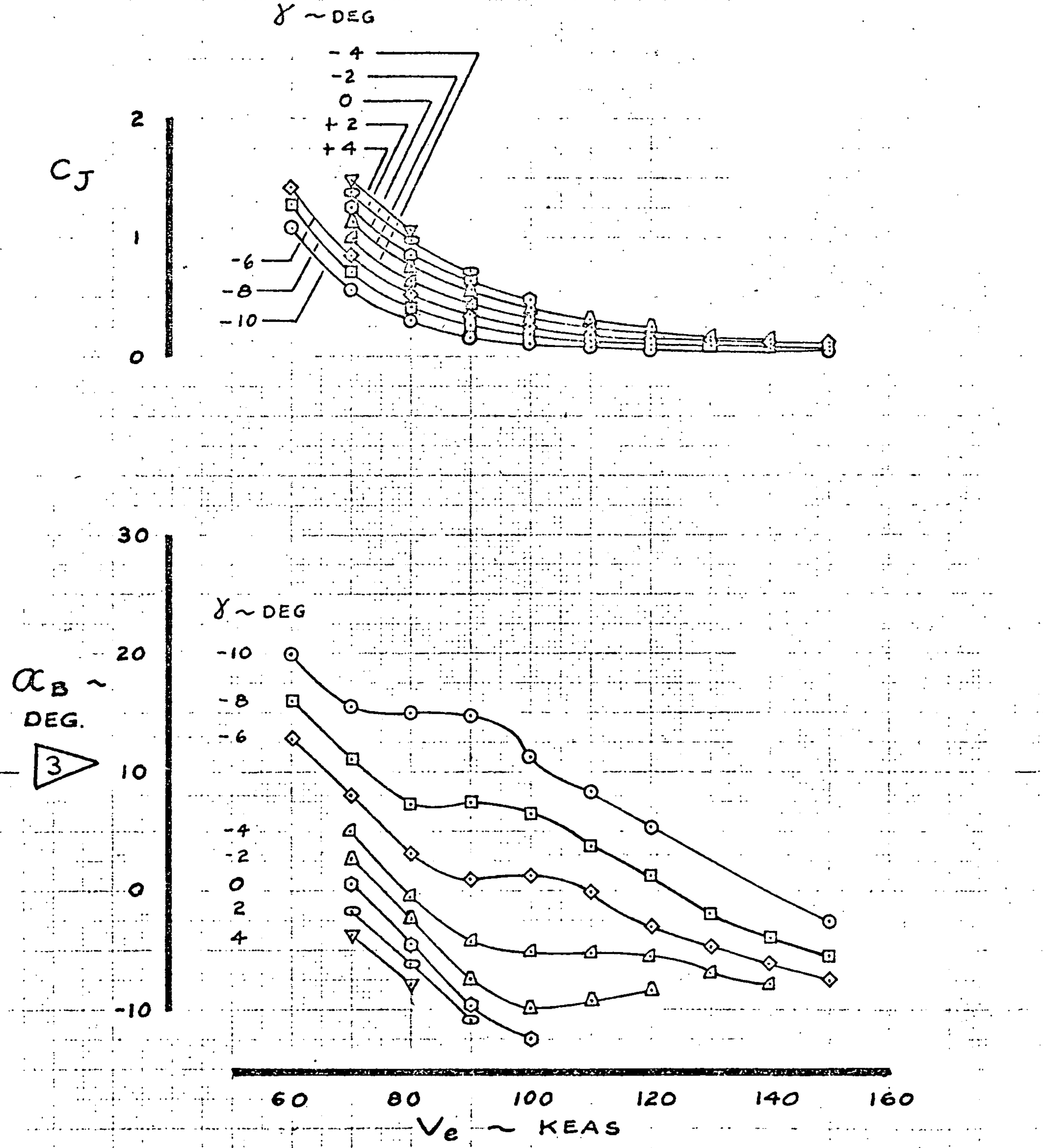
In the current study, the $\Delta\theta/\Delta\gamma$ ratio was controlled by modulating wing flaps. The $\Delta\theta/\Delta\gamma \geq .75$ limit is based on the results shown on Figure 5.2-5. At lower values, the pilot complained of having "insufficient lead information". The column step responses on Figure 5.2-12 show that pitch rate lagged load factor in the systems tested (due to the DLC mechanization) and this was probably more responsible for the adverse comments than was the steady state $\Delta\theta/\Delta\gamma$. It is likely that $\Delta\theta/\Delta\gamma$ ratios approaching zero can be successfully used. Since $\Delta\theta/\Delta\gamma = 1$ for conventional aircraft, it was decided to specify $\Delta\theta/\Delta\gamma \geq .75$ until a lower limit can be supported by test data.

This is considered to be an important area for further study as it impacts on the control power requirements of Section 5.3.1.3.

NOTE:
1. AMES SIMULATION DATA
2. W/S = 78

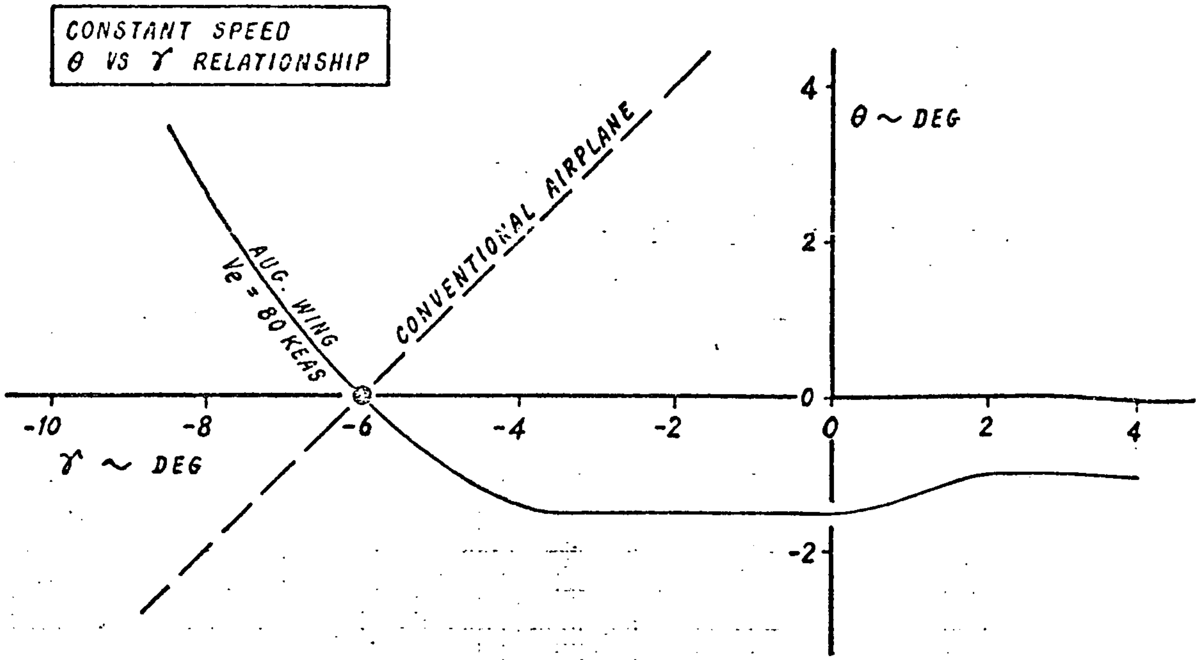
AUGMENTOR WING
4 ENGINES, $\delta_{FLAP} = 70°$
C.G. = .344 $\bar{c}$

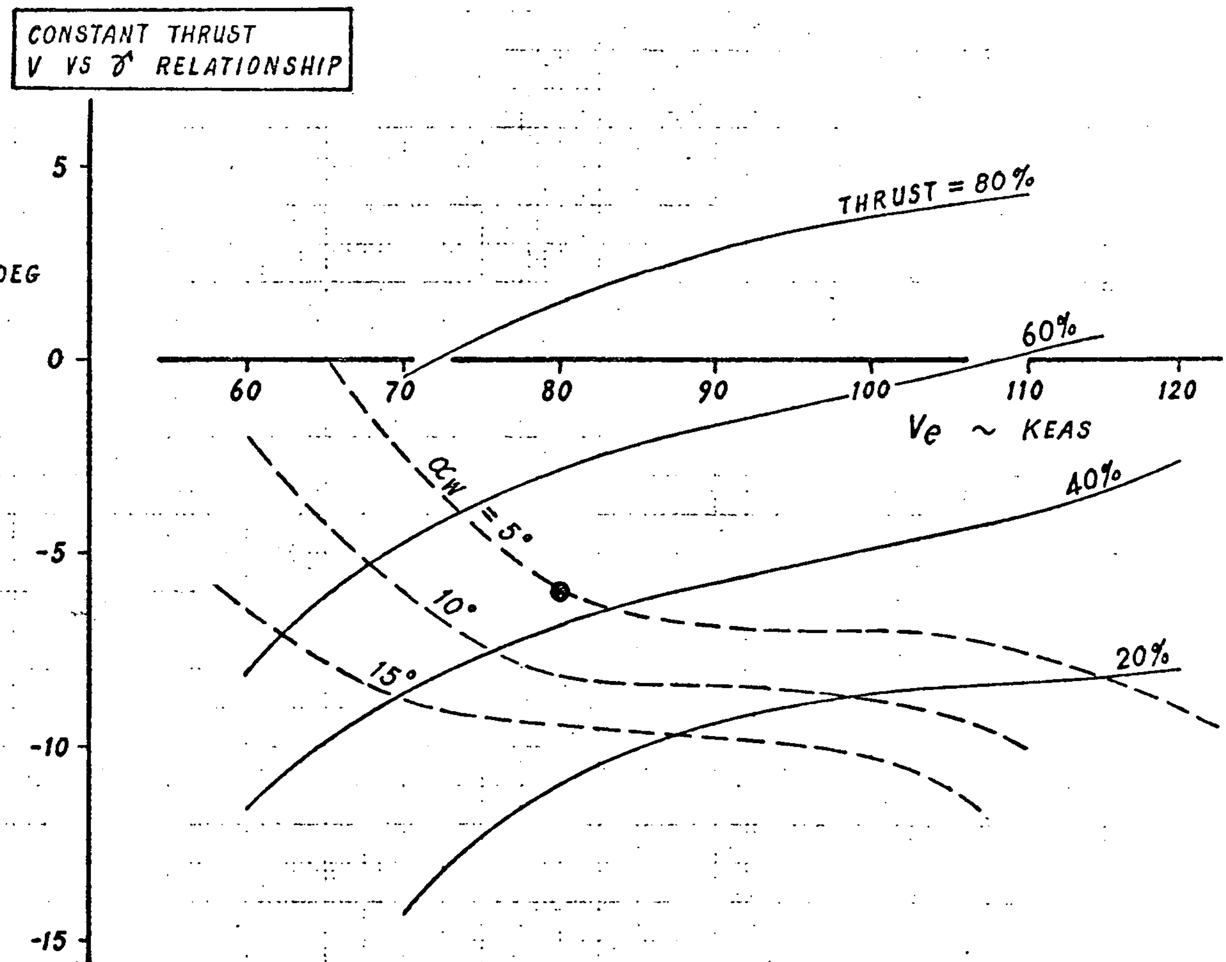
3 $i_w = +2°$



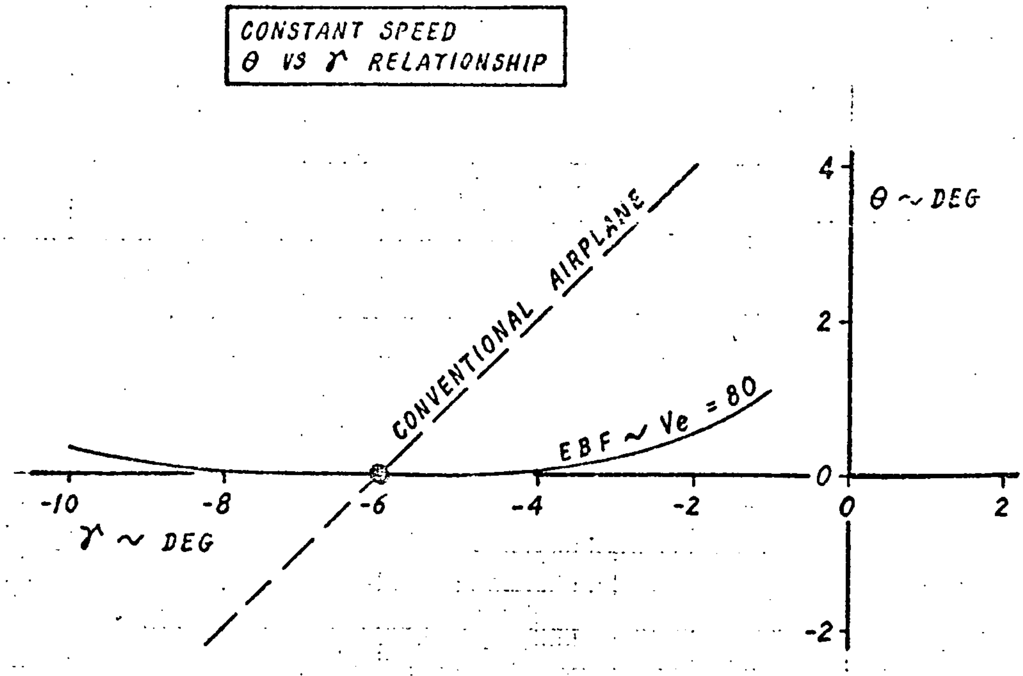
CALC	ALLISON	10-21-71	REVISED	DATE	TRIM DATA AT CONSTANT γ	D6-40409
CHECK						FIG 5.2-2
APR					THE BOEING COMPANY	PAGE 106
APR						

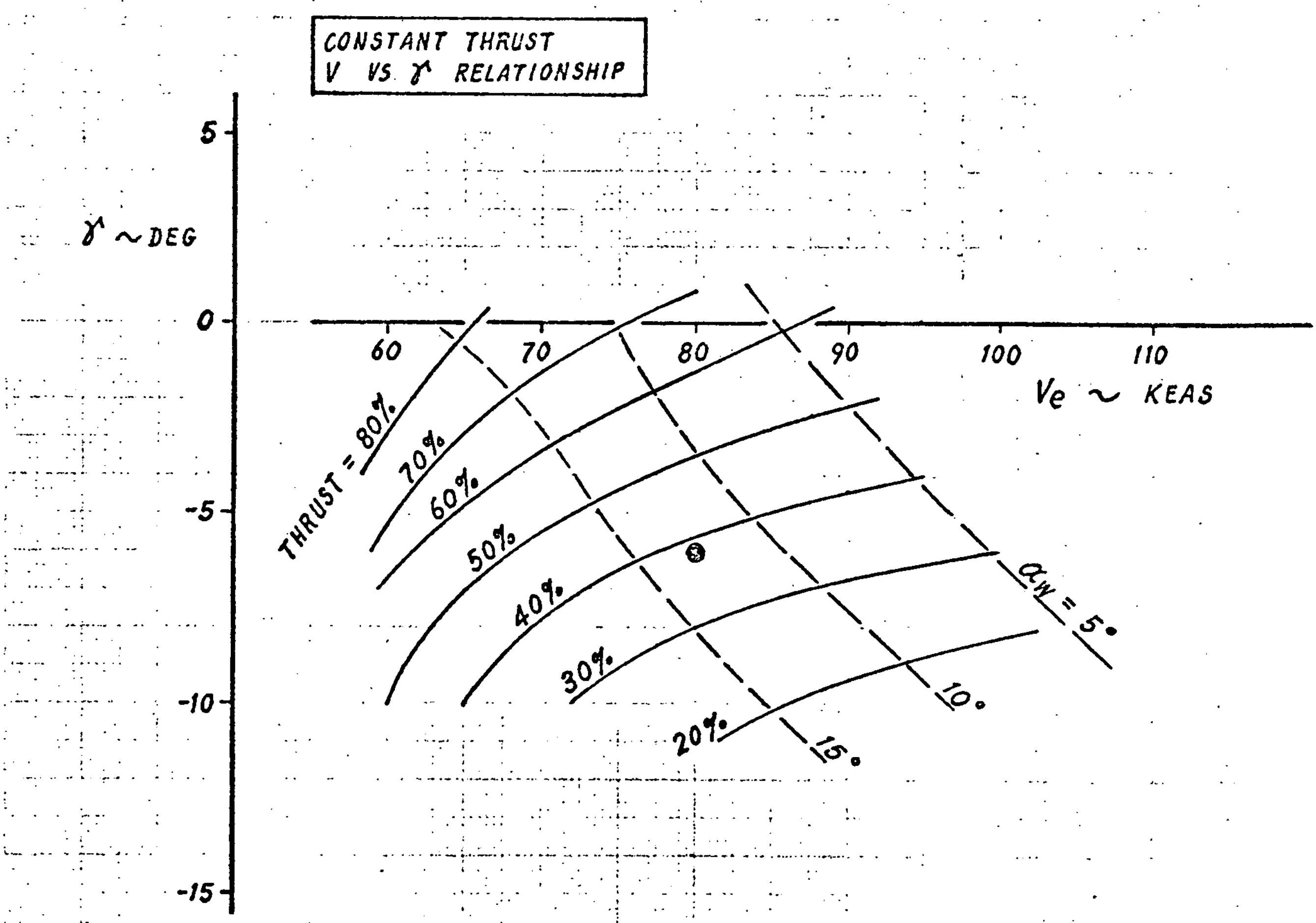
D1 4100 3240





CALC	ALLISON	2 DEC 71	DATE	TRIM DATA ~ AUGMENTOR WING	D6-40409
CHECK					FIG. 5.2-3
APR					
APR				THE BOEING COMPANY	107





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TRIM DATA ~ EXTERNALLY BLOWN FLAP

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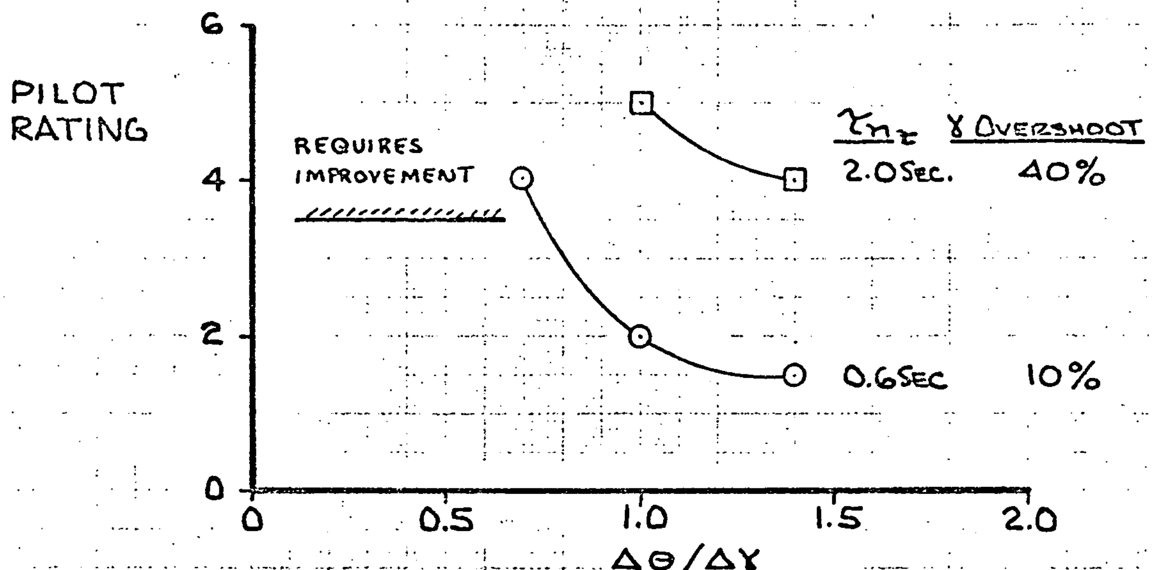
FIG. 5.2-4

THE BOEING COMPANY



NOTE: 1. AMES FSAA STOL SIMULATION DATA
2. APPROACH & LANDING TASK ON STOL RUNWAY. $V_{APP} = 80$ KTS, $\gamma = 6°$
3. DATA OBTAINED WITH AUG. WING MODEL, USING SAS #1 & CONTROL TECHNIQUE #1; i.e. $\delta_{COL} \rightarrow \theta \rightarrow \gamma$, THRUST $\rightarrow$ V
4. PARAMETERS VARIED BY MEANS OF FLAP & BLOWING MODULATION
5. RESPONSE TIME HISTORIES SHOWN ON FIGS 5.2-11 & 5.2-12

CALC	ALLISON	2 DEC 71	REVISED	DATE	EFFECT OF τ_{n_z} & $\Delta\theta/\Delta\gamma$ FOR CONTROL TECHNIQUE #1	D6-40409
CHECK						FIG. 5.2-5
APR					THE BOEING COMPANY	PAGE 109
APR						

$\Delta\theta/\Delta\gamma$ - Control Technique #2

When using this control technique, the pilot associated θ more closely with speed than with γ. Consequently, negative values of $\Delta\theta/\Delta\gamma$ appear to be satisfactory for control on the glide slope. However, for abrupt transient maneuvers, such as the flare, momentary increase in θ should not cause a reduction in γ.

The change in θ associated with flight path changes should not cause the pilot to lose signt of the runway, cause passenger apprehension, or exceed a geometry limit during touchdown.

D. $\Delta\theta/\Delta V$ - Control Technique #1

Normally, a conventional airplane rotates nose up (+) as speed is reduced (-) at constant γ, so $\Delta\theta/\Delta V < 0$. Figure 5.2-2 shows this was not the case for the unaugmented Augmentor Wing configuration, in that there was a slight reversal in the θ vs V relationship at $V \approx 95$ knots. This reversal resulted from non-linear blowing effects on the aerodynamics.

The SAS designed for use with Control Techinque #1 maintained a negative $\Delta\theta/\Delta V$. Since the pilot is associating θ with γ-control more than with speed-control, zero or positive values of $\Delta\theta/\Delta V$ are probably acceptable. However, negative values are very desirable during visual approaches as a warning that speed is changing.

$\Delta\theta/\Delta V$ - Control Technique #2

With this control technique, the pilot is using θ changes to produce speed changes. A nose-up rotation is used to obtain the longitudinal force unbalance necessary to initiate a deceleration. If the $0 > \Delta\theta/\Delta V$ criteria is not satisfied, the pilot would have to nose up to initiate the speed reduction, and then nose down in order to remain in trim when the thrust is adjusted to maintain the glide path. The aircraft would be on the "back side of the θ vs V curve" so to speak.

The further constraint $\Delta\theta/\Delta V > -.5$ deg/knot was taken from Para. 3.2.1.1 of MIL-F-83300. The intent is to insure that excessive attitude changes will not be required to make small speed changes.

E. Effective Thrust Vector Angle, η_T

Aerodynamic and thrust reactions are normally treated as separate terms when analyzing airplanes. This approach is virtually impossible with some powered-lift STOL configurations, due to the powerful effects of blowing.

Due to the inseparable nature of the aerodynamic and thrust effects, an extremely useful tool for analysing a STOL aircraft is a plot of the total forces normal and parallel to the remote velocity vector. Figure 5.2-6 is an example of such a plot, where the "lift" and "drag" coefficients contain the usual aerodynamic lift and drag terms, plus all propulsion effects.

If such a plot were made for a conventional jet airplane, it would look like a family of drag polars with $C_{D_{MIN}}$ shifted as a function of thrust setting. Note the following:

- lines of constant γ can be drawn through the origin. Hence, the maximum climb and descent angles can be determined.

- the approximate trim angle-of-attack and thrust setting is determined by the intersection of the constant γ line with the required trim C_L. (Exact data would require also trimming the pitching moment).
- the initial response to a thrust change can be determined from the slope of a constant α line through the trim point.

The direction of the initial response to a thrust change has been referred to as the effective thrust vector angle, η_T. The effective thrust vector orientation, for a thrust increase, has been indicated on Figure 5.2-6. For this configuration (Augmentor-Wing) a thrust increase always produces predominantly an upward acceleration. At the approach speed (80 knots) a thrust increase also produces a slight forward acceleration, with the forward component increasing as speed is decreased. As speed is increased above the approach speed, the thrust vector begins inclining rearward.

Hence, a thrust application would cause the Augmentor Wing model to climb and accelerate slightly at speeds below the approach speed; whereas it would climb and <u>decelerate</u> (slightly) at speeds above the approach speed. The pilots found the deceleration in response to a thrust increase to be quite disconcerting; and speed control was a prime factor in downgrading the Augmentor Wing in the SAS#2 and NO-SAS modes. On several occasions, the pilots were unable to recover from a high-fast situation in time to land.

As indicated on Figure 5.2-1, the EBF model also had a predominantly upward thrust vector orientation; but the orientation retained a constant, slightly forward inclination for small speed changes around the trim point. The pilots had much less trouble controlling speed.

The differences in the Augmentor Wing and EBF characteristics reflect differences in the data base available for the simulation, rather than fundamental differences in the powered-lift concepts, see Section 1.0.

The thrust vector orientation can be modified, if desired, by using vectored thrust, flap modulation, etc. While η_T is known to be a very important parameter, the allowable limits are unknown. Factors to be considered are discussed separately for the two control techniques.

<u>η_T - Control Technique #1</u>

$\eta_T \approx 0$ is typical for a conventional jet transport. A thrust increase at constant attitude produces predominantly a forward acceleration with very little vertical acceleration. Consequently, thrust can be used to produce an immediate speed change without an immediate effect on γ. If the pilot chooses to convert the thrust change into γ rather than speed, he must change the aircraft attitude until the thrust change is balanced by the change in the gravity vector.

If the thrust vector also has a vertical component, then a thrust change will produce a change in n_z (hence, γ) unless the aircraft is rotated to a new angle-of-attack. The maximum upward inclination of the thrust vector will then be determined by the acceptable limits on the n_z and γ excursions associated with throttle inputs of the magnitude required for speed control. Both ride comfort and pilot workload should be considered.

The $\eta_T < 45°$ suggestion for Technique #1 is completely arbitrary. The angle certainly must be less than 90° in order to produce a forward acceleration. $\eta_T \approx 13°$ is known to be satisfactory from SST simulation experience. For $\eta_T > 45°$ the throttle gives more flight path response than speed response.

η_T - Control Technique #2

When contolling with Technique #2, a throttle input is used to initiate a flight path change, and the effective thrust vector is tilted, via pitch attitude, to control speed. Consequently, a vertical thrust vector orientation ($\eta_T = 90°$) is probably satisfactory with this technique. Larger angles ($\eta_T > 90°$) would be undesirable because of the deceleration that would accompany a climb command. The lower limit is unknown. However, the technique has been successfully used in conventional aircraft when on the back side of the drag curve. In this case, η_T corresponds to α plus or minus the inclination of the engines relative to the α reference line. The suggested $\eta_T > 13°$ may be a little on the high side, but provides for a conversion of 20% of the thrust change directly into n_z; (i.e., sin 13° = .2).

5.2.3 Control System Tested

This section does not contain any criteria; but describes the control systems used in the current study. Representative response time histories are included.

Two powered lift configurations were flown:

- o Augmentor wing
- o Externally blown flap

Each was flown with no stability and control augmentation system (SAS) and with two different concepts:

- o SAS #1 - for use with Control Technique #1
- o SAS #2 - for use with Control Technique #2

SAS #1 employs automatic flap and thrust modulation to force the STOL aircraft to respond more or less like an airplane, whereas SAS #2 improved the natural response of the STOL so as to improve controllability with Technique #2.

Figure 5.2-7 gives a verbal comparison of the system tested. Figures 5.2-8 and 5.2-9 are simplified block diagrams of the two SAS concepts. Note that both systems have attitude feedback and integration in the column command path.

Time responses are presented for the Augmentor Wing aircraft only.

Figure 5.2-10 compares the response to column and throttle inputs for the two types of SAS. Note that:

1. <u>For a column input:</u>
   - o SAS #1 produces a γ change while holding speed, and $\Delta\theta/\Delta\gamma = 1$.
   - o SAS #2 produces a speed change while holding γ.
2. <u>For a throttle input:</u>
   - o SAS #2 produces a γ change while holding speed.

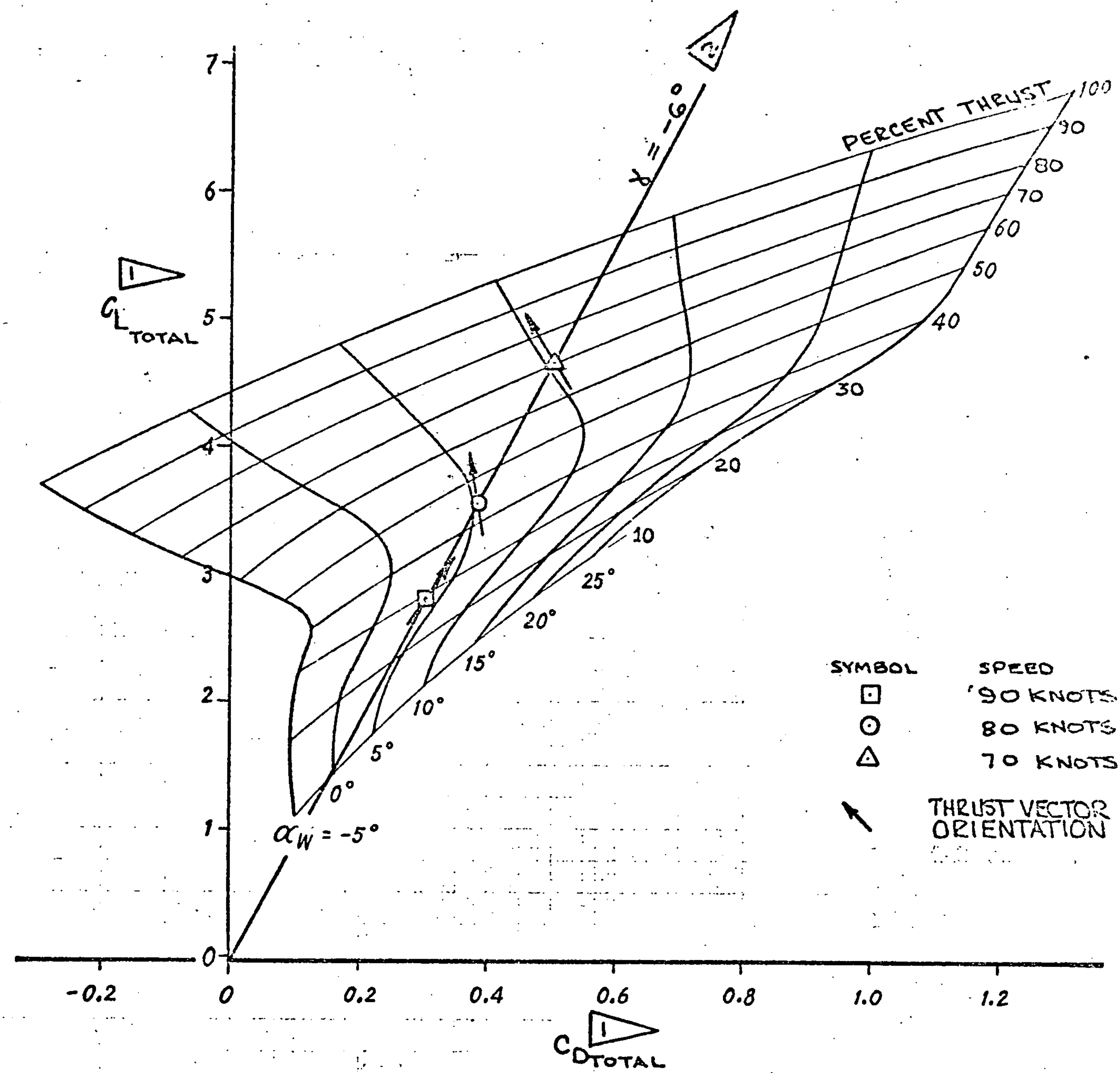
     $\Delta\theta/\Delta\gamma \approx -.5$ (non-linear, depends on trim data, see Fig. 5.2-3).
   - o SAS #1 produces a speed change while holding γ.

The blowing thrust ($\Delta F_{g}B$) and auxiliary flap (δ_{AUX}) time histories show how SAS #1 used these controls to:

- o quicken the n_z response. Feed-forward paths from the column were used to command an immediate thrust increase and flap extension for an aft column step. These combined DLC effects were greater than the elevator download, so there was no initial n_z reversal.
- o maintain $\Delta\theta/\Delta\gamma = 1$. Following the initial flap extension, feedbacks were used to retract the flaps as the maneuver progressed; thereby compensating for the increased lift which resulted from the increased thrust. The steady state thrust increase was, of course, required to hold speed as γ was increased.

Both of the systems were experimental in nature, and neither was optimized.

AUGMENTOR WING
4 ENGINES , $\delta_F = 70°$
$\delta_{STAB} = 0°$



NOTES:

[1] C_D & C_L INCLUDE ALL AERODYNAMIC AND THRUST FORCES IN STABILITY AXES

[2] TAN. $\gamma = -C_D/C_L$

MACK 16 OCT 71

TOTAL FORCE DATA, INCLUDING THRUST REACTIONS

D6-40409

FIG 5.2-6

TYPE OF SAS	PILOT CONTROL, TECHNIQUE REQ'D	INPUTS TO ENGINES AND CONTROL SURFACES	REMARKS
SAS #1 USES AUX-FLAP	#1 Conventional Jet Transport Technique: Column → θ → γ Throttles → V	δ_{Col}, δ_{THL}, α, $\dot{\gamma}$, V → Engines δ_{Col}, δ_{THL}, α, $\dot{\gamma}$, → Aux. Flap δ_{Col}, Q, θ → Elevator	Easy to fly PILOT RATINGS 1.5 → 3 No pilot re-training required. PROBLEM AREAS: o Quick THRUST & AUX. FLAP response required to achieve quick n_z response necessary for STOL. o Complex system o Failure would force pilot to switch to an unfamiliar control technique.
SAS #2 USES AUX-FLAP	#2 Unconventional: Column → θ → V Throttles → γ	δ_{THL}, γ, $\dot{\gamma}$ → Engines δ_{COL}, Q, θ, V → Elevator	Handling qualities require improvement for normal airline operation. PILOT RATINGS 3 → 8 AUG. WING 3 → 5 EBF PROBLEM AREAS: o Requires quick THRUST response o Requires pilot re-training o Sluggish speed control o Difficult to flare consistently
UNAUGMENTED	#2 Unconventional Column → θ → V Throttles → γ	δ_{THL} → Engines δ_{COL} → Elevator	Handling qualities unacceptable for routine operations PILOT RATINGS 6 → 8 AUG. WING 4.5 → 5 EBF PROBLEM AREAS as SAS #2 except more severe. In addition: o Poor pitch damping o Strong thrust/pitch interaction

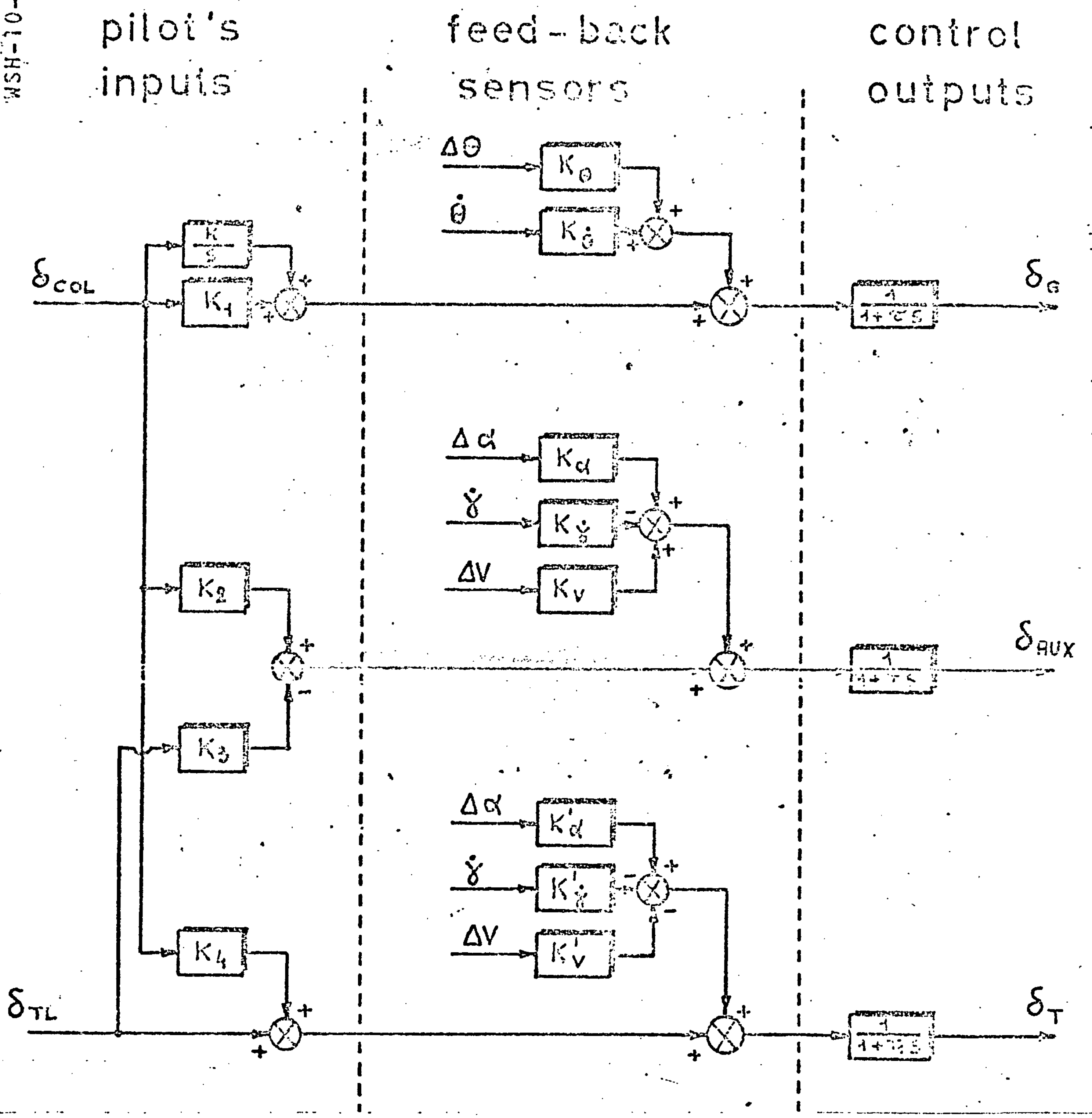
SUMMARY OF HANDLING QUALITIES EVALUATIONS

BOEING

ENGR.	CHECK	APR	APR
Allison			
2 DEC 71			

REVISED DATE

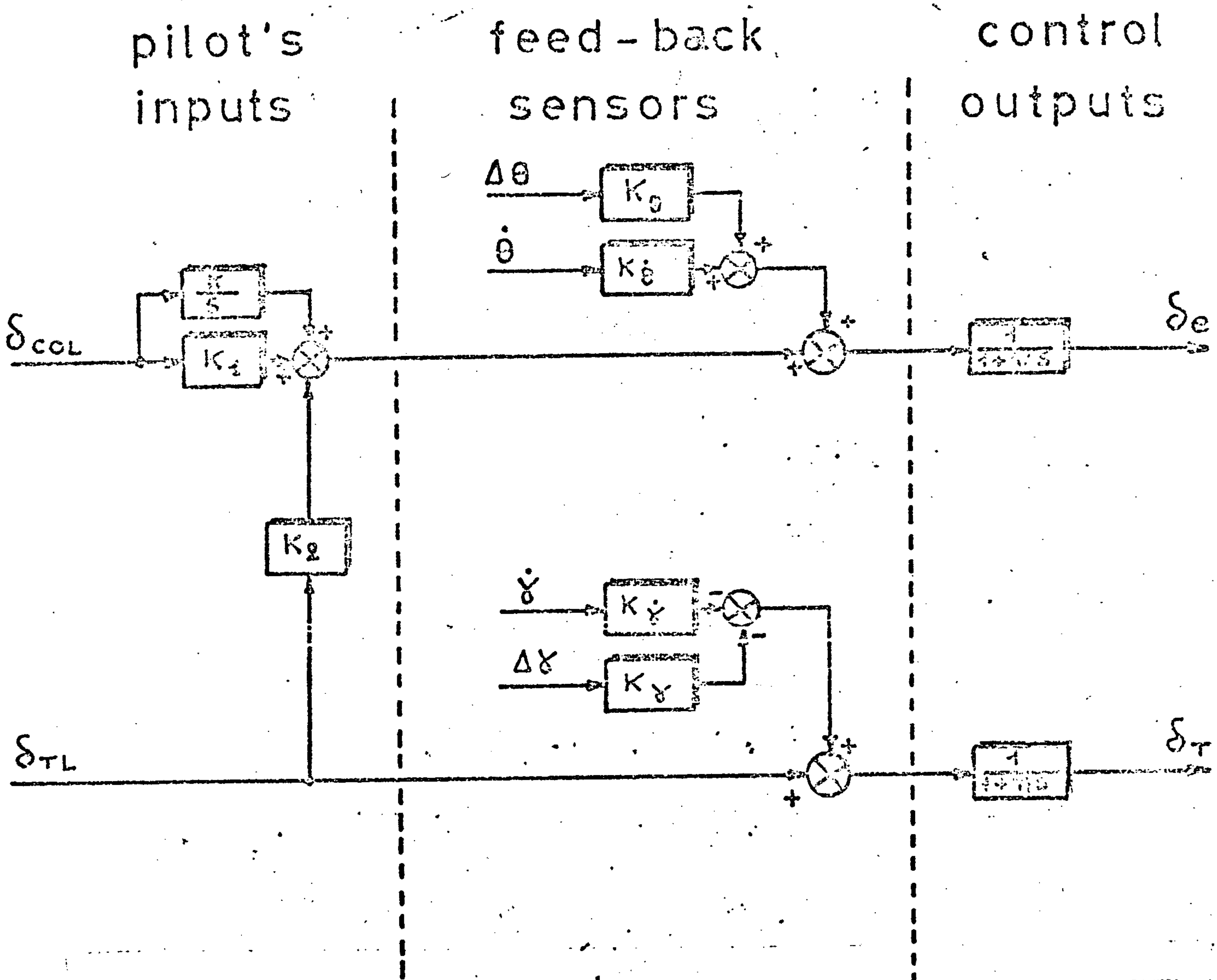
WSH-10-059



NOTE: 1. TO BE USED WITH CONTROL TECHNIQUE # 1

2. FLAPS MODULATED FOR D.L.C., TO CONTROL $\Delta\theta/\Delta\gamma$, AND TO CHANGE EFFECTIVE THRUST VECTOR ORIENTATION

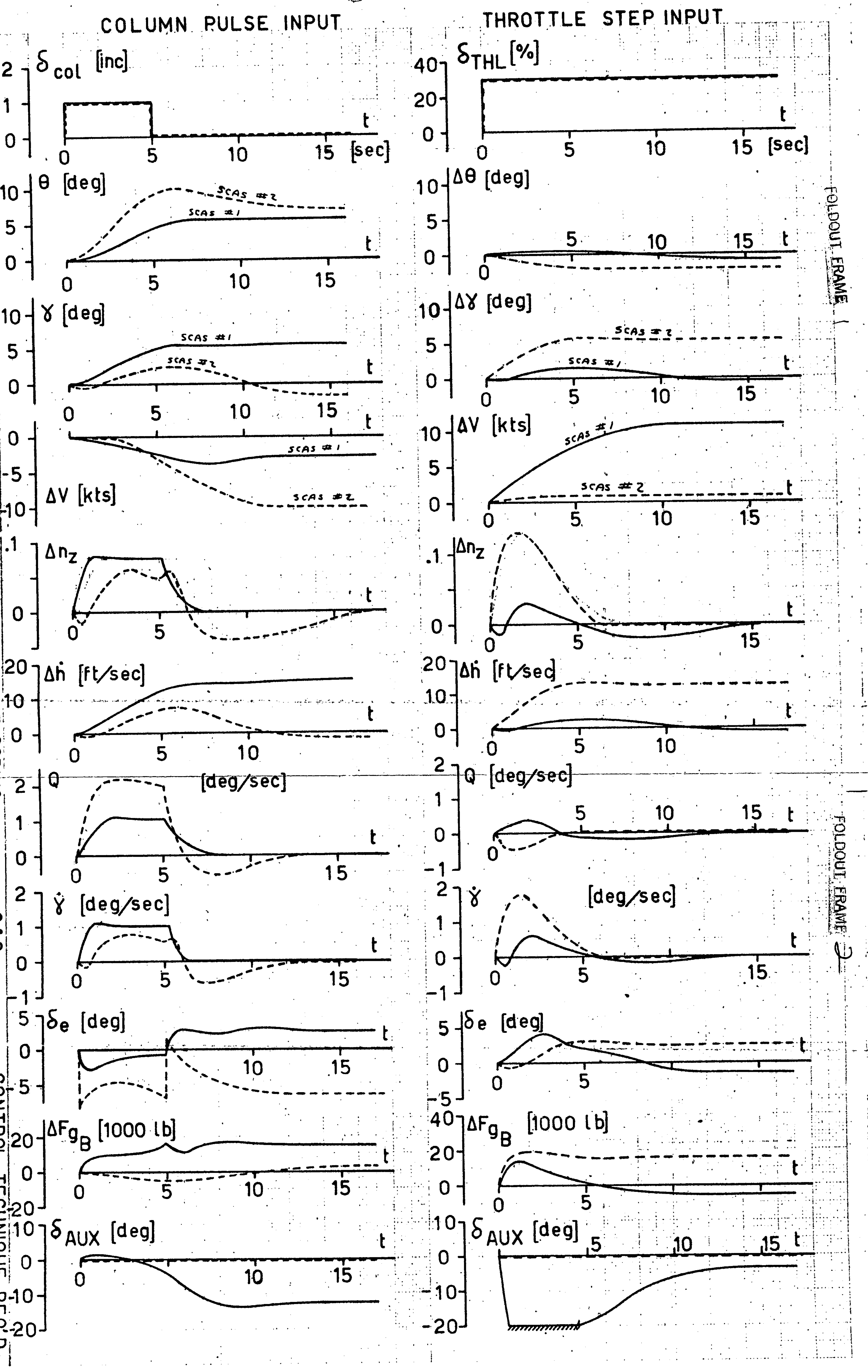
ENGR.	AVAGNINA	NOV 71	REVISED	DATE	SAS # 1	D6-40409
CHECK						FIG 5.2-8
APR						
APR					BOEING	PAGE 113



NOTE: 1. TO BE USED WITH CONTROL TECHNIQUE # 2

2. NO FLAP MODULATION

ENGR.	GWA	Nov 71	REVISED	DATE	SAS # 2	D6-40409
CHECK						FIG 5.2-9
APR						
APR					BOEING	PAGE 119



LINE CODING

SAS

—— #1 with aux. flaps

- - - #2 no aux. flaps

CONTROL TECHNIQUE REQ'D

#1 $\delta_{col} \rightarrow \dot{\theta} \rightarrow \gamma$; $\delta_{THL} \rightarrow V$

#2 $\delta_{col} \rightarrow \theta \rightarrow V$; $\delta_{THL} \rightarrow \gamma$

CALC	G.M. AVAGNINA 1/11/72
CHECK	
APR	
APR	

COMPARISON OF LONGITUDINAL RESPONSES FOR TWO S.A.S. CONCEPTS — AUG WING —

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FIG. 5.2.10

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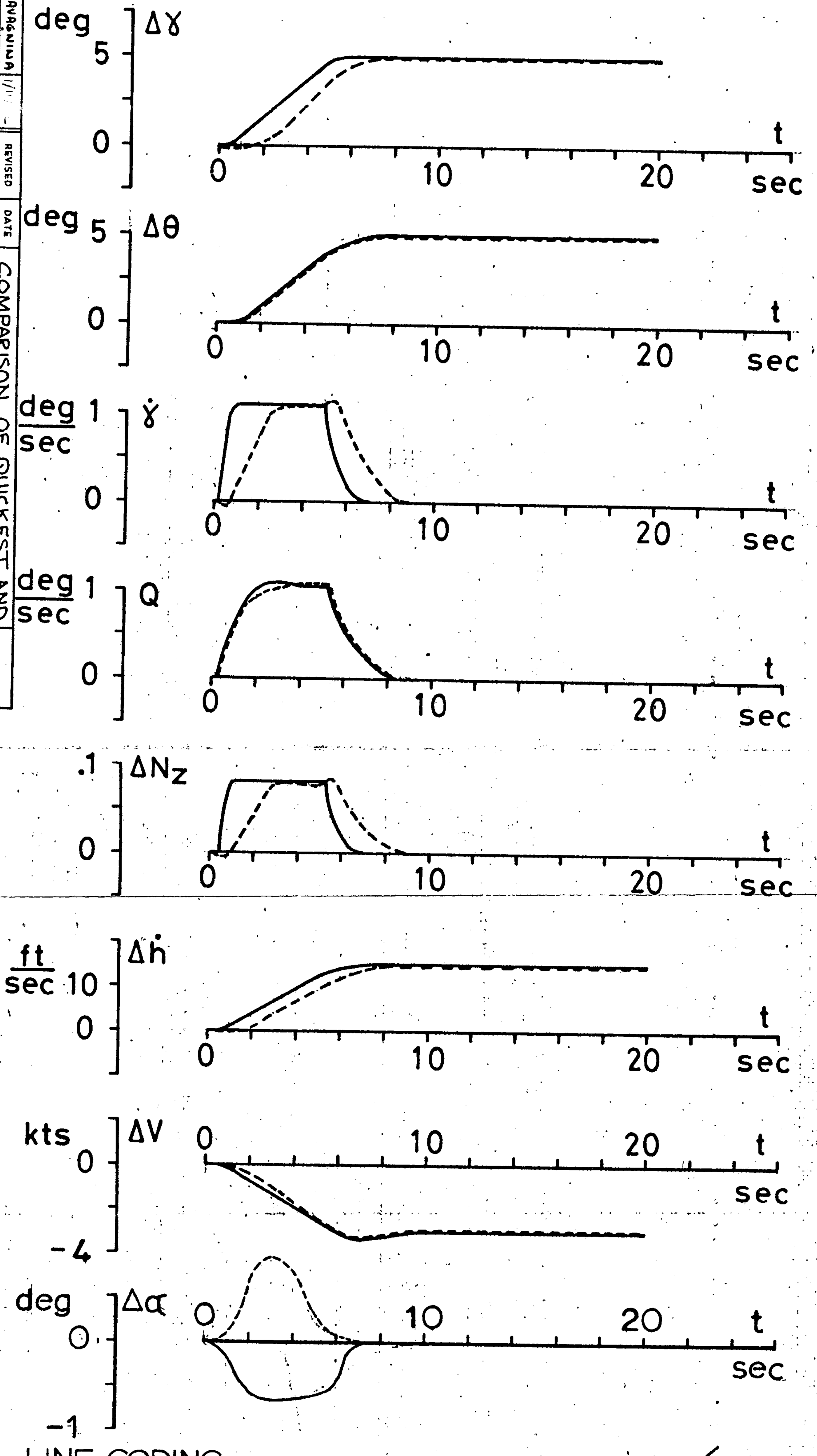
The systems were quite useful for making parametric variations in control characteristics. Due to the limited time available, very few of these variations could be tested and most were concentrated on SAS #1. The most significant parametric variations were:

- τ_{n_z}, the load factor response time constant. Within this document, τ_{n_z} is defined as the time elapsed from the initiation of the control input until n_z reaches 63% of the first peak magnitude. Figure 5.2-11 compares responses for the quickest ($\tau_{n_z} \approx .6$ sec) and most sluggish ($\tau_{n_z} \approx 2.0$ sec) values tested. The variations were achieved primarily by changing the feed forward gains from the column to thrust and flaps. The sluggish response was satisfactory for glide slope control, but was definitely unacceptable for landing on a short runway from a steep glide slope. The quick response was satisfactory for approach and landing.
- $\Delta\theta/\Delta\gamma$, previously defined and discussed in Paragraph 5.3.2. Figure 5.2-12 compares a response with $\Delta\theta/\Delta\gamma = 1.4$ to the two $\Delta\theta/\Delta\gamma = 1$ responses previously shown on Figure 5.2-11. Values of $\Delta\theta/\Delta\gamma = 0.5$, 1.0, and 1.4 were tested by varying the gains to the auxiliary flaps. The pilot favored the $\Delta\theta/\Delta\gamma = 1.4$ value for glide slope and flare controllability; but remarked that it produced a cockpit vision problem when large flight path changes (including flare) were made. The $\Delta\theta/\Delta\gamma = .5$ configuration was deficient in that there was insufficient θ lead information for control.

SAS #1 (with AUX. FLAPS)
RESPONSES TO A 5 SEC. COLUMN PULSE OF 1 IN.



LINE CODING

	τ_{n_z}	$\Delta\theta / \Delta\gamma$
———————— (solid)	.6 sec.	1.0 deg/deg.
- - - - - - - - (dashed)	2 "	1.0 "

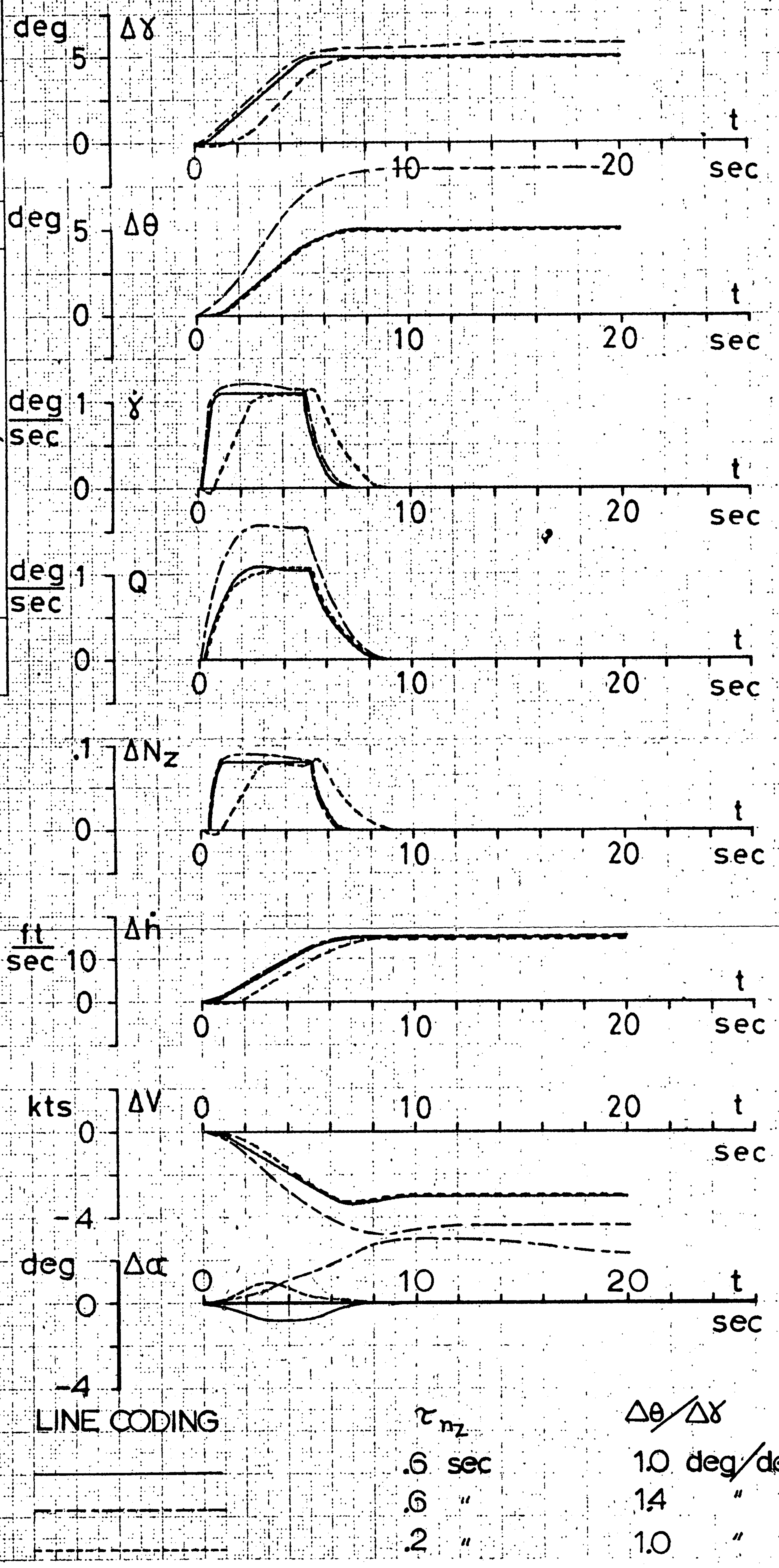
NOTE:
1. SIMILAR PEAK n_z AND ATTITUDE RESPONSE.
2. τ_{n_z} CONTROLLED VIA FLAP AND BLOWING MODULATION
3. τ_{n_z} = 2 SEC UNACCEPTABLE FOR STOL FLARE.
4. τ_{n_z} = .6 SEC SATISFACTORY.

COMPARISON OF QUICKEST AND MOST SLUGGISH RESPONSES TESTED —— SAS #1, $\Delta\theta/\Delta\gamma$ = 1.0

FIG. 5.2-11

CALC	H.M. AVAGNINA
CHECK	
APPD	
APPD	

SAS #1 (With AUX. FLAPS)
RESPONSES TO A 5 SEC. COLUMN PULSE OF 1 IN.



CALC	GM Avagnina 1/11/72
CHECK	
APPD	
APPD	
REVISED	
DATE	

EFFECT OF Δθ/Δγ RATIO VARIATION ON SAS #1 RESPONSE

D6-40409

FIG 5.2-12

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FOLDOUT FRAME

FOLDOUT FRAME 2

5.3 Pitch Control

5.3.1 Pitch Control Power

Criteria: The pitch control power available from a maximum column input shall meet the most severe of the requirements stated in Sections 5.3.1.1, 5.3.1.2, or 5.3.1.3. The static balance requirements (5.3.1.1 and 5.3.1.2) and maneuvering requirements are not additive. However, if the airplane is unstable with respect to angle of attack changes, it must be demonstrated that sufficient control power is installed to provide a restoring moment at the maximum angle of attack that could be reasonably expected during abusive type maneuvers. In this case, there would be a stability augmentation control power requirement in addition to the static balance requirement.

Discussion: Regardless of the augmentation system or the pilot control technique employed, pitch control power is required to assure a static balance of the pitching moments resulting from expected perturbations from the trim point. These requirements are stated in Sections 5.3.1.1 and 5.3.1.2; and include the moments resulting from gusts.

In addition to the static balance requirements, there are pitch control requirements which are dependent on the control technique employed; i.e., whether θ is used to control γ or V. These requirements are stated in Section 5.3.1.3.

The static balance and control requirements are specified as being non-additive because the static balance requirements include the most severe upsetting moments that could reasonably be expected in still air, combined with the maximum upsetting moment expected in turbulence. Since peak gusts are of short duration, it is considered unreasonable to apply maneuvering requirements in addition to the requirements for static balance in turbulence.

If the unaugmented airframe does not have a stable static margin, the control power requirements will probably have to be increased in order to preclude uncontrollable pitch-up during abusive maneuvers such as collision avoidance.

5.3.1.1 Static Balance Out Of Ground Effects

Criteria: With the trim set for approach, the pitch control must be able to provide a static balance, out of ground effects, of the pitching moments resulting from any allowable thrust change combined with speed changes in the range from stall warning speed up to V_{APP} + 15 kts, combined with a step gust from the most critical direction with a velocity of three times the RMS velocity of the design turbulence (3σ gust), combined with either of the following:

- any allowable configuration change, or
- turns and pull-ups to stall warning.

When interpreting this requirement, conditions such as thrust reduction or flap retraction at V_{MIN} need not be considered, as these are not "allowable" inputs. Allowable inputs are gear and flap retraction for go-around, etc.

Discussion: Configuration changes are not combined with the turns or pull-ups because configuration changes are normally made in 1 g flight, and the speed variations required should assure an adequate control margin for small load factor disturbances.

Maneuvers to stall warning are specified rather than to "stall" (α_{Max}, etc.) because it is unlikely that a 3σ gust would be encountered in conjunction with a maneuver outside of the normal flight envelope.

5.3.1.2 Static Balance in Ground Effect

Criteria:

With trim set for approach, the pitch control must be able to provide a static balance, in the most adverse ground effect, of the pitching moments resulting from the most adverse combination of:

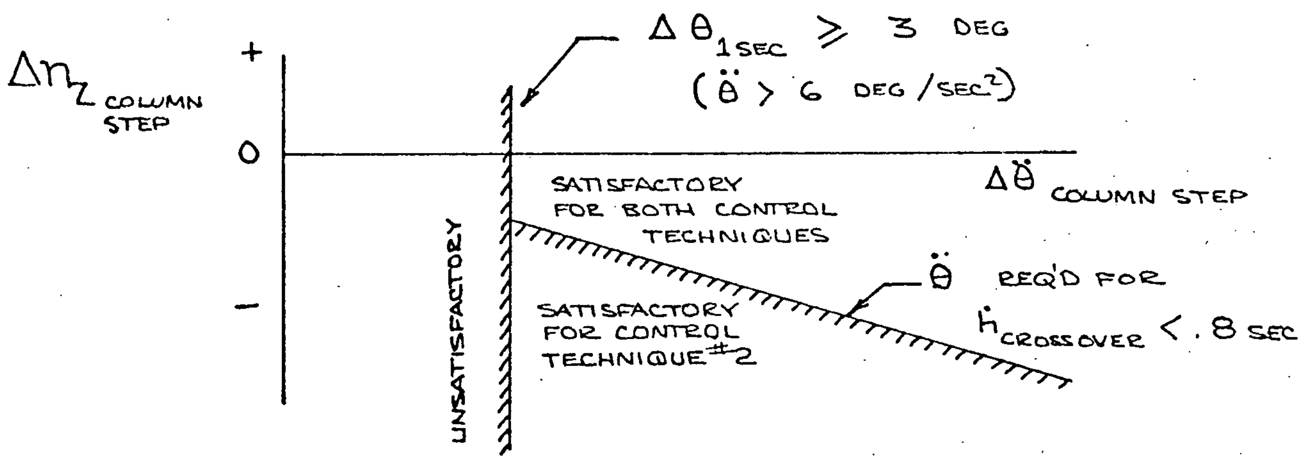
- angles of attack up to $\alpha = \alpha_{Max}$, and down to the lesser of $\alpha_{Approach}$ or $\theta_{Ground\ Roll}$

- o speeds in the range of V_{Min} to 1.1 V_{App}
- o Thrust in the range from flight-idle to go-around thrust

Discussion: During a late flare, the pilot may control into the "stick shaker" (α_{Warn}) in order to prevent a hard landing. The requirement for a static balance of pitching moments to α_{Max} assures an excess of control power at "stick shaker". Control capability to the "geometry limit" was not specified because the α_{Max} requirement is more severe; i.e., α may exceed $\alpha_{Geometry\ Limit}$ when the aircraft is descending during the flare.

5.3.1.3 Maneuvering

Criteria: With the aircraft trimmed for approach at approach speed, the pitch acceleration resulting from a full aft column step shall be within the satisfactory region indicated in the sketch below.



The Δn_z and $\ddot{\theta}$ terms refer to the initial peak acceleration (at $t \approx 0$) resulting from a column step input. The Δn_z term includes the change in lift resulting from actuation of the elevator and direct lift control (DLC) devices.

For an aircraft intended to be flared by means of column inputs only (Control Technique #1), $\ddot{\theta}$ must be adequate to provide a positive rate of climb ($\dot{h}$) increment within 0.8 seconds after an aft column step input of the magnitude normally used for flare. The effects of DLC devices should be included in determining the $\ddot{\theta}$ requirement.

Regardless of the control technique or the DLC capability, it must be possible to achieve a 3 degree attitude change within one second after a full aft column step.

The $\ddot{\theta}$ requirements apply with the aircraft trimmed for symmetric flight, but controlled into the most critical steady sideslip required for engine-out or crosswind conditions.

The requirements should be satisfied in the most severe ground effects encountered at altitudes down to 1/2 of the nominal flare altitude.

Discussion: Pitch attitude changes will be required either to control flight path (Technique #1) or airspeed (Technique #2). Requirements for the two techniques will be discussed separately.

Technique #1 - One fundamental requirement for satisfactory flight path control is the sink rate crossover time ($t_{\dot{h}}$), defined as the time required to achieve a positive change in rate of climb following an aft column step input. For a conventional airplane with elevator control, the crossover time is usually approximated from

$$t_{\dot{h}} = \sqrt{\frac{6\, I_{YY}}{q S C_{L_\alpha}} \left\{ \frac{-C_{L_{\delta_e}}}{\bar{c}\, C_{m_{\delta_e}}} \right\}}\ .$$

For the more general cases of an aircraft with powered lift effects and DLC surfaces, the C_{L_α} and the $C_{L_{\delta_e}}/C_{m_{\delta_e}}$ terms can be modified to include these effects. With a little manipulation the crossover time equations can be solved in terms of the $\ddot{\theta}$ required to produce a desired $t_{\dot{h}}$ as follows:

$$\Delta \ddot{\theta}_{\substack{\text{COLUMN}\\ \text{STEP}}} = \frac{-6}{(t_{\dot{h}})^2} \left\{ \frac{\Delta n_{z\,\text{COLUMN STEP}}}{n_z/\alpha} \right\} \qquad \text{EQN:- 5.3.1.3}$$

where all DLC and blowing effects are included in the Δn_z and n_z/α terms.

As discussed in 5.4.4, $t_{\dot{h}} \leq .8$ sec. is desired. After decreasing $t_{\dot{h}}$ by the amount of anticipated control system lags, the required $\ddot{\theta}$ can be computed from equation 5.3.1.3. Note that $\Delta n_{z_{column\ step}}$ is negative for an elevator, which requires a positive $\ddot{\theta}$. If DLC is used, $\Delta n_{z_{column\ step}}$ could become zero or positive. In this case, the $\Delta\theta_{1\ sec} = 3$ degree requirement would apply.

The $\Delta\theta_{1\ sec} = 3$ deg requirement is based on the following:

- o Consistent with Table 2.1 of Reference 7 which recommends $\ddot{\theta}_{MIN} = .05 - .2$ Rad/Sec^2 for STOL and $\ddot{\theta}_{MIN} = .1 = .4$ Rad/Sec^2 for hover.
- o Necessary in order for $\dot{\theta}$ to lead n_z with the recommended n_z response time and θ/γ ratio. (See 5.2.2).
- o Adequate for flare. Some STOL aircraft may require large attitude changes during the flare to arrest the sink rate and to avoid striking the nose gear. The required $\ddot{\theta}$ would depend on the configuration, but $\ddot{\theta} \approx 6$ deg/sec^2 does not appear unreasonable.

In the present study with a system that uses DLC to meet the crossover time requirement, it was noted that a maximum pitch acceleration of 3 deg/sec^2 was used by the pilot to flare in turbulence. However, an additional $\ddot{\theta}$ increment to account for speed deviation below approach speed is desirable.

<u>Technique #2</u> - When θ is used to control speed, rapid θ response is required to achieve adequate speed response. In addition, it is necessary to control the thrust/pitch interactions and to rotate in the flare. The θ requirements for STOL operation using this technique are expected to be similar to the VTOL requirements of References 7 and 8.

The Reference 8 requirement for $\Delta\theta_{1\ sec} \geq 3$ deg was used since this format includes the effects of control system lags. Assuming no lags and a constant acceleration for the first second, $\ddot{\theta} \geq 6$ deg/sec^2 would be required; which is consistent with the .1 to .3 rad/sec^2 requirement of Reference 7.

During the current study, one configuration was flown (with Technique #2) for which control power was limited by the column/elevator gearing to $\ddot{\theta}_{Max} \approx 5$ deg/sec^2. The pilot used full column when attempting to hold airspeed during a flare in turbulence, and felt the control power was inadequate.

5.3.2 Pitch Control Sensitivity

Criteria: Pitch control sensitivity in terms of $\ddot{\theta}/\delta_{Col}$ should meet the requirements set forth below, with the aircraft trimmed for approach at the approach speed.

Control Technique #1 - Pitch control sensitivity should be set to meet the $\ddot{\theta}$ requirements of 5.3.1.3 in combination with the n_z/δ_{Col} requirements of 5.4.3.1.

Control Technique #2 - $\ddot{\theta}/\delta_{Col} \geqslant 3$ (deg/sec^2)/inch is required.

Discussion:

Control Technique #1 - In a conventional unaugmented airplane, pitch control sensitivity is determined by the column/elevator gearing. As discussed in Reference 3, the relationship between $\ddot{\theta}/\delta_{Col}$ and n_z/δ_{Col} for a second order representation of a stable airplane is defined by the short period frequency ($w_{n_{sp}}$) and the load factor response to α (n_z/α), i.e.,

$$\frac{\ddot{\theta}}{\delta_{COL}} \times \frac{\delta_{COL}}{n_z} \approx \frac{\omega_{n_{sp}}^2}{n_z/\alpha}$$

When augmentation systems are used, the response to control need not be second-order; and the initial acceleration can be controlled independently of the steady state load factor. Both the command paths and the feedback paths may contain shaping functions. The n_z/δ_{Col} requirements will establish the steady state gains. Direct lift control and elevator gearing can then be traded to meet the initial $\ddot{\theta}/\delta_{Col}$ requirement.

Control Technique #2 - The $\ddot{\theta}/\delta_{Col}$ requirements for STOL flight in this mode are expected to be similar, but a little less severe than VTOL hover requirements. As shown in Reference 8, longitudinal acceleration during hover can be approximated from the change in inclination of the thrust vector; i.e.

$$\Delta \dot{u} \approx -g\theta$$

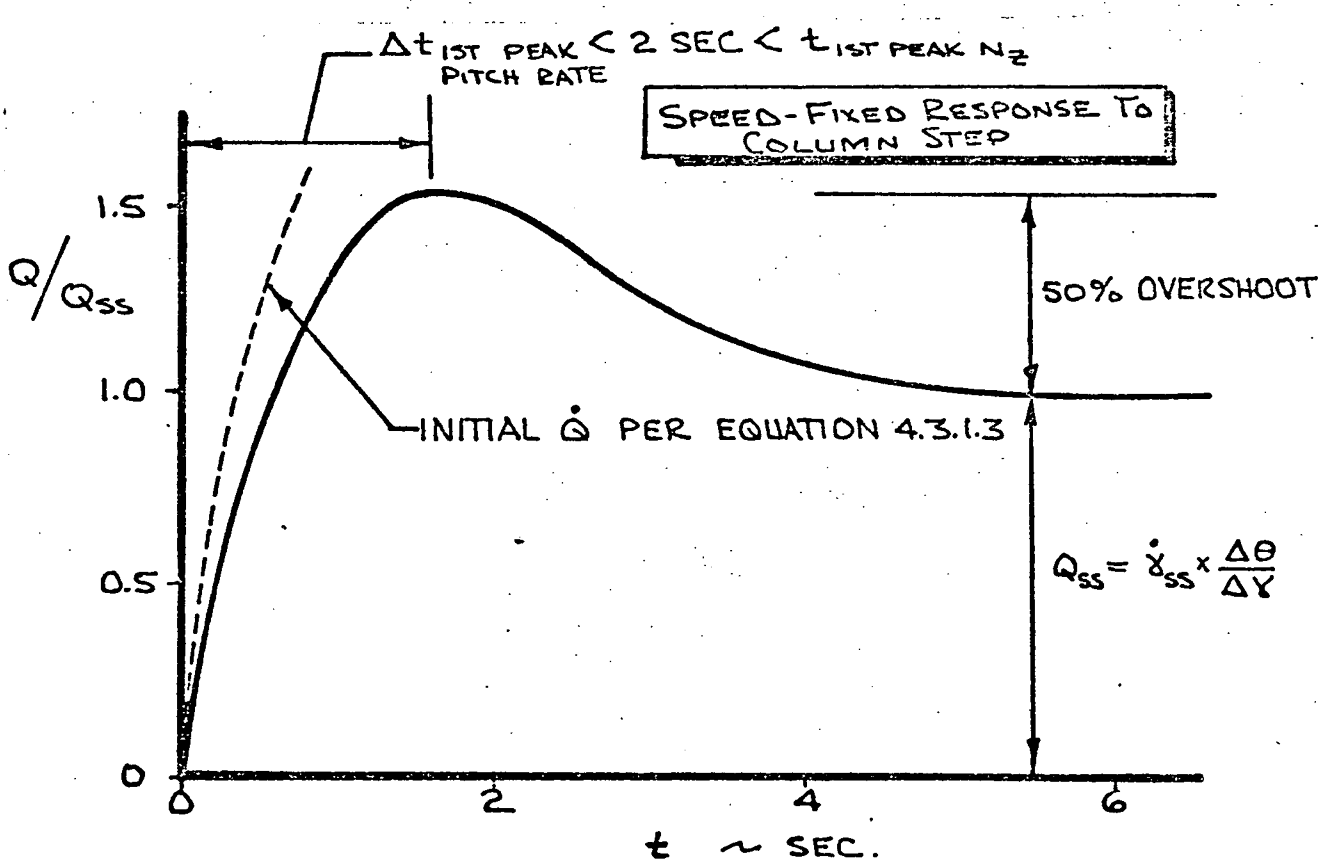
STOL aircraft operate at speeds well above hover, and a significant longitudinal force is also obtained from the change in aerodynamic drag ($C_{D_\alpha} \cdot \Delta\alpha$). In addition, the STOL task normally requires holding the airspeed as opposed to the VTOL requirement for quickly changing airspeed in order to translate or stop over a desired area. Consequently, the $\ddot{\theta}/\delta_{Col}$ requirement for STOL was chosen to be a little lower than the Reference 7 requirement for VTOL, and to meet the minimum control power requirement of 5.3.1.3 with about a 2-inch column deflection.

In the current study, $\ddot{\theta}/\delta_{Col}$ sensitivities of 0.8 deg/sec^2/inch and 3 deg/sec^2/inch were tested with systems designed for control technique #2. With the lower sensitivity, the pilot used full column (6 inches) when attempting to hold airspeed during a flare in turbulence; and downrated the system due to inadequate control power. With the higher sensitivity, attitude response was considered satisfactory.

5.3.3 Pitch Rate Response

<u>Criteria:</u> Pitch rate response to a column step input should be generally compatible with the frequency and damping requirements of the longitudinal dynamic stability criteria of Section 6.2, herein. However, it is not necessary for the pitch rate response to be second order, provided it is smooth and has no unusual characteristics that would detract from controllability or passenger comfort. The response shape sketched below is considered to represent a desirable response, and is presented as a guide.

<u>Control Technique #1</u> - For a wings-level pullup conducted at approximately constant airspeed; and with n_z/α typical of CTOL transports at landing approach, the pitch rate response to a column step should have the characteristics indicated below:



Control Technique #2 - The pitch rate response to a column step input should be similar to the response shown above for Technique #1, except:

- o the requirement applies with speed varying
- o the initial $\dot{Q}$ and Q_{SS} requirements do not apply.

Discussion: Unlike the case of a fighter gunnery task where precise attitude control is important, per se, attitude control during landing approach is important only as it relates to the ability to control speed and flight path. Consequently, the response must be quick enough to give good sink rate control (Technique #1) or airspeed control (Technique #2). Yet it must be sufficiently smooth and predictable so that the pilot can put the nose approximately where he wants it and keep it there while waiting to see if he gets the desired rate/speed response.

Pitch response criteria for airplanes are frequently defined in terms of the frequency and damping of an equivalent second order system which represents the pair of roots (real or complex) dominating the short term angle of attack response. The equivalent second order system requirements of Reference 8 have been prescribed in Section 6.2, herein, as dynamic stability criteria for the response to external disturbances. However, there is no reason to require the control response of an augmented aircraft to approximate a second order response if a better response can be obtained by command shaping. The design criteria used for the Supersonic Transport control system approached this problem by defining an allowable pitch rate response envelope and a desired response shape for several flight conditions.

Control Technique #1 - The desired response shown for Technique #1 is similar to the SST low speed requirement, which was shown to be near optimum for that vehicle by extensive fixed-base simulation studies. However, the optimum shape is a function of several configuration-dependent variables, including n_z/α and the effective tail length. These two effects have been partially accounted for by prescribing an initial slope, $\dot{Q}$, that will give a satisfactory sink rate crossover time.

Control Technique #2 - Since attitude changes are not required to initiate flight path changes, the initial $\dot{Q}$ requirement to obtain the desired $\dot{h}$ crossover time, and the relationship of $\dot{Q}$ to $\dot{\gamma}$ do not apply. However, a fairly quick attitude change is required for good speed control.

Current Study Results - The shape of the pitch rate response was similar for both control system concepts tested. It can be seen from Figure 5.2-11 that the pitch rate response criteria recommended herein were not met in that:

- o the overshoot was less than 50%
- o with SAS #1, the first peak of $\dot{\gamma}$ occurred prior to the first peak of Q (this was achieved through powerful DLC)

While the pilots felt the pitch response was generally satisfactory, compliance with the criteria would probably have corrected two of the reported deficiencies:

- o more attitude "lead" information was desired for flight path control with SAS #1
- o attitude control was considered sluggish with SAS #2.

5.4 Flight Path Control

5.4.1 Glide Slope Control Power

5.4.1.1 Incremental Normal Acceleration

Criteria: The load factor available at stall warning shall equal or exceed the greater of the values shown below. The requirement applies at approach speed, in free air, with trim set for approach, and with a thrust setting not exceeding the thrust required for constant speed in the maneuver.

$$n_{z_{\text{glide slope}}} \geq n_{z_{\text{flare}}} \quad \sim \text{ see 5.4.2}$$

$$n_{z_{\text{glide slope}}} \geq \frac{1}{\cos \phi_{max}} + .1 \text{ g}$$

where

$$\underset{\text{(Deg)}}{\phi_{max}} = \underset{\text{(Knotts)}}{.25 \, V_{APP}} \approx \phi \text{ required for } \dot{\psi} = 3 \text{ °/sec turn rate @ } V = 1.15 \, V_{APP}$$

Discussion: The above criteria are similar to the vertical flight control requirements of Reference 7, but differ in several respects. The n_z requirements stated above refer to the capability at stall warning at the trimmed thrust setting, rather than to the maximum n_z available at maximum thrust. This was done to assure that stall warning would not be encountered during normal maneuvers at the power settings normally used in those maneuvers.

"Normal" maneuvers for landing approach are shallow turns and the landing flare. The load factor increment required for turns was specified in terms of the turn rate, so as to provide a rational variation of the n_z requirement as a function of approach speed. The choice of a standard 3 deg/sec turn rate at speeds up to 1.15 V_{App} is arbitrary; but the resultant bank angles appear reasonable. Note that the maximum required bank angle is 30 degrees at a 120 knot approach speed, which is consistent with current CTOL operating practice. During the current study, the pilots felt uncomfortable when the bank angle exceeded 20° at 80 knots, which correlates exactly with the criteria.

The arbitrary allowance of .1 g was added to the steady turn requirement to account for typical perturbations about the nominal value.

Unlike Reference 7, no distinction is made between the n_z resulting from thrust and the n_z due to α. It is felt that the basic requirement is to have sufficient n_z capability, regardless of the means of obtaining it; and that a criteria presented in this manner will be applicable to any configuration from a pure airplane to a pure vectored-thrust machine.

5.4.1.2 Incremental Flight Path Angle

Criteria: Considering the design glide slope under no-wind conditions to be the nominal flight path angle (γ); the constant-speed glide slope modulation capability shall not be less than the following:

Steeper: $\Delta\gamma = -2° \text{ @ } V_{App} + 10 \text{ kts}$

Shallower: $\Delta\gamma = \Delta\gamma_{\text{Still Air}} + \Delta\gamma_{\text{Design Headwind}}$

where: $\Delta\gamma_{\text{Still Air}}$ = the greater of:

- $+2° \text{ @ } V_{App} - 10 \text{ Kts}$
- $20\,(\partial\gamma/\partial V) \text{ @ } V_{App}$ — $\begin{cases}\gamma \text{ in degrees} \\ V \text{ in knots}\end{cases}$

and $$\Delta\gamma_{\text{Design Headwind}} = \gamma \times \frac{V_{wind}}{V_{App}}$$

Discussion: When approaching on the back side of the drag curve, the ability to make glide slope corrections becomes a strong function of the speed deviations and the slope of the γ vs V curve. The $\Delta\gamma = \pm 2°$ criteria of References 5 and 7 have been modified to account for ±10 knot speed variations from the approach speed.

This requirement was established by limiting flight-idle thrust and maximum thrust, and requiring the pilot to recapture the glide slope from high/fast and low/slow conditions. Glide slope deviations of 0.7 degrees, combined with 10 knots speed deviations, were chosen arbitrarily. The pilot felt that deviations of this magnitude represented realistic worst cases for attempting to salvage a bad approach, provided recapture was initiated above an altitude of 500 feet.

The thrust limits tested are shown in Figure 5.4-1, along with the pilot comments. These comments appeared to substantiate the $\Delta\gamma = \pm 2°$ requirement if the $\Delta\gamma$ capability was measured at the limits of the speed deviations. Figure 5.4-2 shows the thrust limits corresponding to the recommended criteria. These limits were then tested in turbulence (RMS = 6.5 ft/sec) and in headwinds and tailwinds with wind shear. The pilot flew a normal approach with no intentional offsets. The limits were further verified by tracking a 1° steeper than normal glide slope on instruments to an altitude of 100 feet at a speed 10 knots below approach speed; and then recovering from this situation to flare and land on the runway.

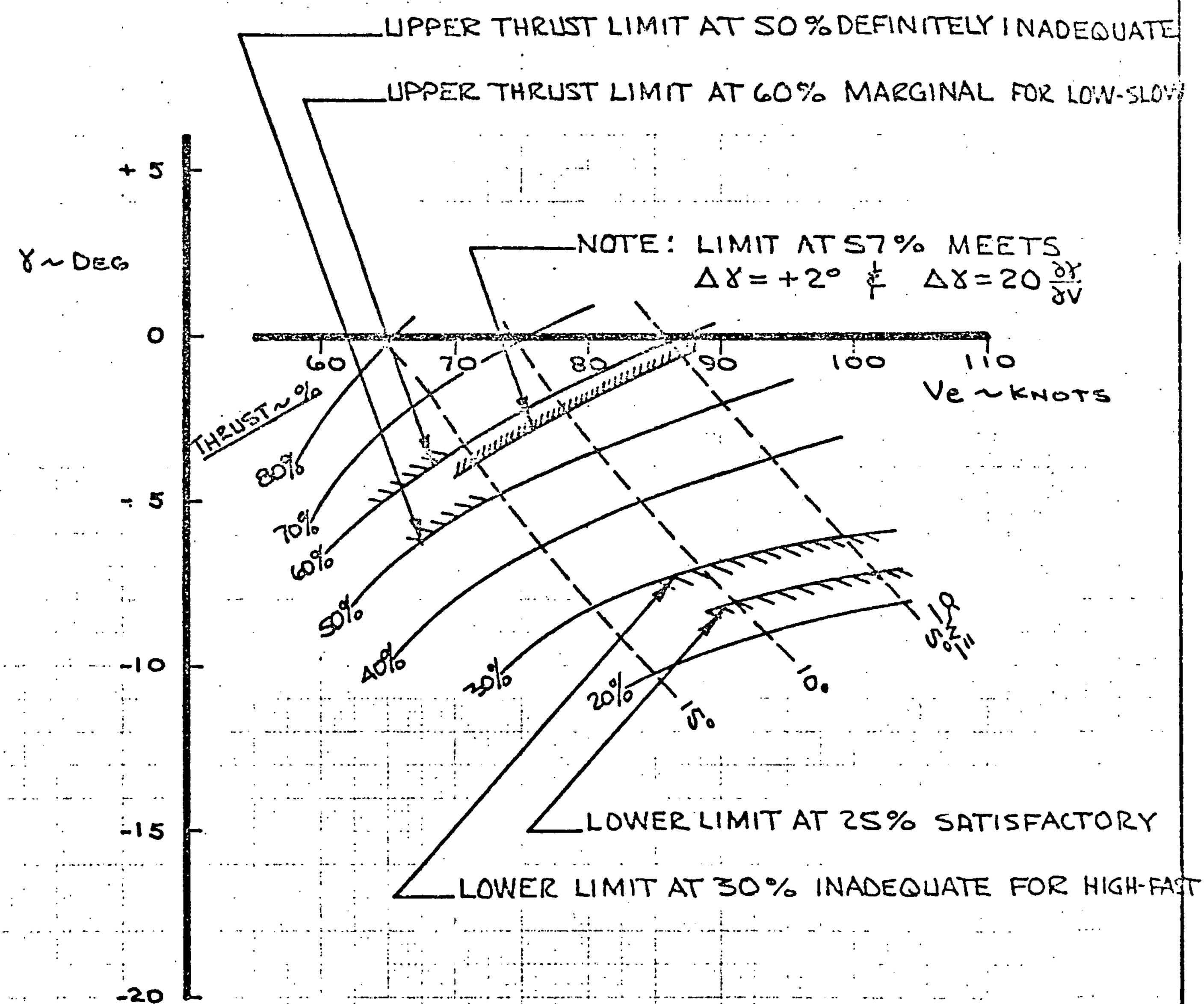
A tailwind allowance was not added to the "steeper γ" requirement because the required descent capability may be a prime factor in selecting the flight-idle thrust setting. It is desirable to keep flight idle as high as possible on powered lift configurations in order to minimize the likelihood of inadvertent stalls resulting from throttle abuse. The $\Delta\gamma = -2°$ requirement will handle expected tailwinds on a good approach at typical STOL speeds. Failure to regain glide slope from a high/fast situation in a tailwind will result in a go-around; but will not compromise safety.

The case of headwinds is entirely different in that headwinds are the rule rather than the exception, and the ability to regain glide slope from a low/slow situation is mandatory for flight safety.

The requirement for a positive $\Delta\gamma$ capability of at least 20 $(\partial\gamma/\partial V)$ was based on a prior unpiloted response study of recovery from longitudinal gusts. The criteria are intended to provide a recovery capability comparable to current CTOL transports. This requirement will probably set the positive

$\Delta\gamma$ capability for very unstable values of $\partial\gamma/\partial V$ whereas the $\Delta\gamma = +2°$ requirement will predominate at more stable values. For the configurations tested in the current study, approximately the same $\Delta\gamma$ capability resulted from both criteria.

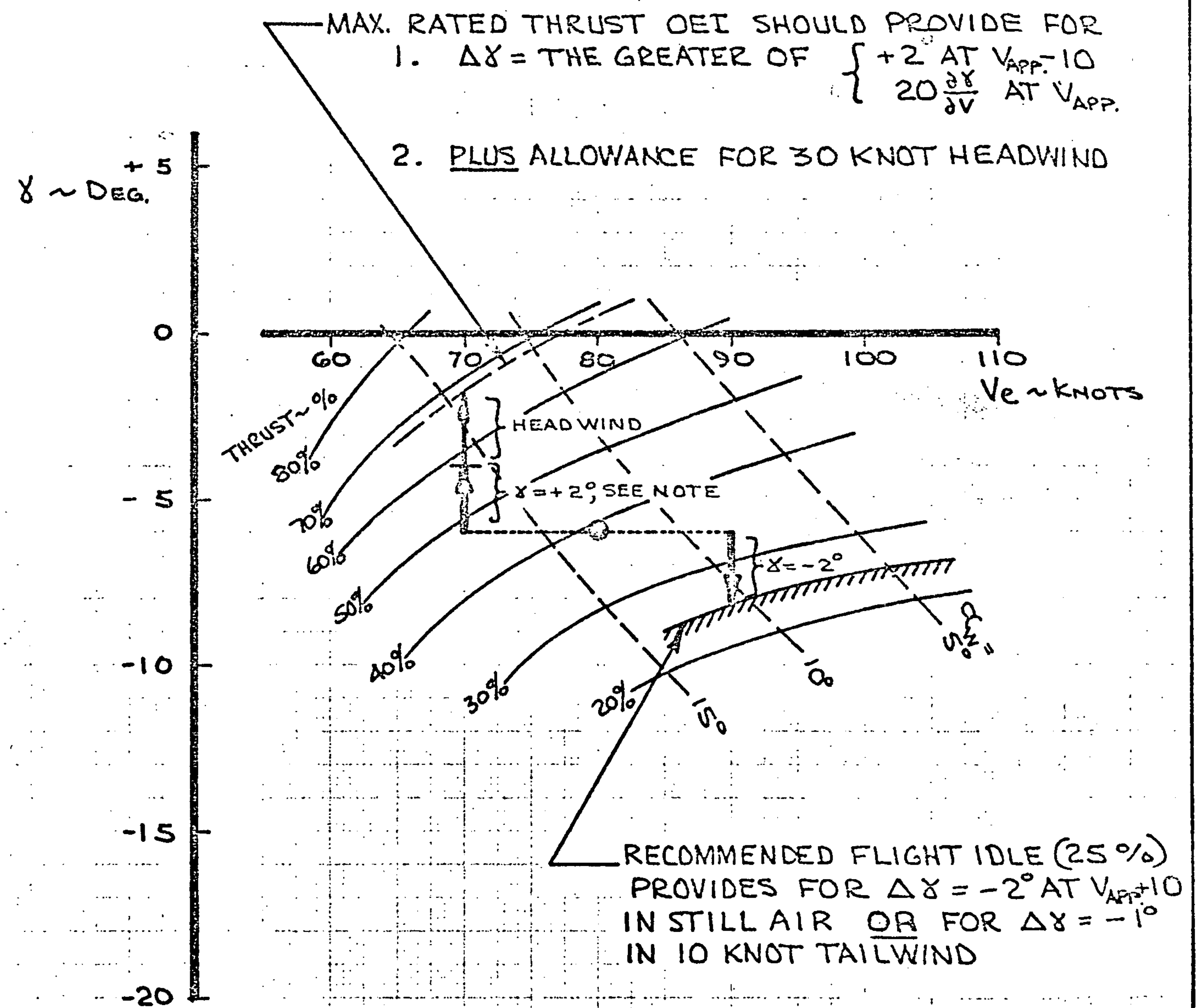
PILOT COMMENTS FOR STILL-AIR EVALUATION



CALC	ALLISON	1/4/72	REVISED	DATE	EVALUATION OF GLIDE SLOPE CONTROL WITH LIMITS ON THRUST ~ EBF, STILL AIR		D6-40409
CHECK							FIG 5.4-1
APR							
APR					THE BOEING COMPANY		PAGE 139

THRUST REQUIREMENTS FOR GLIDE SLOPE CONTROL

NOTE: FOR THIS CONFIGURATION THE SAME THRUST INCREMENT IS REQ'D TO MEET THE $\Delta\gamma = +2$ & THE $20\frac{\partial\gamma}{\partial V}$ REQ'M'TS



CALC	ALLISON	1/7/72	REVISED	DATE	THRUST REQ'M'TS FOR GLIDE SLOPE CONTROL RELATED TO TRIM DATA ~ EBF	D6-40409
CHECK						
APR						FIG. 5.4-2
APR					THE BOEING COMPANY	PAGE 140

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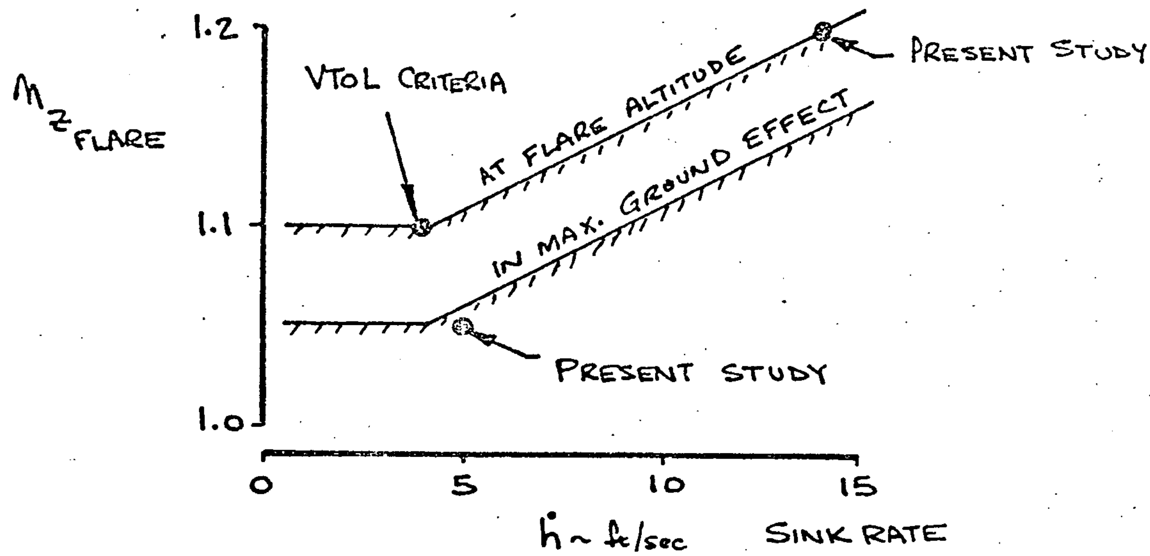
5.4.2 Flare Control Power

5.4.2.1 Incremental n_z

Criteria: The load factor (n_z) available at stall warning shall not be less than the values shown below. The requirement applies at approach speed, with trim set for approach, with ground effects corresponding to the altitudes shown, and with a thrust setting not exceeding the thrust required for constant speed in flare.



Discussion of Requirement at Flare Altitude: CTOL transports typically have a 1.3 g load factor (n_z) capability in free air, at stall warning, at approach speed; and the n_z capability tends to increase in ground effect. This flare n_z capability results from the approach speed margin which is selected to satisfy several requirements, and exceeds the n_z normally required for flare of a CTOL plane.

STOL transports approaching at sink rates comparable to CTOL will use higher flare load factors due to the more abrupt flare required by the closer proximity between glide slope transmitter and the aim point. The STOL flare requirement is less than current CTOL transport capability because it reflects only the flare control requirement, independent of other approach margin considerations.

The flare requirement is presented as a function of approach sink rate for three reasons:

- o To blend into the VTOL height control requirement of References 7 & 8
- o To be consistent with the $n_z = 1.1$ g requirement for glide slope control at nominally constant sink rate
- o To reflect the increased load factor requirement for arresting the sink rate as V_{APP} and/or γ_{APP} are increased.

As discussed under Section 5.4.4.1, n_z control in an airplane or STOL aircraft is analogous to vertical path control in a VTOL. In both cases, n_z is used to control sink rate. The n_z required to flare depends on the sink rate change required, and the time available to make the change.

The VTOL data on Figure 5.4-3 indicate that the n_z requirements increase as the vertical damping, z_w, decreases. Reference 8 points out that, for a VTOL craft,

$$z_w = \frac{g}{wo}\left(\frac{T}{W} - 1\right)$$

where wo represents the vertical velocity, $\dot{h}$. Since (T/W-1) is the same as Δn_z for a VTOL, a line of constant sink rate is a straight line in the z_w vs n_z plane. Slopes corresponding to several sink rates are shown on Figure 5.4-3 for reference. When these slopes were matched to the pilot opinion date of Reference 8 as shown on Figure 5.4-3, the requirements appeared to support the proposed variation of $n_{z_{flare}}$ with sink rate, although the levels required were a little less conservative than those of this section.

The flare load factor was recorded during all approaches. It was found that Δn_z rarely exceeded .2 g in the flare. The maximum observed was .25 g. The current study was conducted with an 80 knot approach speed on a 6 degree glide slope, which gave a sink rate of 14 ft/sec.

As sink rate is reduced, the load factor requirement is also reduced.

Discussion of Requirement in Ground Effect: Proximity to the ground can greatly reduce the maximum lift capability of a powered lift STOL aircraft. It is conceivable that some configuration could meet the n_z requirement at flare initiation altitude, but be unable to sustain a flare to touchdown.

The intent of the n_z requirement in ground effect is to assure the capability to control sink rate throughout the flare, and to assure that an undershot approach can be "dragged" up to the runway. Neither would be possible if the aircraft does not have an $n_z > 1$ g capability. Since this requirement might be critical in sizing the powered lift system, an experiment was conducted in the current study to determine the minimum n_z requirement as follows:

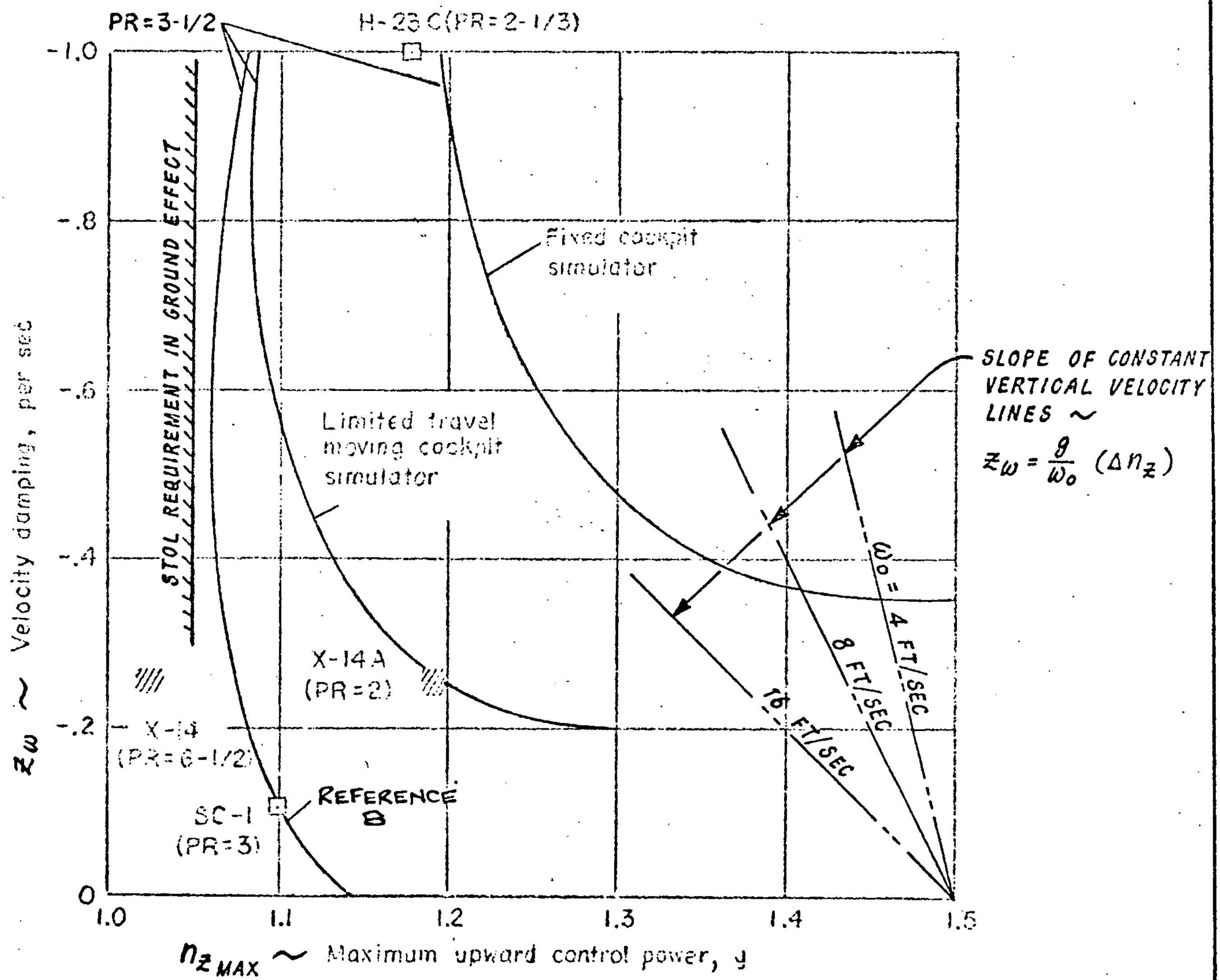
- o the lift capability (including all thrust effects) was limited to a 1.05 g capability at approach speed.
- o the simulator was set up over the threshold of the long (10,000 ft) runway with a 6 ft/sec rate of descent, which was the nominal sink rate desired for touchdown.
- o the pilot overflared from this condition and made a porpoising fly-by just above the runway, setting up and recovering from conditions typical of misjudged flares. Control Technique #2 was used. Vertical speeds of ±5 ft/sec were typical.
- o the experiment was repeated several times. The strip chart recording of lift coefficient showed that the lift limit was being encountered during conditions which the pilot felt were realistic worst cases, and for which the lift capability felt marginal.

Figure 5.4-3 shows that the results of this experiment correlate very well with the results of a prior moving base simulator test of VTOL height control requirements. As previously discussed, the required n_z capability is a function of sink rate.

NOTE :

1. STOL DATA FROM FSAA SIMULATION OF EFB. $\tau_{THRUST} = 1$ SEC
2. VTOL HOVER DATA FROM NASA TND-2451; AUG '64

Control sensitivity (g/inch)		First order time constant (τ)	
		Simulators	= 0 -> .2 sec
		X-14 & X-14A =	.28 sec
H-23C	= .37	SC-1	= .10 sec
SC-1	= .10	H-23C	= .25 sec
(X-14 & X-14A not determined)			



			REVISED	DATE		
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CHECK						FIG. 5.4-3
APPD						
APPD					THE BOEING COMPANY	PAGE 144

5.4.2.2 Incremental γ

Criteria: With the aircraft trimmed for approach, it should be possible to flare from the approach γ to level flight while maintaining the approach configuration and the approach speed. The requirement for sustaining level flight should be met in free air and at the altitude for the most adverse ground effect.

Discussion: While STOL aircraft will not normally be flared to zero sink rate, the capability to do so is desirable to avoid landing short following a misjudged approach.

This requirement applies to all viable landing configurations, including one engine-out appraches where the selected approach configuration must be chosen so that the criteria can be met. Once the airplane is committed to an all-engine approach, a subsequent engine failure requires only that the landing be completed and that the go-around requirements of Section 3.3 be met. The field length requirements of Section 3.6 will cover the possibility of short landings from such approaches.

5.4.3 Flight Path Control Sensitivity

Criteria: Column and throttle sensitivity shall not limit the maneuver capability or performance of the aircraft, and shall not result in undesirable handling characteristics. Specific requirements are:

Control Technique #1 - Pull forces and aft deflection of the column should be required to maintain increases in normal acceleration. The desired column sensitivity is

$$n_z/\delta_{col} \approx 0.1 \text{ g/inch}$$

based on the steady state n_z during an essentially constant speed wings-level pullup. The nominal column deflection required to sustain a standard 3 deg/sec level turn should not exceed half of the available column deflection, and should not require column forces in excess of 20 lb (including breakout).

Increasing nose-up pitch rate should be required to maintain increases in normal acceleration. The desired sensitivity, in terms of $\Delta\theta/\Delta\gamma$ is specified in Section 5.2.2.

Control Technique #2 - Forward throttle deflection should be required to maintain increases in n_z and γ. The desired sensitivities are:

$$n_z/\delta_{col} < .1 \text{ g/inch}$$

$$n_z/\delta_{THL} \approx 0.1 \text{ g/inch}$$

based on the first peak of the n_z response to a control input.

Pull forces and aft deflection of the column should be required to maintain increases in nose-up pitch rate. Transient pitch rates during throttle inputs with the column centered are allowable, provided the resultant attitude change is in the direction required for holding constant airspeed. Column sensitivity should meet the pitch acceleration requirements for Section 5.3.2, and the level turn requirements stated above for Control Technique #1.

Discussion

Control Technique #1 - The optimum column sensitivity and feel spring gradient depends on several factors, including the aircraft response characteristics and the static mistrims (e.g. gear and flap extension) that must be controlled.

The $n_z/\delta_{col} \approx .1$ g/inch recommendation is typical of current CTOL airplanes (for landing approach), and was used during the current STOL simulation study in combination with a 5 lb/inch feel spring gradient. Since the n_z capability during a STOL approach will normally be less than 0.5 g, the .1 g/inch sensitivity will not limit the flare capability.

The sensitivity is specified for a wings level pullup, and will normally be less than measured in a steady turn, particularly if the SAS uses a high gain pitch rate feedback loop. The column force and deflection limits for a level turn are intended to assure an allowance for control in turbulence and to minimize the need for retrimming in a turn.

Control Technique #2 - In this mode the column is used to control attitude primarily for the purpose of controlling airspeed. However, a transient load factor response may result from a column input, and should meet the sensitivity requirements stated to avoid overcontrolling and excessive coupling of speed control (column) inputs into flight path. The $n_z/\delta_{THL} = .1$ g/inch requirement was taken from the vertical velocity response requirement of Paragraph 4.6 of Reference 7.

Sensitivity requirements in this mode were not studied during the current STOL simulation. A configuration considered marginally satisfactory had sensitivity characteristics as follows:

$$n_z/\delta_{col} = .05 \text{ g/inch}$$
$$n_z/\delta_{THL} = .04 \text{ g/inch}$$

5.4.4 Flight Path Response

5.4.4.1 Load Factor and Vertical Speed Response

Criteria: Regardless of the control technique employed:

- o The load factor response time constant, τ_{n_z}, shall not exceed one second; where τ_{n_z} is defined as the time elapsed from the initiation of the flight path control input until n_z reaches 63% of the first peak magnitude
- o The vertical speed crossover time, $t_{\dot{h}}$, shall not exceed 0.8 seconds; where $t_{\dot{h}}$ is defined as the time required to achieve a positive (upward) change in vertical speed following a climb command.

Further requirements are:

Control Technique #1 - For a column step input at approximately constant airspeed, the shape of the n_z response should be approximately:

- o first order, or
- o well-damped second order, with the overshoot magnitude not to exceed 10% of the steady state magnitude.

Control Technique #2 - For a throttle step input at approximately constant airspeed, the shape of the n_z response should be smooth and with no sign reversals.

Discussion: Flying qualities criteria for airplanes normally do not specify flight path response requirements, in addition to the short period frequency and damping requirements, because the flight path of an airplane is controlled via angle-of-attack modulation. Powered-lift STOL aircraft, however, have the capability for producing large lift changes through blowing modulation without changing angle-of-attack. High authority DLC surfaces would also provide this capability. Consequently, flight path response criteria are also required.

Load factor, n_z, has been used in defining the flight path response criteria because the only way to change flight path angle is to change n_z, i.e.

$$\dot{\gamma} = n_z \times (g/V)$$

If speed is more or less constant, as in CTOL and STOL approaches, sink rate is controlled by changing γ.

In a VTOL landing, sink rate is controlled through n_z directly. Hence, n_z control for a CTOL or STOL is analagous to vertical path control for a VTOL. It appears that the τ_{n_z} requirements are not dependent on the control technique used. While similar, the τ_{n_z} requirements for CTOL, STOL and VTOL may differ depending on approach conditions and the nature of the flare task.

For STOL landings on a short runway, a quick load factor response in the flare is required for accurate control of the touchdown point. Current study results indicate the n_z response must be quicker for STOL than for CTOL.

The requirements on the shape of the n_z response are intended to assure predictability of the flight path, and a smooth ride.

Vertical speed crossover time was specified rather than n_z crossover time because:

- o γ control is the primary requirement, although n_z is used to control it
- o the control system lag time, which is a part of the crossover time, represents a different percentage of the n_z and $\dot{h}$ crossover times. Using an $\dot{h}$ crossover criteria allows a quicker n_z response to be used to compensate for part of the initial system lag time.

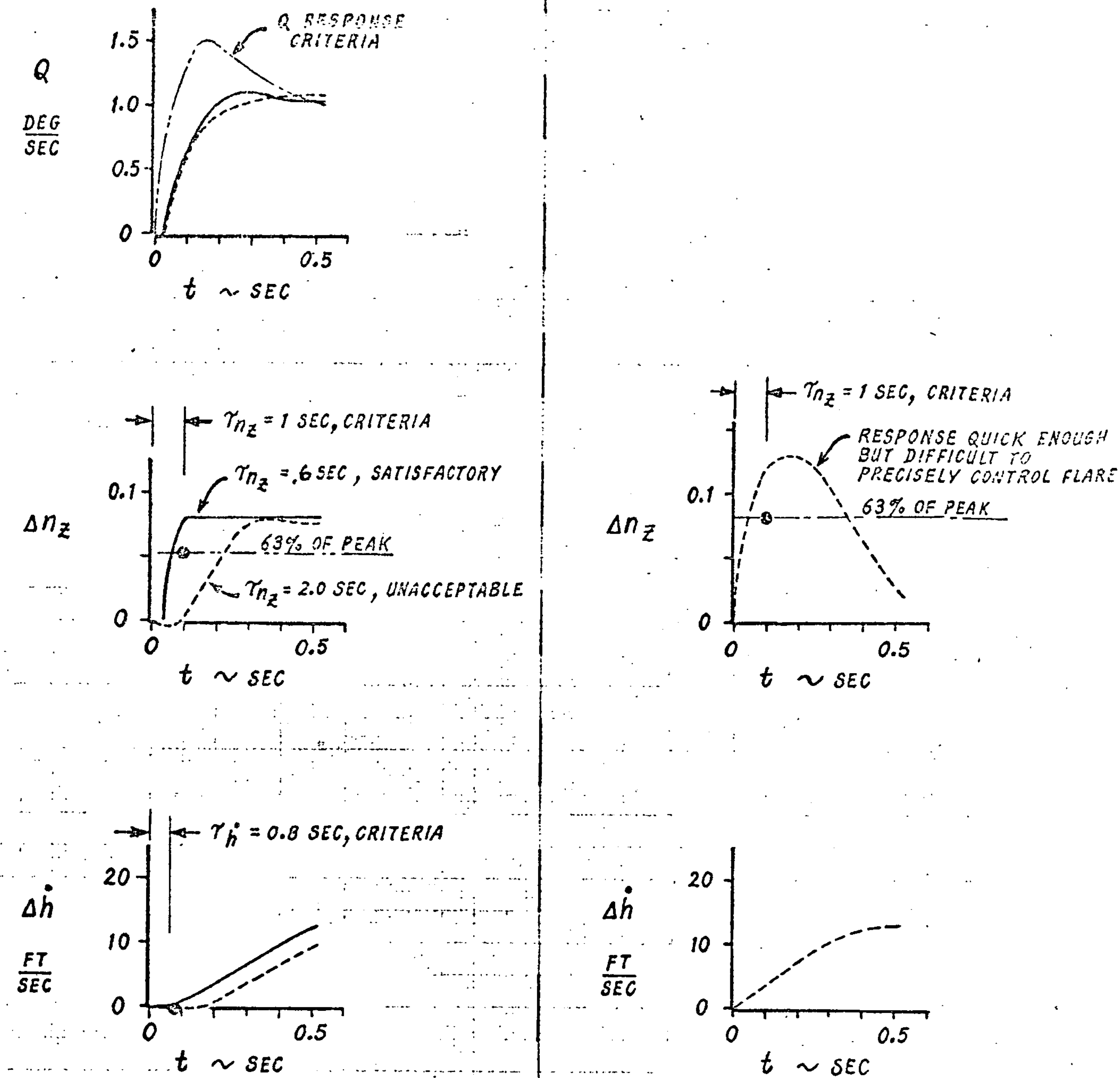
Figure 5.4-4 compares the recommended criteria against the responses of some of the configurations tested. More complete time histories are shown in Section 5.2.3.

Study results are discussed separately for the two control techniques:

Technique #1 - Load factor response requirements were studied using SAS #1. The results are shown on Figure 5.4-5. The study was begun with $\tau_{n_z} \approx 2$ secs. and $t_{\dot{h}} \approx 1.6$ secs. This is more sluggish than desired for CTOL, but was

RESPONSE TO COLUMN STEP ~ SAS #1

RESPONSE TO THROTTLE STEP ~ SAS #2



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CURRENT STUDY RESPONSES COMPARED WITH FLIGHT PATH CONTROL CRITERIA

D6-40409

FIG 5.4-4

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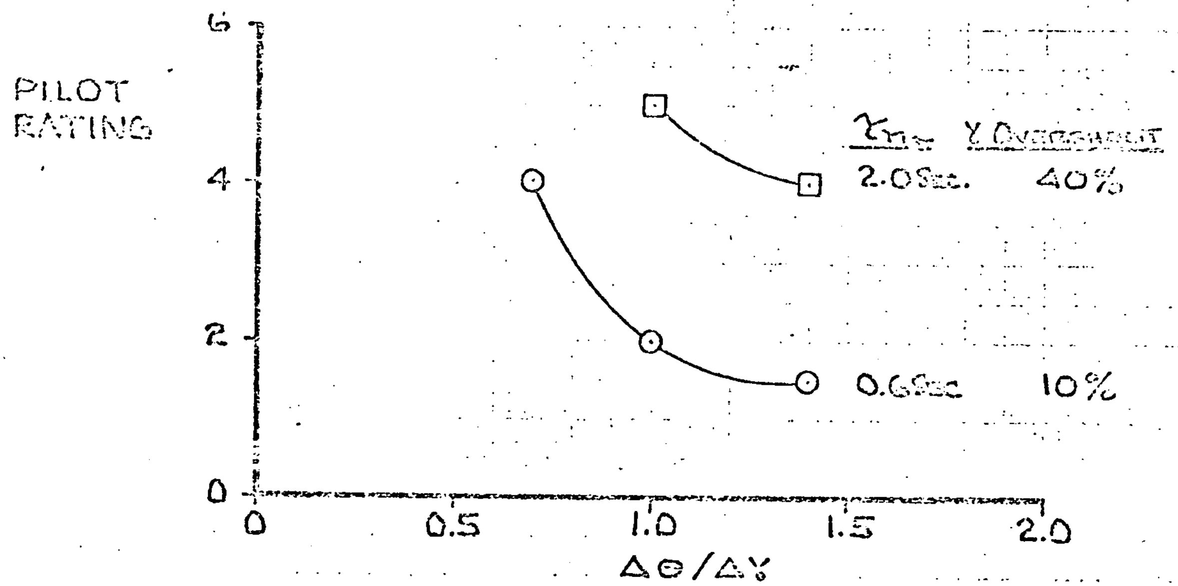

NOTE: 1. AMES FSAA STOL SIMULATION DATA
2. APPROACH & LANDING TASK ON STOL RUNWAY. V_{APP} = 80 KTS, γ = 6°
3. DATA OBTAINED WITH AUG. WING MODEL, USING SAC'L & CONTROL TECHNIQUE #1 $\delta_{COL} \rightarrow \theta \rightarrow \gamma$, THRUST $\rightarrow$ V
4. PARAMETERS VARIED BY MEANS OF FLAP & BLOWING MODULATION
5. RESPONSE TIME HISTORIES SHOWN ON FIGS 5.2-11, 5.4-4

CALC	ALLISON	2 DEC 71	REVISED	DATE	EFFECT OF τ_{n_z} & $\Delta\theta/\Delta\gamma$ FOR CONTROL TECHNIQUE #1	D6-40409
CHECK						FIG 5.4-5
APR					THE BOEING COMPANY	
APR						151

attainable without a direct forward path from the column into the flaps and engines. The characteristics were satisfactory for glide slope control and for landings in still air on the long runway. However, none of the pilots (all were extremely competent experimental test pilots) was able to land it on the STOLport from a 6 degree glide slope. On the long runway, the pilots could initiate a more gentle flare from a higher altitude. This type of flare could not be used on the STOLport due to the excessive air distance consumed by such a flare. When the flare was delayed to the altitude necessary for hitting the aim point, the response was so sluggish that:

- the flare was not completed, resulting in a hard landing; or
- in attempting to compensate for the sluggish response, the pilots used excessive column inputs, which resulted in ballooning and floating beyond the touchdown zone. This overcontrolling tendency can be attributed to the sluggish n_z response, since a large τ_{n_z} gives a large γ overshoot when the column is released.

By using a direct column input to the flaps and the thrust command, the response was quickened to give $\tau_{n_z} \approx .6$ sec., and $t_{\dot{h}} \approx .6$ sec. With this configuration, all of the pilots could hit the aim point zone on the STOLport at the desired sink rate; and all rated the configuration as a 2 on the Cooper Scale.

Unfortunately, it could not be determined from the test results whether the improvement in controllability was due primarily to the improved crossover time or the overall improvement in the response time. The recommended criteria were, therefore, arrived at by consideration of:

- Current test results, which indicate $\tau_{n_z} = t_{\dot{h}} = 0.6$ sec is a little more responsive than necessary
- Prior Boeing fixed base-simulator work regarding DLC systems which indicated $t_{\dot{h}} < 1$ sec. to be desirable for CTOL

- o The Reference 7, Paragraph 4.3 vertical flight path control requirements for aircraft response
- o The Reference 8, Paragraph 3.2.5.3 (VTOL) requirement for a 100 ft/min rate of climb within one second following a climb command

5.4.4.2 Thrust Response

Criteria: There are no specific requirements for thrust response. However, thrust lags may strongly influence the capability to meet the n_z and $\dot{t}_h$ requirements; particularly for systems designed for use with Control Technique #2.

Discussion: Reference 7 specifies a maximum time constant of 0.5 for both the thrust response and the vertical aircraft response (at constant attitude) during STOL flare and touchdown. Reference 8 specifies a maximum thrust response time constant of 0.3 seconds for thrust changes of $\Delta T/W = 0.5$ magnitude during CTOL hover, but states no such requirement for STOL. The need for emphasizing the propulsion system as an integral part of the flight control system of some STOL aircraft is recognized. However, it is felt that there is no more need to specify the thrust response characteristics for a STOL aircraft than to specify the elevator or the DLC surface actuator response characteristics. Instead, the approach taken herein is to specify the desired total aircraft response. With this approach, DLC surfaces can be used, if required, to compensate for thrust lags.

The effects of thrust response were investigated briefly in the current study for Control Technique #2. The EBF configuration was used for the study, with augmentation limited to the pitch rate and attitude loops of SAS #2; i.e., no SAS inputs to thrust.

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The VTOL hover data on Figure 5.4-6 were used in establishing the References 7 and 8 requirements for thrust response. Those data were obtained in the Ames Height Control simulator by introducing a first order lag between the height controller and the lifting system command.

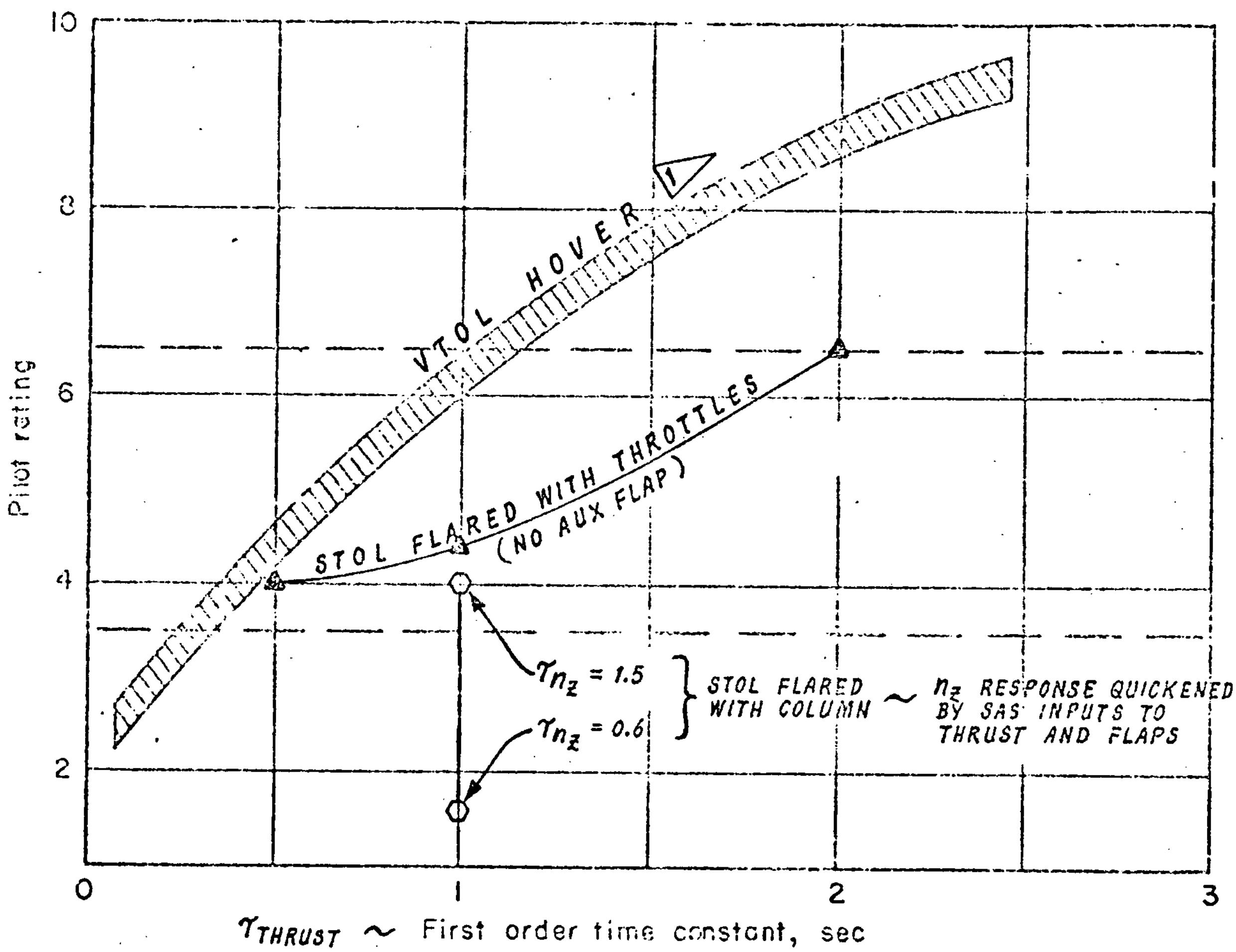
In the current study, engine dynamics were represented as a first order lag between the throttles and the thrust output. Since the throttles were used as the primary flight path control, the experiment was similar to the Reference 10 experiment; except the task was a STOL approach, flare, and landing task rather than a VTOL hover task. As seen from Figure 5.4-6, the current STOL study results correlated very well with the prior VTOL study results in that similar response times are required for satisfactory control. However, it appears that longer response times will be acceptable (though unsatisfactory) for STOL, probably because angle-of-attack changes can be used at STOL speeds to assist in initiating flight path changes.

The majority of the current study was conducted with a thrust time constant of one second. This thrust response, in conjunction with SAS #2, gave marginally satisfactory flying qualities.

NOTE :

1. VTOL HOVER DATA FROM NASA TND-2451 ; AUG. '64
- Maximum upward control power = 1.15 g
- Control sensitivity = 0.1 g/inch
- No velocity damping

2. STOL DATA FROM FSAA SIMULATION OF EBF



CALC	ALLISON	2 DEC 71	REVISED	DATE	COMPARISON OF STOL AND VTOL REQUIREMENTS FOR THRUST RESPONSE	D6-40409
CHECK						FIG 5.4-6
APPD						
APPD					THE BOEING COMPANY	PAGE 155

5.5 Speed Control

A complete coverage of longitudinal control must include a discussion of speed control as well as pitch attitude and flight path control. Speed control requirements would include criteria for control power, sensitivity and the shape of the speed response to controller inputs. The present study produced very little data to substantiate suggested criteria or to guide construction of new requirements. Further work is needed in this area to complete the definition of the complete criteria set. The following are offered as suggestions.

5.5.1 Speed Control Power

Criteria: It must be possible to hold the desired approach speed, and to change speed as required to transition to CTOL flights.

Discussion: While the requirements appear trivial, they are stated to draw attention to potential conflicts in designing the SAS. The speed-hold capability of the augmentation system should not be emphasized to the exclusion of the requirements for changing speed.

5.5.2 Speed Control Sensitivity

Criteria: The speed controller shall provide direct and rapid control of speed.

Control Technique #1 - Forward movement of the throttle should produce an increase in speed, while aft movement should produce a decrease. Throttle position should indicate the amount of speed change capability remaining. Throttle sensitivity in terms of longitudinal g's per inch is unknown.

Control Technique #2 - A nose-down pitch attitude change should produce an increase in speed, while a nose-up change should produce a decrease. The $\Delta\theta/\Delta V$ requirements of Section 5.2 should be met.

Discussion: The directions of speed control movement are conventional for both techniques.

When using Technique #1, the requirement that throttle position should indicate the speed control remaining is consistent with current unaugmented airplanes; and is considered necessary as a warning that the flight path angle is approaching values beyond which the trim speed can no longer be maintained (stall or overspeed warning).

When using Technique #2, speed control requirements should probably be similar to VTOL criteria.

5.5.3 Speed Response

Criteria: Unknown

Discussion: While the criteria for a satisfactory speed response have not been determined, the time histories in Section 5 may be useful. Speed control was considered:

- Excellent with SAS #1
- Poor with SAS #2 (sluggish, excessive attitude change required)

5.5.4 Stick Force During Speed Changes

Criteria: If speed is changed at constant flight path angle from a trimmed condition, the slope of the stick force vs speed curve (excluding breakout) should be in the range

$$0 \leqslant \left| F_S / V \right| \leqslant 1 \text{ lb/knot}$$

If the stick force required to maintain straight flight is other than zero, a push force should be required if speed is increased, and a pull force should be required if speed is decreased.

Discussion: When speed is changed, a pitching moment normally results which must be balanced by the pitch control to maintain straight flight. Assuming that the stability criteria of Section 6.0 are met, a zero gradient of stick force vs speed is preferred and can be attained by using an augmentation system with γ feedback. If the aircraft does not use γ feedback, it will probably be impossible to meet the Section 6.0 stability criteria without having a stable stick force vs speed gradient. The upper limit on the stable gradient is taken from extensive fixed-base simulator studies of the Supersonic Transport.

While the precise value of the upper limit is somewhat arbitrary, the pilots participating in the SST studies agreed it was in the vicinity of 1 lb/knot. The limit is stated to direct attention to the fact that excessively stable stick force vs speed gradients can be objectionable due to the excessive retrimming required during speed changes, and due to adverse effects on pitch dynamics.

Page

6.0 LONGITUDINAL STABILITY

6.1 Longitudinal Static Stability

Criteria: With the most critical loading, for all steady flight conditions at which the aircraft might be operated continuously, the aircraft should possess the stability characteristics outlined below.

Discussion: As stated in Reference 7, "the primary purpose of stability is to reduce divergences in airspeed, attitude, or angle-of-attack, which, if undetected by the pilot, could result in an unsafe condition in the form of either large attitudes or insufficient control for recovery."

The requirements pertain to continuous operating conditions because the pilot will normally be actively controlling the aircraft when outside these conditions; e.g., speeds below stall warning.

6.1.1 Angle-Of-Attack Stability

Criteria: If the aircraft is perturbed from a trimmed condition by a gust input with column and throttles fixed, there shall be a tendency for α to return to the trimmed value. However, this tendency should not be so pronounced as to be objectionable in turbulence. If the aircraft is perturbed from a trimmed condition by a flight path or speed control input, there should be no tendency for α to change farther from the trimmed value after the cockpit controls are released (except for momentary dynamic overshoots).

Discussion: The intent of the requirement is to assure that the aircraft will not fly into a stalled condition if trimmed by the pilot and then left unattended while his attention is diverted by some other task (e.g., navigation problem, system malfunction, etc.).

Separate specification of requirements in response to gust and control inputs is necessary to accommodate sophisticated control systems which allow the pilot to command a new flight condition and then "hold" that condition when the controls are released. As an example, consider a control system, designed for use with Control Technique #2 which allows the pilot to increase γ at constant speed by pushing on the throttle. The angle-of-attack may decrease due to the increased blowing (see Figure 5.2-2). This is certainly a safe situation, although there is no "tendency for α to return to the trimmed value" as is required for the gust input.

6.1.2 Speed Stability

<u>Criteria:</u> If the aircraft is perturbed from a trimmed condition by any input except a commanded speed change and the controls are then released (hands off), there shall be a tendency for speed to return to the trimmed value. If perturbed by a speed change command, speed should not diverge beyond the commanded value.

<u>Discussion:</u> The intent is to assure that the aircraft will not change speed significantly from the trim speed if left unattended by the pilot. Stick force vs speed gradients are not specified. While such gradients provide a convenient means for flight checking static margin, they do not necessarily assure speed stability; e.g. a strong speed vs thrust instability could offset the stable static margin and produce a speed instability. Further, stick force variation with speed is a nuisance when the pilot is intentionally changing speed and the requirement for it would preclude using a stability augmentation system having a γ-hold capability.

6.1.3 Flight Path Stability

<u>Criteria:</u> If the aircraft is perturbed from a trimmed flight condition by any input other than commanded changes in speed or flight path (γ), the aircraft with cockpit controls free should tend to return to the trimmed γ.

If the aircraft is perturbed by a commanded change in speed (V), a stable γ vs V relationship is desirable; i.e., steady state reduction in speed should be associated with steady state increases in climb angle.

However, an unstable γ vs V relationship is allowable provided the slope $\partial\gamma/\partial V$ does not exceed the limits prescribed under Section 5.2.2.

Discussion: The intent is to assure that the aircraft will not deviate (steady state) from the trimmed flight path if left unattended by the pilot. A phugoid about the trimmed γ is acceptable.

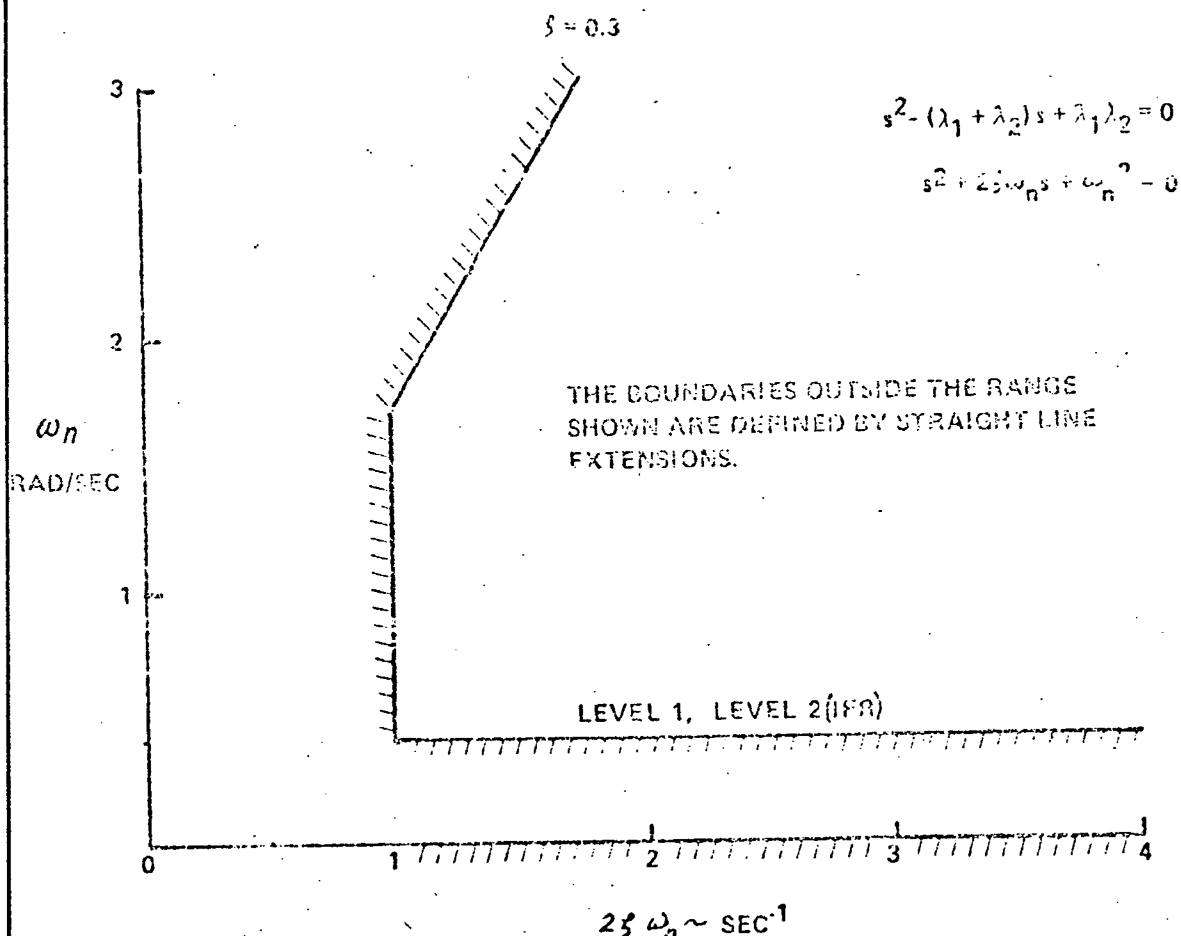
The slope of the γ vs V curve relates to flight on the back side of the drag curve as previously discussed under Section 5.2.2.

6.2 Longitudinal Dynamic Stability

Criteria: If the aircraft is disturbed by a gust input; or, during flight test, by a control input representative of a gust disturbance, there shall be no divergence (aperiodic or oscillatory) in the aircraft response. In addition, the frequency and damping of the pair of roots (real or imaginary) that dominate the short term angle-of-attack response shall meet requirements of Figure 6.2-1.

Discussion: This requirement is similar to the Reference 8 criteria, except the response characteristics have been limited to stability (i.e. response to external disturbance) rather than control inputs. Response requirements of Section 5 should be met for control inputs.

As discussed in Reference 8, the Figure 6.2-1 limits assure a positive maneuver margin.



REFERENCE : MIL-F-83300

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CALC	ALLISON	15 FEB 72			SHORT TERM LONGITUDINAL RESPONSE REQUIREMENTS	D6-40409
CHECK						FIG. 6.2-1
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7.0 LATERAL-DIRECTIONAL CONTROL

The present study was made using a linear model of the lateral-directional aerodynamic characteristics and control system. Sufficient control power was provided to give good handling qualities and allow adequate control of engine failures. A lateral-directional stability augmentation system, SAS, was provided to improve the basic vehicle characteristics.

The fully augmented lateral-directional handling qualities were rated 1.5 to 2 (Cooper scale) by the pilots. The unaugmented characteristics were rated unsatisfactory to unacceptable.

The control power was sufficient to make satisfactory approaches and landings in 30 knot crosswinds. Both the wing low, steady sideslip method, and the crabbed approach with a decrab maneuver just prior to touchdown, were evaluated.

Landings and go-arounds with an engine failure were performed. The effects of large rolling and yawing moments due to an engine failure were simulated. The lateral-directional SAS made these failures easy to control. Landings and go-arounds with a failed engine were easily performed, and were rated satisfactory by the pilots.

The selected criteria presented in this document are compatible with the optimum vehicle characteristics that were chosen for this study. Since the main purpose of the study was to investigate longitudinal handling qualities and performance requirements, no attempt was made to define minimum levels for lateral-directional control. Therefore, the following criteria are based on criteria recommended in the literature. Applicable results from the present study are used to supplement these criteria.

7.1 Directional Control

7.1.1 General

Criteria: Right rudder pedal deflection and force shall produce nose right yaw, and left pedal deflection and force shall produce nose left yaw.

Yaw control power shall be sufficient to coordinate turns, provide trim and maneuver capability during steady side slips or with the most critical engine failure(s), and provide sufficient control for decrab maneuvers.

These criteria shall be satisfied from initially trimmed conditions at the specified reference approach speeds and wind conditions.

Discussion: The design of the directional axis is governed by maneuver, trim, and upset control considerations. For crosswind operation large rudder power is required for the decrab maneuver. The decrab maneuver also makes significant demands on directional dynamic response because is is done immediately prior to touchdown.

Many STOL configurations have large yawing moments resulting from engine-out conditions. Trimming the roll axis often increases these moments. The yaw control power must be adequate to permit safe landings to be made in the wind environment that existed when the landing commitment was made.

7.1.2 Steady Sideslips

Criteria: Steady state sideslips capability shall be the lesser of 25 degrees or $\sin^{-1}$. (30/airspeed in knots) at the reference approach speed.

Discussion: These criteria are based on Reference 8. During low speed approaches in cross winds large sideslip angles may be encountered. Airlines have requested the capability to operate in 30 knot crosswinds. Present day short-haul transports do operate in crosswinds of this magnitude or larger. Therefore, the criterion stated will give the STOL aircraft the same capability as present day aircraft in the speed range above about 70 knots.

Below 70 knots the steady sideslip requirement has been relaxed. This has been done to reflect practical design problems associated with these low landing speeds.

7.1.3 Heading Response: Crosswind Landing

Criteria: With the aircraft initially in trimmed symmetrical flight, the heading response to an abrupt maximum pedal input shall be within the range, 10 deg $\leqslant \Delta\psi_{2.0} \leqslant$ 15 deg. This shall be accomplished with the wings held level at the reference approach speed.

Discussion: During an approach in a crosswind, airline pilots usually crab the aircraft to eliminate sideslip and maintain localizer track. During flare and touchdown the pilot must decrab the aircraft so that the aircraft touches down with low side velocity and with the aircraft aimed down the centerline of the runway. For good handling qualities, and for precise touchdowns, the pilot must be able to decrab quickly before the side velocity becomes excessive. He must also keep the wings reasonably level to avoid striking a nacelle. The present study showed that the aircraft could be successfully decrabbed from an initial crab angle of 10 - 20 degrees with a directional response as low as $\Delta\psi_{2\,secs} = 10°$, although $\Delta\psi_{2\,secs} = 15°$ was considered to be about optimum. Boeing STOL simulator studies have shown that the decrab maneuver should be performed in 2 to 3 seconds and that $\Delta\psi_{2.0} \approx 10°$ is a minimum level of yaw response for acceptable operation.

An aircraft with a crosswind gear would not need this decrab capability.

7.1.4 Heading Response: Engine-Out

Criteria: The aircraft shall be able to safely decrab for landing in a 30 knot crosswind with the most critical engine inoperative and the remaining engines at the higher approach power required for continued landing with one engine inoperative.

Discussion: If an engine fails during an approach, below the critical height where a go-around can be performed without touching down, the pilot will have to continue the approach and make a landing. The landing need not occur at zero crab, but must be made safely.

7.1.5 Go-Around

Criteria: It shall be possible to maintain straight flight with one critical engine inoperative and the remaining engines at maximum power, at the reference approach speed. A heavy turbulence environment shall be assumed.

Discussion: The airplane must be controllable with one engine failed. It must be possible to apply power for go-around or maneuvering without wandering beyond the confines of the landing approach corridor. Since the landing commitment may be made before the engine failure occurs, a heavy turbulence environment must be assumed.

7.1.6 Minimum Control Speed

Criteria: It shall be possible to maintain straight flight with a bank angle less than 5 degrees with a critical engine inoperative at $V \leqslant 0.9 V_{app}$. in the landing configuration. The power setting on the remaining engines shall be maximum power.

Discussion: The above requirements are intended to give a control margin with respect to the approach speed so that the pilot can perform a go-around following an engine failure. In the present study the following sequence was used to make a go-around following an engine failure. After recognizing an engine failure, the pilot arrested the rate of sink and applied power. In general, while performing the go-around maneuver following an engine failure, the pilot selected full thrust and completed the go-around without further throttle changes. As soon as the engines began to spin up he selected go-around flaps and began to climb. The pilot was able to maintain speed during the level-off maneuver and while he was transitioning to go around flaps.

During the climb out the 10% speed margin was sufficient.

7.1.7 Sensitivity and Linearity

Criteria: With the aircraft initially trimmed in straight flight at the approach speed, the yaw angular acceleration following an abrupt 1.0 inch input of the yaw control should be: 0.05 to 0.10 rad/sec^2.

For sideslip angles between ±15 degree, variation of yaw cockpit control deflection and force with β shall be essentially linear. At greater sideslip angles, an increase in deflection is required for an increase in sideslip and a gradual lessening of force is acceptable. However, pedal force shall not drop to 1/2 the maximum value.

Discussion: These criteria are taken from Reference 7. The directional control system used in the present study conformed to these criteria and was found to be satisfactory. The directional force characteristics are defined in Section 4.4. The pedal to rudder gearing was linear.

7.2 Lateral Control

7.2.1 General

Criteria: Right roll control deflection and force shall produce a right wing down change in bank angle, and vice versa.

Roll control power shall be sufficient to attain a desired bank angle quickly, to maintain wings level during a decrab maneuver, and to provide trim and maneuver capability with an engine failed. For all conditions, sufficient roll control to counter the effects of turbulence is required.

Discussion: During the landing approach the basic maneuvers which lateral control must be designed for are:

(a) Tracking the localizer beam. The pilot desires to make small lateral and directional corrections using wheel alone. The roll response to wheel must be free from time lags for small control inputs.

(b) Large turns. When capturing the localizer beam or when making a sidestep maneuver to correct a lateral offset, the pilot will make large turns. Crisp bank angle capture characteristics are required. For all turns, the roll rate and cross coupling characteristics are important.

(c) Decrab - The pilot needs sufficient lateral control power to keep the wings level during the decrab maneuver, plus some reserve to counteract gust disturbances.

(d) Engine Failure Recovery. During approach and landing, safe recovery from a critical engine failure must be possible. The pilot must be able to safely control the transient and arrive at a satisfactory trim condition. He has the option of continuing the landing or making a go-around. Since there may be no prior warning of the failure, full turbulence environment must be assumed.

7.2.2 Roll Control Power

Criteria: Roll control power for maneuvering shall be defined by the time to achieve a specified bank angle, ϕ, in a time t_{ϕ} secs., in response to a maximum cockpit roll controller input. For purposes of comparison, it shall be assumed that the lateral control surfaces have a ramp response of 0.3 sec. to this command.

At the reference approach speed and in the design turbulence environment, the lateral control power shall be adequate to provide:

(1) a minimum bank angle response of $t_{30} \leqslant 4.0$ sec. in addition to any lateral trim requirements. For example, with a critical engine failed and a crosswind level as defined in Section 7.1.2, the roll control shall be able to develop a bank angle response equivalent to $t_{30} = 4.0$ sec. for a continued landing or go-around.

(2) for maneuvers initiated from symmetrical flight conditions, $t_{30} \leqslant 2.5$ seconds is required for satisfactory operation.

In addition, at the minimum control speed as defined in Section 7.1.6, there shall be sufficient lateral control power to balance the aircraft with the most critical engine failed and the remaining engines at full power. Turbulence is not required at the minimum control speed.

Discussion: Roll control requirements can be divided into two categories.

(1) Roll control required for normal maneuver starting from symmetrical flight such as tracking the localizer, turns and sidesteps.

(2) Minimum roll control power for maneuver and stabilization required in addition to the roll control required to hold a steady sideslip and/or to trim an engine failure.

The normal maneuvering requirement ($t_{30} = 2.5$ sec.) was taken from Reference 5. Reference 11 recommends a minimum maneuvering requirement of $t_{30} = 5.5$ sec., and Reference 6 recommends a minimum roll control requirement of t_{30} = 4 sec. for a maximum surface rate roll command.

From the results of the present study it is difficult to define the minimum t_{30} requirements. For large wheel deflections the lateral control system became saturated due to the high gains needed to give proper sensitivity around neutral wheel deflection. When saturated the roll mode time constant increased from about 0.4 sec. for nominal deflections, to about 1. sec. for large deflections. Therefore, t_{30} for a given roll power, $L_{\delta} \cdot \delta_{MAX}$, varied according to the size of the wheel command and depended on the amount of trim command. A block diagram of the roll control system is shown in Figure 7.2-1. The static gain for the wheel (P = R = 0) is about 3. That is, maximum rolling moment (initial roll rate and yaw rate zero) could be commanded with 20 degree of wheel.

Figure 7.2-2 shows the lateral directional time history of an approach and landing in a 30 knot crosswind. The pilot was able to satisfactorily control the bank angle and to make a heading change of 20 degree to align the aircraft with the runway centerline.

The incremental roll control power, $L_{\delta} \cdot \delta$, 1/sec^2, available beyond trim varied from about 0.26 (β = 17°) to 0.2 (β = 22°). Roll maneuverability available beyond trim was in the region, $5.5 > t_{30} > 4.0$ secs., for an assumed roll mode time constant variation of .4 sec. to 1.0 sec.

For normal maneuvers, $\Delta L_{\delta} \cdot \delta_{MAX} = 0.5$. With a roll mode time constant of 0.4 sec., t_{30} then varied from about 1.5 to 2.25 secs. for a maximum surface rate command.

During the present study engine failures were simulated. The relation between available and required roll control power is shown in Figure 7.2-3. Heading changes were made during a go-around following an engine failure to evaluate the roll control. The roll control power was judged to be satisfactory. This margin corresponds to $\left\{ \Delta L_{\delta} \cdot \delta \right\}$ beyond trim

$= 0.2$ rad/sec^2, $\tau_R = 1.1$ sec. or $t_{30} = 4.0$ sec.

7.2.3 Sensitivity and Linearity

Criteria: The sensitivity of the lateral control system shall be:

$\frac{\phi_1}{\delta} \geqslant 0.72$ deg/inch. This requirement shall be met for the smallest controller motion that is used by the pilot in approach and landing tracking maneuvers.

In addition, the variation of aircraft roll response with controller deflection shall have no abrupt discontinuities.

Discussion: These criteria are based on References 5 and 11. Lateral control sensitivity is important because it provides information that allows the pilot to anticipate the control input magnitude required for a given task.

The lateral sensitivity is expressed in terms of a bank angle change in one second because this retains the dependence on roll time constant. The dependence of lateral sensitivity on roll mode time constant has been clearly demonstrated in Reference 12.

The lateral sensitivity requirement must be met for the smallest controller motion that will be used in approach and landing tracking maneuvers in order to limit control system lags. Boeing experience has repeatedly shown that the pilot is very aware of transport lags that inhibit the development of initial roll (or pitch) rate. A pilot can detect a transport lag of 0.1 sec. in roll rate and will downgrade the configuration if it exists.

Specific linearity requirements have not been imposed because Reference 12 has shown that pilot opinion is a function of the combined nonlinearities in the feel system and gearing. This reference shows that the pilot will tolerate a certain amount of nonlinearity, but these systems can potentially cause problems with coordination, PIO's, harmony, differential sensitivity, and other force related problems. If significant nonlinearities are to be incorporated into either the feel system or gearing, attention should be given to the pilot reaction to the specific system in performing his total task.

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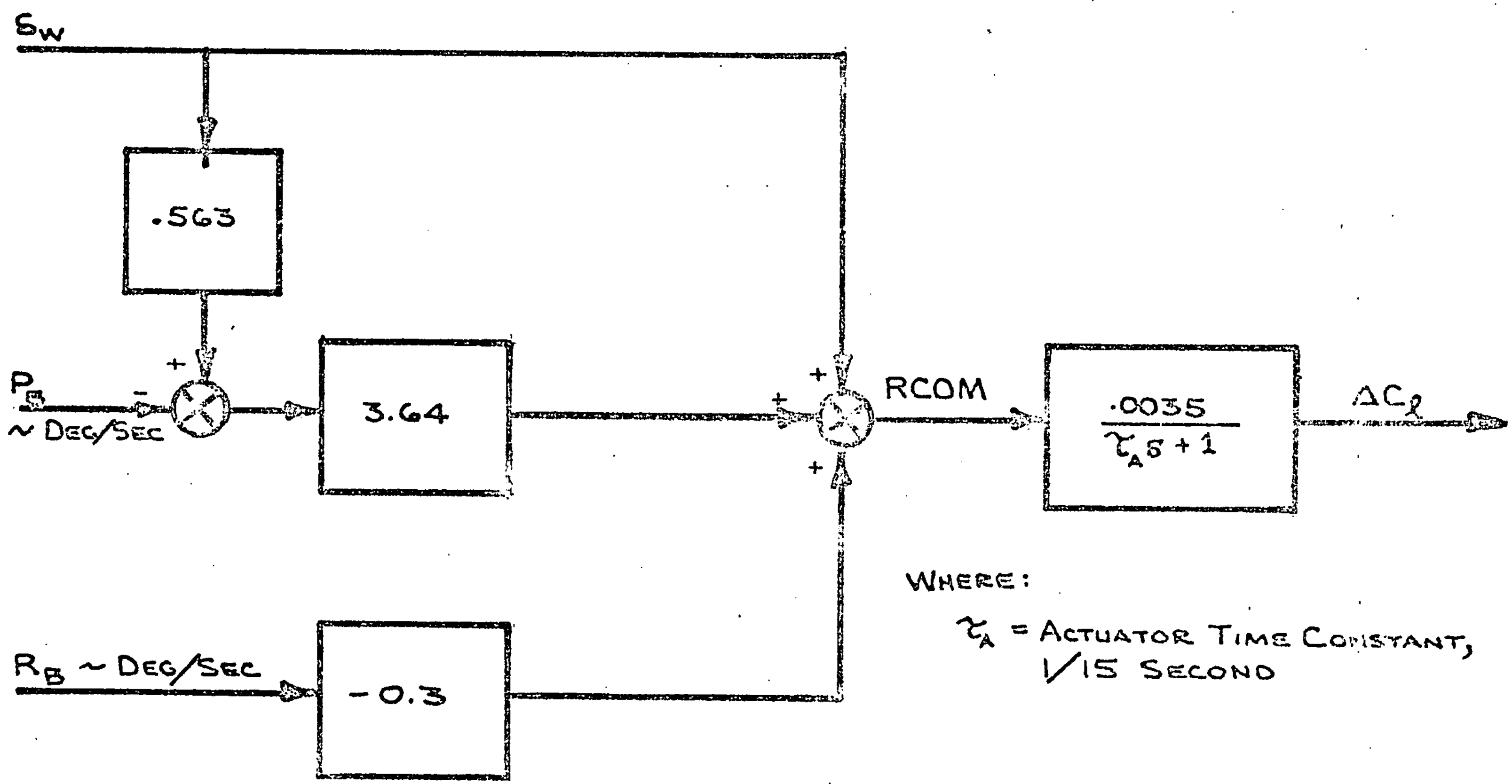


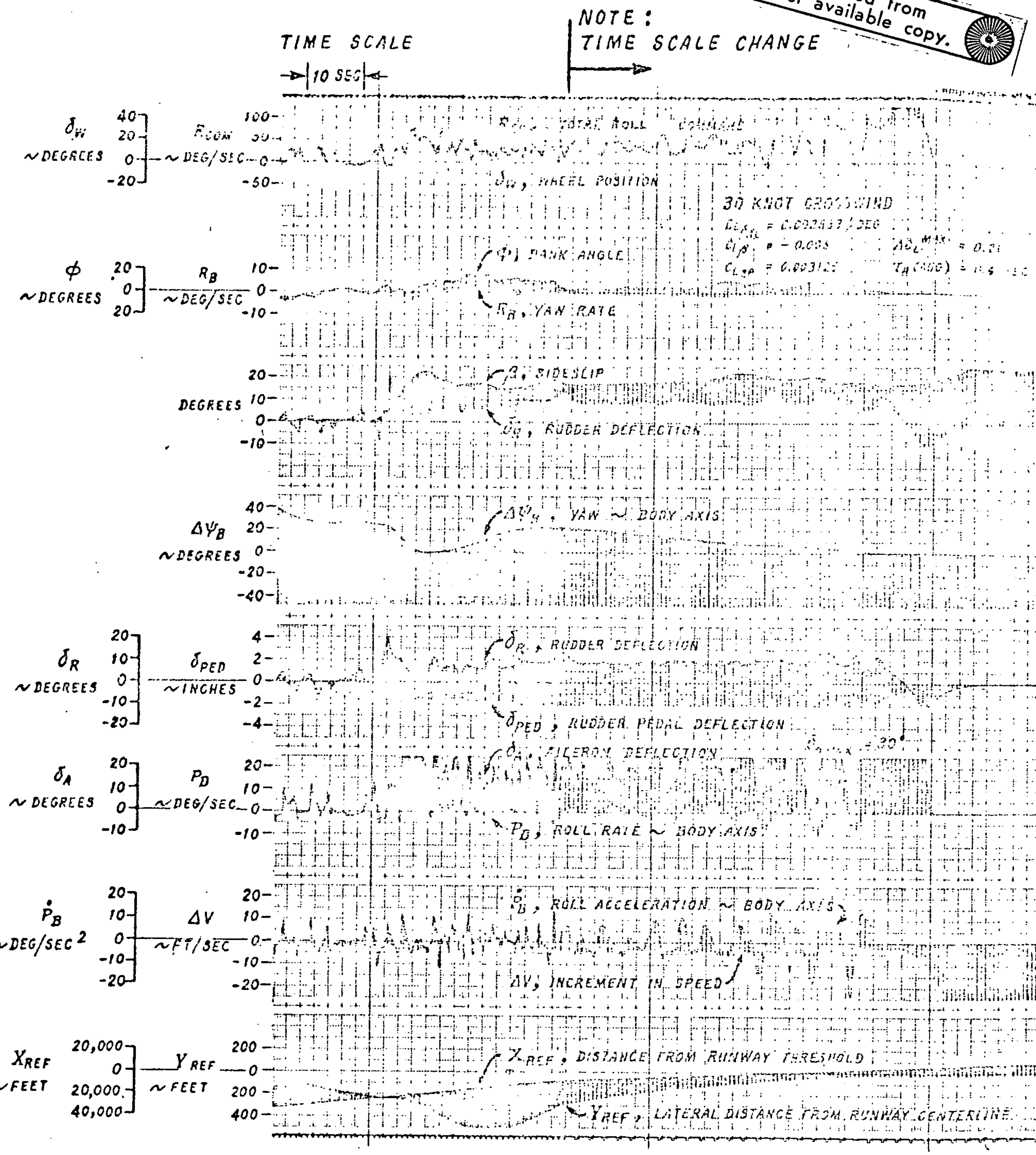
WHERE:

τ_A = ACTUATOR TIME CONSTANT, 1/15 SECOND

NOTE: δ_W IS LIMITED TO ±60 DEG., ΔC_ℓ IS LIMITED TO ±0.21 @ RCOM = ±60.

ROLL CONTROL SYSTEM

FIG. 7.2-1



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CALC					APPROACH AND LANDING IN A 30 KNOT CROSSWIND	D6-40409
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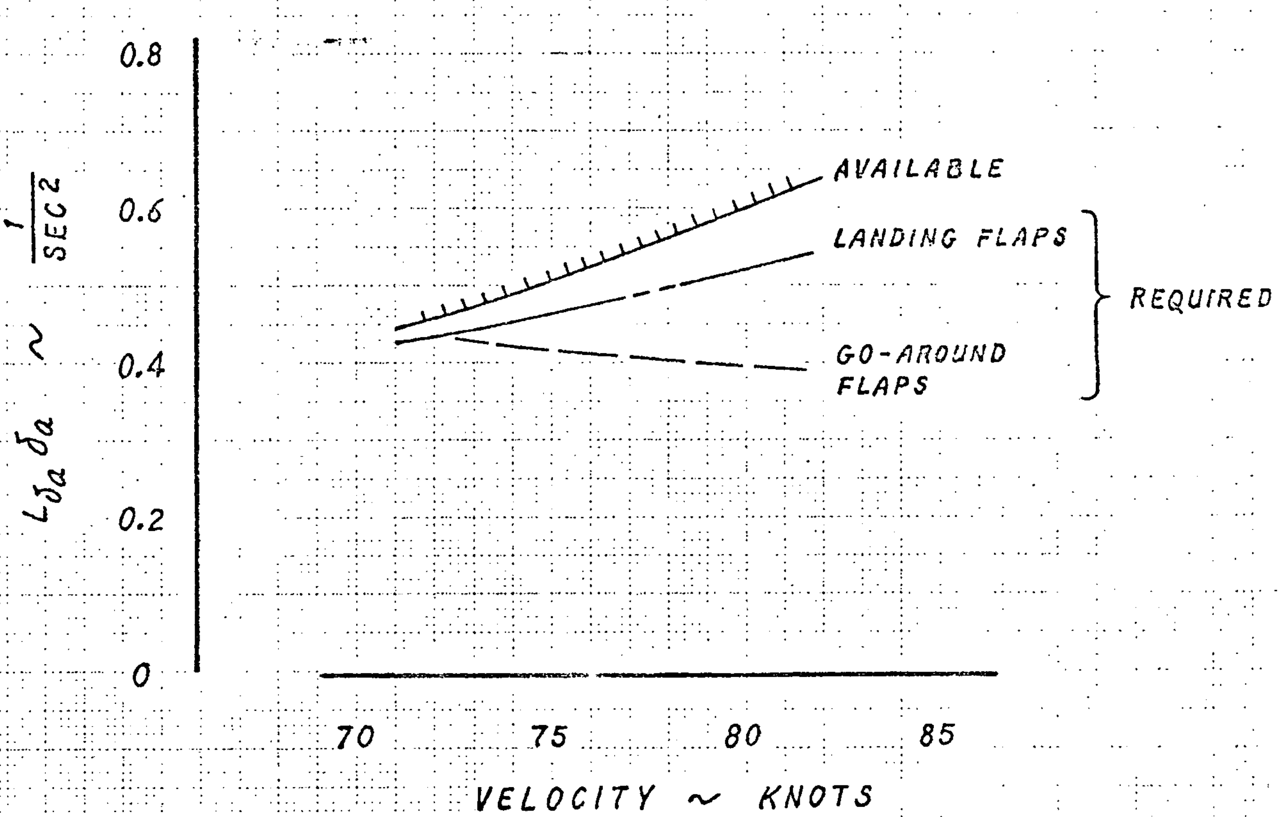
NOTE : 1. $\beta = 0$
2. τ_R = 1.1 SEC. UNAUGMENTED



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ROLL CONTROL MARGINS BEYOND BALANCE OF ENGINE FAILURE

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FIG 7.2-3

THE BOEING COMPANY

7.3 Lateral Directional Cross Coupling

Criteria: From trimmed conditions at the selected reference approach speeds in STOL operation, the magnitude of cross coupling following abrupt roll control inputs (yaw control free) up to that required to meet the roll control power criteria of Section 7.2.2 should lie between the following values of the ratio of maximum sideslip angle to bank angle:-

$$-.1 \leqslant (\Delta\beta / \Delta\phi)_{Max} \leqslant +.3$$

Also, from the same trimmed conditions, the value of the parameter $\phi osc/\phi av$ following a yaw-control-free abrupt impulse roll control command shall be within the limits specified in Figure 7.3-1.

Additional requirements on lateral accelerations due to roll and yaw control inputs may be necessary to insure adequate pilot performance and provide passenger comfort.

Discussion: The sideslip criteria of Reference 5 has been justified as a good indicator of lateral directional cross coupling in a number of simulator evaluations. As modified by the results of Reference 11 to include proverse yaw conditions, this parameter is the most tested of the many proposals in this field. The $\{\Delta\beta/\Delta\phi_1\}\{\phi/\beta\}_d$ parameter proposed by Reference 8 has theoretical and analytical justification based on the author's efforts at curve fitting existing pilot rating data. Until specific tests have been conducted to validate this criterion and its variation with the parameter ψ_B the Reference 5 criterion will be used.

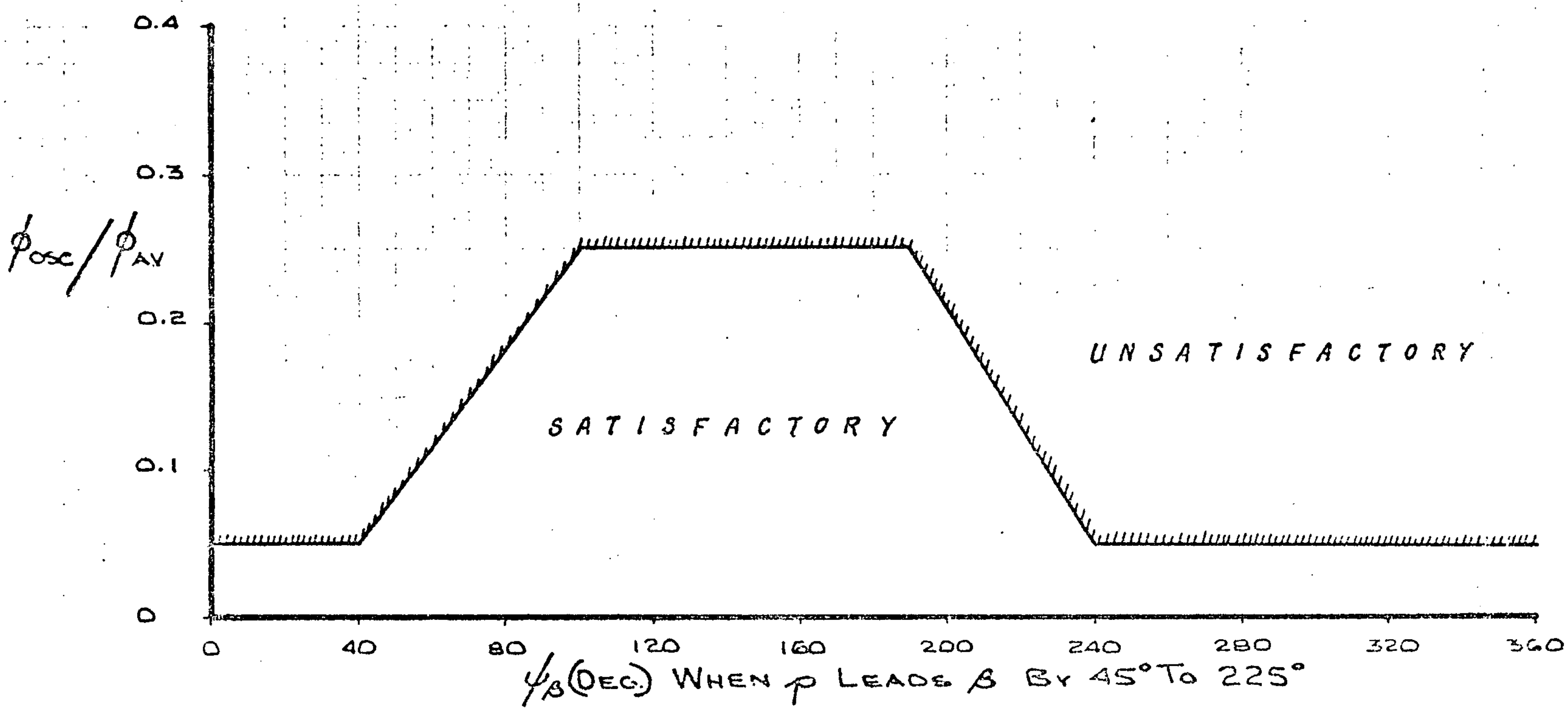
Another approach to measurement of this type of coupling is through the lag in heading response, T_ψ, as suggested in Reference 11. This suggestion has merit since the indications are that the primary piloting problem arises not from the sideslip itself but rather from its effect on precision of heading control. Again there is insufficient experience with this criterion to propose it sa a measure of commercial STOL handling qualities.

However the sideslip criterion is probably a poor indicator of piloting problems for airplane configurations having high values of $(\phi/\beta)_d$ or adverse yaw due to aileron (See Reference 8). In these cases the predominant handling qualities deficiency is in control of the resulting oscillatory bank angle response to pilot inputs. Pilot opinion data has been correlated (Reference 3) to the parameter (Posc/Pav) in response to a pulse input on the controls, and this is the criterion chosen here. The variation of (ϕ_{osc}/ϕ_{av}) allowed with the sideslip phasing parameter ψ_{β} is shown on Figure 7.3-1 and is quoted only for the case of positive dihedral effects since this is a further requirement stated elsewhere in this criteria document.

The possibility of requirements limiting lateral accelerations felt in the crew and passenger cabins due to operation of the yaw and roll controls arises from consideration of the large yawing moments from lateral controls on high lift wings, and the possible uses of high gain wheel-rudder interconnect systems for good handling qualities. Specific criteria cannot be quoted until the subjective effects of "jerk" and acceleration levels are separated and understood.

BANK ANGLE OSCILLATION LIMITATIONS

NOTE: RESPONSE TO PULSE CONTROL INPUT



LATERAL DIRECTIONAL CROSS-COUPLING CRITERIA — FOR BANK ANGLE OSCILLATIONS

D6-40409

FIG. 7.3-1

Page

8.0 LATERAL-DIRECTIONAL STABILITY

8.1 General

At typical STOL landing speeds the inter-relationships between lateral-directional control power and stability are even more important than usual. The cross-coupling between controls, and the inertial and dynamic cross-coupling, become dominant items in determining airplane response. Typically criteria are written for specific response parameters in isolation, but it is important to understand that the successful airplane meets all the criteria in combination and also does not lie close to any individual boundaries. For example, it is entirely possible to meet the roll mode time constant criteria by the use of augmentation at the expense of reasonable roll control power, but this must be avoided by designing for both requirements together.

8.2 Roll Mode Time Constant

Criteria: The effective roll-mode time constant, $\tau_R \leqslant 1.0$ seconds, as measured in response to a step input of the pilot's lateral controls. Other criteria quoted ensure an essentially single degree of freedom roll response.

Discussion: The real criteria here must include the effects of control system mechanization (actuation lags, rate saturation, etc.) as well as airplane dynamics. The total system damping is the parameter which will affect the pilot's precision of control in turning and maneuvering flight. As discussed in Reference 7, it is difficult for the pilot to distinguish between excessive control lag and low aerodynamic roll damping. Either will produce a requirement for large lead equalization from the pilot in order to prevent bank angle oscillations when attempting to stabilize at a desired bank angle.

The available experimental data relating pilot opinion variation to roll mode time constant has a fairly broad band defining the division of acceptable and satisfactory handling qualities ($\tau_R = 1.0$ to 2.0 seconds.) Very short natural time constants are generally avoided because of the possibility of large accelerations being produced in gusty conditions. The STOL configurations of the present study had natural roll mode time constants of approximately 1.2 seconds. Roll damping augmentation reduced this to about .4 seconds, this value being chosen in conjunction with the roll control sensitivity for optimum pilot rating, as shown in Reference 12. No adverse comments were generated due to accelerations in gusts although these were simulated by the large lateral motion excursions available on the six degree of freedom FSAA motion simulator. There is, however, still some doubt as to the realism of the turbulence simulation, especially in the roll axis.

The requirements for accurate control of bank angle are likely to become more restrictive for the STOL approach because of the rapid heading changes that can develop for small bank errors at these low speeds. The criterion was therefore chosen at the low end of the uncertainty band.

8.3 Spiral Stability

Criteria: With the aircraft trimmed for wings-level, zero-yaw-rate flight, the spiral characteristics should be such that with the lateral control free and following an intentional small bank angle input ($\phi = 10°$) no increase in bank angle build-up is desired; however, in no case should the bank angle double in less than 20 seconds. In addition, there should be no objectionable coupling between the conventional roll and spiral modes.

Discussion: The form of the Reference 7 criteria is used to deliberately include the effects of flight control system characteristics, lateral-directional trim changes with speed, and possible lateral trim changes due to fuel slosh, etc. Too much positive spiral stability is undesirable because of the large wheel requirements in steady turns; too little stability will require constant pilot attention to prevent the rapid buildup of large heading errors. Neutral stability is therefore desired, although a reasonable amount of instability will remain satisfactory. The present study airplanes had neutral spiral stability with augmentation, and were slightly stable ($t_{1/2} > 35$ secs.) for the basic airplane.

8.4 Lateral-Directional Static Stability

Criteria: Static directional stability is required, and will be demonstrated by the tendency to recover from a yawed condition with directional controls free. Static lateral stability is required and will be demonstrated by the tendency to reduce bank angle from a steady yawed condition when the lateral controls are freed. However, the static lateral stability may be limited by the requirements of Section 7.2.2.

Discussion: Other criteria are specified under lateral/directional dynamic stability which may be more restrictive than this simple requirement for recovery from skids. However, this is an easily demonstrated characteristic and one which is clear in it's message to the pilot. This qualitative check on static stability is therefore required in addition to the quantitative requirements for steady sideslip characteristics, Dutch roll frequency, and spiral stability specified elsewhere.

8.5 Lateral-Directional Dynamic Stability

Criteria: The lateral-directional oscillation shall have the following characteristics with controls fixed or free:

Dutch roll damping ratio, $\zeta_D \geq .08$

Dutch roll frequency, $\omega_{ND} \geq .04$ rads/sec

Discussion: The Dutch roll damping requirement is exhaustively discussed in Reference 8. What little experimental data there is on the requirements for a minimum Dutch roll frequency for STOL airplanes is open to several interpretations. However, enough unfavorable pilot opinion data is available (Reference 5) to cast doubt on the level of $W_{ND} = .25$ rads/sec set by Reference 8. For the purposes of this criteria document (i.e., good handling qualities for commercial operations) a conservative estimate has been made for this requirement of $W_{ND} \geqslant .40$ rads/sec or $T_D \leqslant 15$ secs., regardless of damping ratio.

9.0 COMPARISON OF CRITERIA FROM LITERATURE SURVEY

The following pages are the result of a literature survey of existing and proposed criteria for performance margins, and handling qualities, for commercial transport airplanes, military airplanes, and STOL research airplanes. The criteria were summarized and presented side-by-side for easy comparison. The resulting data were used to influence both the organization and content of the criteria proposed in this document.

LANDING AND GO-AROUND PERFORMANCE CRITERIA

FOLDOUT FRAME 2

ITEM \ SOURCE	FAR 25 FEDERAL AVIATION REGULATIONS VOLUME III, PART 25	FAR XX FAA TENTATIVE AIRWORTHINESS STANDARDS FOR POWERED LIFT TRANSPORT CATEGORY AIRCRAFT AUG. 1970	NASA TN D-5594 AIRWORTHINESS CONSIDERATIONS FOR STOL AIRCRAFT JAN. 1970	BREGUET 941 CHANGES TO FEDERAL AVIATION REGULATIONS, PART 25 OCT. 27, 1969	ARB PROVISIONAL REQUIREMENTS FOR POWERED LIFT AIRCRAFT	AIAA PAPER NO. 70-1240 STOL DEFINITION & FIELD LENGTH CRITERIA BY R.K. RANSONE OCT. 1970	BOEING-MASD DESIGN REQUIREMENTS FOR NORMAL OPERATIONS SEPT. 1971	BOEING - GRUMMAN ESTRA DESIGN CRITERIA AUG. 1971	BOEING - VERTOL PRELIMINARY PROPOSAL FOR STOL LANDING CRITERIA MAR. 1970	BOEING - AERITALIA PERFORMANCE MARGINS FOR POWERED LIFT STOL AIRCRAFT (TO SAME STANDARD OF SAFETY AS PRESENT CONVEN'L TYPES) AUG. 1971
MINIMUM APPROACH SPEED	$>1.30\ V_s$ Idle Power		$>1.15 V_{Min.}$ AEO & Approach Power OEI & Max. Power When no large aerodynamic changes caused by engine failure.	$>1.15 V_{Min.}$ AEO & Approach Power $>1.30 V_{Min.}$ OEI & Max. Power	$>1.3\ V_{Min.}$ AEO & Approach Power $>V_{mcl}$+10kts.AEO,+5kts.OEI $>V_{mpc}$+10kts.AEO,+5kts.OEI V at $>1.25 V_{Min.}$ Vt max.$>$V at + 15 kts. $V_{t\ min.}$ demonstrated at 30'		$>V_{Min}$ + 20 kts. $>1.25\ V_{Min}$ $>1.1\ V_{mca}$ OEI and Approach Power	$>1.2 V_{Min.}$ OEI & Approach Power $>1.1\ V_{mca}$ OEI & Level Flt. Power ($V_{mca} < 1.2\ V_{Min.}$)	$>V_{Min.}$ + 20 kts. OEI	$>1.15\ V_{Min}$ AEO & Approach Power
MANEUVER MARGIN		Criteria selected must have adequate safety margins from critical minimum values for: a) Speed deviations b) Maneuvers c) Gust protection d) Cross wind e) Wind shear f) Flight path control g) Execution of configuration changes	$>$.15g with pitch angle $>$.25g with power	$>$.25g AEO & Approach Power	$>$.35g Total AEO & Approach Power Maneuver capability down to 85% reference speed should not decrease at rate $> V^2$	$>$.30g total will permit a fully arrested sink rate at 35' after reakout at 100' (Cat.II). With 2 sec. pilot allowance.	$>$.35g total OEI and Approach Power	$>$.25g with pitch angle OEI and Approach Power	$>$.20g AEO $>$.10g OEI	LANDING $>$.45g AEO & Max. Power $>$.30g OEI & Max. Power GO-AROUND $>$.30g AEO & Max. Power $>$.25g OEI & Max. Power
VERTICAL GUST MARGIN	Margins for maneuver and gust protection in the approach are covered by speed margin.		Speed margin gives protection from a [illegible] kt. vertical gust without buffet.			Margin must give: ΔHeight $\not>$ −35' ΔLoad Factor $\not>\pm$ .3g ΔBank Angle $\not>$ 15° ΔPitch Angle $\not>$ 5° As a result of: Vertical gust = 25% Windspeed. Horizontal gust = 50% Windspeed in the worst direction.		Protection from 7 1/2 kt. vertical gust.	Protection from 7kt. vertical gust without buffet and 12 kt. without excessive buffet or loss of control.	LANDING 15° AEO, Approach Power 10° OEI, Approach Power GO-AROUND 10° AEO & Max. Power 10° OEI & Max. Power
HORIZONTAL GUST MARGIN			Speed margin gives protection from [illegible] kt. horizontal gust with altitude loss $<$ [illegible]'					V_{Min} also defined as: Speed at which the required flight path gradient can no longer be held with maximum continuous power.		Maximum acceleration along the glide slope AEO is $>$ 0.1g ($\Delta\gamma$ = 5.7°) with maximum power.
FLIGHT PATH CRITERIA		Approach to be made from altitude $>$ 500'	Control, $\Delta\gamma$ = ±[illegible] For altitudes $<$ [illegible]' ROS $<$ 1000 fpm	Control, $\Delta\gamma = \pm 2°$ for glide slopes $>$ 5°		Optimum final approach & landing gradient of −7 1/2° Steeper gradients incur impractical performance requirements for capture, follow & recapture.	For altitudes $<$ 100' ROS $<$ 800 to 1000 fpm	At flare initiation ROS $<$ 1000 fpm	Control, $\Delta\gamma$ = 1 1/2°	Control, $\Delta\gamma$ = 2° OEI & Max. power
FLARE CRITERIA		ROS at TD $<$ 3 fps					ROS at TD $<$ 5 fps .10g margin at TD OEI.	ROS at TD $<$ 6 fps		ROS at TD 4 to 6 fps AEO. ROS at TD 6 to 8 fps OEI. g margins to maximum body attitude on ground at V_{app}: .20g AEO & .10g OEI (free air).
FIELD LENGTH	Distance from 50' + 0.6 1.15 factor when wet. Landing distance measured in best atmospheric conditions.	Distance from 35' + 0.6	Single factor not suitable. Less emphasis should be placed on maximum performance and more on consistent performance within [illegible] environment.	Distance from 35' + 0.6 for $\gamma_{App} > 5°$.	Distance from 30' X $1.24 - 0.1(C_{DG}+C_{DA})$ or 1.11 } AEO $1.19 - 0.1(C_{DG}+C_{DA})$ or 1.08 } OEI Landing distance measured in worst atmospheric conditions.	1800' useable field length defined. A rational statistical probability method is used to define limits and procedures which will provide the desired safety.	Distance from threshold height.	Distance from 35' + 0.6		
GO-AROUND CLIMB GRADIENT	$>$ .032 AEO in the landing configuration with speed $<$ 1.3 Vs. $>$ .027 OEI in the approach configuration with speed $<$ 1.5 Vs.	$>$.032 or 250 fpm AEO in landing configuration $>$.027 or 225 fpm OEI in approach configuration At V_{app} and maximum power from any point during approach & landing.	Positive climb gradient desired in the landing configuration OEI. Configuration change allowed if it is without loss of lift or adverse moments to give 6°, AEO climb angle for go-around and $>$200 fpm OEI.	$>$.032 or 250 fpm AEO $>$.027 or 225 fpm OEI With maximum power and speeds $<$1.3 Vs AEO, and $<$ 1.15 Vs OEI. Configuration change allowed if it is accomplished by the same motion as required to increase thrust provided no loss of lift or appreciable change in control forces.	$>$ .032 AEO landing config. or that achieved in 5 secs. $>$ .024 OEI app. config. or that achieved in 15 secs. With speeds $>1.2\ V_{min}$ in appropriate configuration and maximum power and $>V_{mpc_1}$, $<V_{at_0}$ } AEO $>V_{mpc_1}$ + 5kt., $<V_{app_1}$ } OEI Suffices 0-AEO, 1-OEI.	A go-around will be possible without touch-down at any time before the landing decision. After the decision to land is made a touch & go will be permitted. Fast and reliable configuration changes will be allowed.	$>$ .030 AEO with maximum power	$>$ .032 AEO $>$ .027 OEI At V_{app} and go-around power with configuration change. V_{min} in go-around configuration & power must not be less than V_{min} in landing configuration and approach power. Noted that practical capability should be compatible with takeoff.	$>$.052 or 100 fpm OEI	$>$ .032 AEO $>$ .012 OEI At V_{app} and maximum power. With configuration change.

ABBREVIATIONS		
V_{Min}	Lowest speed at 1g for particular configuration, (engines and flaps) and not less than stall speed or speed at maximum allowable α or speed for uncontrollable roll, yaw, pitch or intolerable buffets	
V_{mca}	Minimum speed at which if the critical engine is made inoperative a recovery can be made to straight and level flight with $<5°$ bank angle.	
$V_{t\ max/min}$	Maximum and minimum threshold speeds.	
V_{mpc}	Minimum power controllability speed.	
V_{mcl}	Approach minimum control speed.	
V_{at}	Target threshold speed.	
AEO	All engines operating	
OEI	One engine inoperable	

FOLDOUT FRAME 1

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