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by

P. C. Rumsey, R. E. Spitzer, and
W. L. B. Glende

January 1972

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Prepared under Contract No. NAS2-6025 by
THE BOEING COMPANY
Commercial Airplane Division
Renton, Washington

for
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
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CAT. 02

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RENTON, WASHINGTON

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TITLE: RESULTS OF A PILOTED SIMULATION OF THE MODIFIED AUGMENTOR-WING BUFFALO (VOLUME I)

MODEL Modified C-8A

ISSUE NO. ______ TO: ______ (DATE)

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1.0 SUMMARY

A piloted moving-base simulation was conducted at NASA-Ames Research Center to investigate design requirements and operational characteristics of the augmentor-wing Buffalo modification. The six degree-of-freedom motion simulation and color television visual display gave excellent reproduction of in-flight pilot cues for transitions, approaches and engine failure transients. However, lack of depth perception in the visual display greatly affected the pilot's ability to judge the flare maneuver.

System design requirements were investigated for the lateral and directional flight control systems, the lateral and directional axes stability augmentation systems, the engine and Pegasus nozzle control systems, and the hydraulic systems. A number of questions pertaining to structural design criteria were investigated. As a result of this testing a great deal was learned concerning operational techniques for STOL landings, control of engine failures and pilot techniques for improving engine-out go-around performance. Flight test procedures have been suggested for maintaining a high level of safety in the event of engine failure or SAS failure.

Design changes have been identified to correct deficiencies in areas of the airplane control systems covered by the existing NASA contract. Other areas have been identified where airplane flying qualities could be improved by further study.

The overall assessment of the modified Buffalo was that airplane handling qualities and operational characteristics were adequate to perform the mission for which the modification was intended.

2.0 INTRODUCTION

A C-8A (Buffalo) airplane is being modified for STOL research with an augmentor-wing jet flap system. The modification is being done by deHavilland Aircraft of Canada Ltd. (DHC) and The Boeing Company under contract to the Department of Industry, Trade and Commerce (DITC) and the National Aeronautics and Space Agency (NASA), respectively. A 3-view general arrangement drawing of the modified C-8A is presented in Figure 2.0-1. As a part of this program it was considered desirable to conduct a simulator investigation of the expected flight characteristics of the Modified C-8A airplane.

Previous simulations have been conducted by DHC to investigate longitudinal control and flying qualities, Reference 1, and operational characteristics, Reference 2, of an augmentor-wing flight test vehicle of generally similar configuration to the present modification. These simulations proved the overall feasibility of the configuration. Unresolved questions still remained however, especially in the area of control after an engine failure, engine-out landings and go-arounds, and certain design features dependent on pilot opinion. Further investigation of this topic required an improved moving base simulation to generate more realistic motion cues. Such a facility became available when the six degree of freedom moving base Flight Simulator for Advanced Aircraft went into full time operation at ARC. It was also recognized that the Boeing design differed in sufficient detail from the earlier DHC configurations to warrant further simulation, and the present series of tests were conceived.

2.1 PURPOSE OF THE NASA/BOEING SIMULATION

As planning for the simulation proceeded side-by-side with the initial design development of the airplane, it became clear that the design requirements for many of the airplane systems could be readily defined on the simulator. A survey of the technical staff and project design groups working on the modified C-8A airplane yielded a large number of questions suitable for solution in this way. The main purpose of the piloted simulator investigation gradually became design oriented rather than an investigation of operational procedures. The engine failure problem however remained a primary subject for investigation.

2.2 BACKGROUND DATA SOURCES

This report presents only the results of the piloted simulator work. The simulator facility is not described in detail nor is the mathematical model of the airplane and its systems. These background data are referred to below and reference sources are indicated.

The aerodynamic data used in the airplane mathematical model was built up from the DHC wind tunnel testing and analysis contained in References 3 through 10. These data were suitably corrected to the modified C-8A configuration using the methods described in References 11 and 12. The final data are gathered together in Reference 13 the simulator math model specification document.

Reference 13 also contains a complete description of the simulator cab layout and the airplane control systems. These data were based on an early issue of the Configuration Control Document, Reference 14.

The conversion of all these data into a digital simulation is described in Reference 15 which includes a program listing, data tables, and the trim and dynamic check subroutines used. The NASA derived atmospheric turbulence model is explained in Reference 16.

A description of the large amplitude, six degree-of-freedom FSAA motion base is contained in Reference 17 and various unpublished NASA documents. A closed circuit color television circuit was used to display an outside world visual scene to the pilot. The capabilities of this system are fully described in Reference 18.

2.3 OVERALL SUMMARY OF TESTING

Piloted simulator work consisted of ten days of testing yielding 47½ hrs. of simulator flying time. Three pilots participated in the testing, Bob Innis from NASA, Bob Fowler from DHC and Tom Edmonds from Boeing. A day-by-day, pilot-by-pilot summary of the testing is given on Figure 2.3-1. In this figure the flying hours are split up into subjects of investigation, the engine failure investigation being included under the Lateral Control System heading. Of particular note are the extended testing periods accomplished on Nov. 3 thru 5 and the fact that each pilot at one time or another achieved a 3-4 hr. session of uninterrupted flying in the simulator without undue fatigue. The extended test periods were only possible due to the exceptional reliability of each constituent part of the total simulation complex and the efforts of NASA personnel and their subcontractors in maintaining the integration interface. The lack of pilot fatigue is a tribute to the realism and ease of use of the FSAA moving base and cab.

Each investigation was controlled by the use of an overall test plan detailing pilot briefing, tasks, parameters for each run and questions to be answered by pilot comment and engineering analysis. To ensure that simulator deficiencies and pilot learning curves did not affect the final judgements each pilot conducted an extensive orientation flying task averaging some 4 hours prior to any formal evaluations. This orientation flying included landings and approaches under various atmospheric disturbances, with and without

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the stability augmentation system operating, with various flap deflections and approach speeds, with engine failures and with all engines operating. General flying at all flap configurations was included, and each pilot conducted a number of transitions from cruise to landing. In part, this orientation period was also used to obtain checkout data of the statics and dynamics of the simulation.

Check out data and pilot comments about the simulation have been included in this report as appendices (Section 7.0). Each pilot was asked to make a summary statement about the simulator study. These reports are presented in Appendix 7.1 contained at the end of this volume.

The remainder of the appendices have been included in a second volume (D6-24806-2) due to their lengthy nature. The simulator check out documentation is presented in Appendix 7.2 (Vol II). Records of each day's flying were kept in a log giving brief details of each configuration tests,and, where appropriate, pilot comments on the conditions. These daily logs are included in Appendix 7.3 (Vol II) and, together with the rolls of analog time-histories (which are annotated), form the complete story of the accomplished testing.

Pilot comments were taped and later transcribed. Appendix 7.4 (Vol. II) contains all of the transcripts available. These constitute a somewhat incomplete record of pilot comments due to lack of tapes, one broken tape, and a partially successful attempt to reduce the recording time to those comments of particular value to the investigation. (Ocasionally the pilot would make comments with the tape recorder switched off). The overall summary statement from each pilot (Appendix 7.1) should be used to cover any gaps.

Aspect Ratio	7.2
Wing Area	865.0 ft^2
Span	78.75 ft
Flap Span -	23.0 ft

MODIFIED C-8A

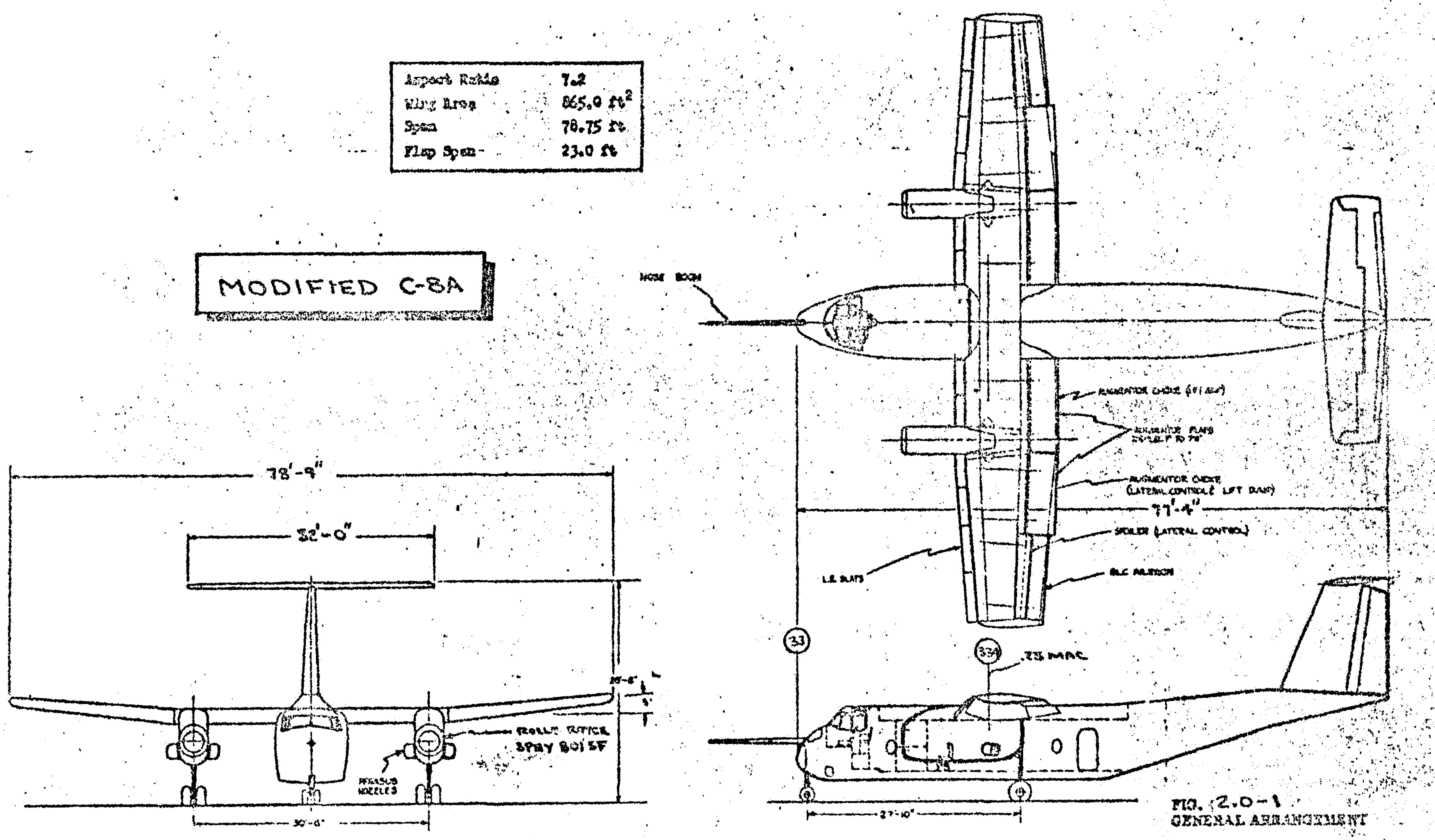

FIG. 2.0-1
GENERAL ARRANGEMENT

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SIMULATOR INVESTIGATION - SUMMARY OF PILOTED TIME

DATE	SUBJECT OF INVESTIGATION	Piloted Time INNIS	EDMONDS	FOWLER	DAILY TOTAL HR:MIN
Oct. 27	Simulation Checkout and Pilot Familiarization	1:00	1:30		2:30
Oct. 28		1:15	1:00		2:15
Oct. 29		1:45	2:00		3:45
Oct. 30	Lateral Control Power and Sensitivity	3:00	2:45		5:45
Oct. 31	Engine-Out Control	2:45	2:45		5:30
Nov. 2	Manual Reversion Hydraulics Failures Trim Rates	2:15	1:45		
	Pilot Familiarization			1:00	5:00
Nov. 3	SAS Evaluation	2:15	1:30		
	Piloted Familiarization			2:30	6:15
Nov. 4	Lateral Control			1:45	
	SAS Failures	2:30			
	Engine-Out Control			1:30	5:45
Nov. 5	SAS Evaluation	:30	2:00	1:30	
	Structural Design Criteria		1:00	2:00	
	Other		:45		7:45
Nov. 6	Other	1:00	1:00	1:00	3:00
	TOTALS	18:15	18:00	11:15	47:30

FIG 2.3-1

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3.0 DESIGN DATA FROM THE SIMULATOR

This section of the report includes analysis of the simulator investigations into areas affecting systems design and design requirements. The results are separated into sub-sections dealing with particular airplane systems.

A subsequent section of the report deals in greater detail with pilot techniques and operational aspects investigated in the simulator. All design changes arising from the simulator results are summarized in Section 5.1.

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3.1 LATERAL CONTROL SYSTEM DESIGN

3.1.1 Simulation and Testing

The lateral control system programmed into the NASA-Ames simulation was based on an early design cycle; however, that system turned out very similar to the final design concept for the flight test vehicle. Rolling moment was generated by three types of control surfaces:

- Outboard blown ailerons
- Spoilers located ahead of the outboard ailerons
- Augmentor choke surfaces located on the outer flap panels.

Ailerons were drooped as a function of flap deflection and deflected differentially through the pilot's wheel. Rolling moment and aerodynamic interactions were simulated as complex functions of individual surface deflection, flap angle, angle of attack, engine power setting (blowing coefficient) and airspeed. The distribution of engine fan air for blowing the flaps, ailerons, and fuselage corresponded to the design level of blowing coefficient on each component. Each aileron was blown by air supplied only from the engine on the opposite wing. This feature plus asymmetric flap blowing provided rolling moment compensation in opposition to the hot thrust rolling moment in the event of an engine failure.

The dual hydraulic power arrangement to the lateral control was simulated by programming features which deactivated various surfaces. Manual reversion (two hydraulic failures) also required changing the wheel force gradients in the FSAA cab. The feel forces simulated for normal operation matched the output of a mechanical feel-and-centering spring. Lateral trim was provided by offsetting the feel-and-centering spring which moved the pilot's wheel and consequently the control surfaces.

Pilots who evaluated the lateral control system were briefed on the simulation and airplane systems. The pilots were made aware of the design support orientation of the simulator study. Simulator limitations and lateral control design concepts were explained. Particular attention was given to the engine-out rolling moment phenomenon. Existing wind tunnel data show a lack of dihedral effect on the airplane. The pilots were apprised of this fact and that C_{ℓ_β} would be set at various values in the study.

Piloting tasks for lateral control evaluation centered on landing approach in the STOL configuration. Pilots were asked to make rapid turns with heading changes up to 20° as well as tracking the localizer on the glide slope. The localizer to the STOL runway could be offset by 200 ft. In this case the pilot was required to perform a sidestep maneuver at 300 ft. altitude and land on the STOL runway. In addition, certain evaluations were conducted from a go-around and climbout at takeoff flap settings.

The lateral control system characteristics were evaluated for normal airplane operation with variations in dihedral effect, rolling moment of inertia, lateral control sensitivity, aileron droop angle, lateral trim and wheel forces among others. Evaluations were conducted both SAS on and SAS off. Various levels of turbulence were simulated along with large discrete lateral gusts. Lateral control was evaluated for certain critical airplane failure conditions:

- Engine failure
- One and two hydraulic systems shut down
- Burst air ducting line

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3.1.2 Control Characteristics

Lateral control characteristics were evaluated for the most part at landing approach along a 7.5 deg. descent path at V_{App}=60 kts & δ_F=75°(δa_{droop}=30°). Most approach conditions were conducted at 40,000 lbs. gross weight. At approach power setting, maximum control power was $C_{\ell_{max}}$ = .16 which generated $\ddot{\phi}$ = .4 RAD/SEC^2 at the nominal roll inertia of I_{XX} = 335,000 SLUG-FT.^{2}. The three control surfaces were programmed with the pilot's wheel to produce a smooth rolling moment function with a "convex" shape (see Section 7.2 or Figure 3.1-1). By virtue of mixing the ailerons and spoilers, at the trim angle-of-attack the yawing moment produced by the lateral control system was nearly zero. Moderate lift losses occurred with control wheel input due mainly to the augmentor choke.

Lateral control power for tracking tasks was found adequate in moderate turbulence with the SAS turned on. Considerably less than full control input was required. Turn entries and sidestep maneuvers are strongly affected by the dynamic characteristics of the airplane. In the SAS-off mode aerodynamic cross-coupling through the rate derivatives and poor spiral characteristics with C_{ℓ_β} = 0 made turn coordination very difficult. Large adverse sideslip angles were induced ($\Delta\beta/\Delta\phi$ = .65) even though the lateral control input by itself produced very little yawing moment.

With the lateral/directional SAS turned on the poor dynamic characteristics of the airplane were greatly improved. The sideslip induced in a turn entry decreased ($\Delta\beta/\Delta\phi$ = .2) and the time lag in heading change was less than 2 seconds. When asked, the pilots stated that lateral control cross-coupling was not a problem on the airplane.

3.1.3 Lateral Control Sensitivity

Lateral control sensitivity was evaluated by all three pilots. Control sensitivity was increased in two ways. The first way was to decrease the airplane's rolling moment of inertia. Figure 3.1-1 illustrates the airplane's roll acceleration characteristics at landing approach. The original Buffalo characteristics are shown for comparison. Reducing I_{XX} not only improves sensitivity ($\ddot{\phi}/\delta_W$) but also increases the magnitude of $\ddot{\phi}$ at full wheel and reduces the roll mode time constant, all items which should tend to improve pilot opinion. Roll sensitivity was also increased by factoring the control wheel signal, effectively reducing the amount of wheel deflection to achieve a given rolling moment. Maximum rolling acceleration was not changed by this technique.

The results of the lateral control sensitivity study are summarized in Figure 3.1-2. The lowest level of $\ddot{\phi}/\delta_w$ in the data corresponds to the basic airplane ($\delta_{w_{max}} = 75^\circ$ and $I_{XX} = 335{,}000$ SLUG-FT^2). The tested condition of reduced roll inertia ($I_{XX} = 240{,}000$ SLUG-FT^2) is also noted on the Figure. Other sensitivities were generated by the factoring technique. Data from other airplanes are shown on the Figure for reference.

The data show improvement in pilot rating with increased sensitivity; however, even with very high sensitivity it was not possible to improve the airplane beyond a pilot rating of C.R. = 3. While sensitivity could be raised by reducing wheel travel, mechanical advantage would be lost for manual reversion. Reduction in roll inertia raised the pilot rating to an acceptable level. A reduction in roll inertia to $I_{XX} = 260{,}000$ SLUG-FT^2 is readily achievable by modifying the fuel use sequence to empty the outboard tanks early in the

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flight. This modification will be included on the augmentor wing flight test vehicle to enhance lateral control sensitivity.

3.1.4 Control in Heavy Turbulence

The turbulence subroutine was developed by NASA and generated random turbulence of a selected RMS magnitude. For the simulator study the magnitude was arbitrarily varied until the pilot judged the turbulence level as "light", "moderate", or "heavy". Large discrete lateral gusts were added to the turbulence at random time intervals to increase pilot workload. Generally speaking, the pilots rated the turbulence as "realistic".

Pilots were asked to fly approaches at 50 and 60 knots in heavy turbulence (a turbulence level at landing beyond normal operational experience of the pilots). The airplane configuration included the higher set of inertias (I_{XX} = 335,000 SLUG-FT^2) and rate limiting on the lateral control system to increase workload. The lateral/directional SAS was turned on for all conditions. Attempted landings with a rate limit at $\dot{\delta}_w$ = 100 DEG/SEC were rated "unacceptable". The rate limiting was raised to $\dot{\delta}_w$ = 150 DEG/SEC and the pilots were able to land the airplane with "marginally acceptable" lateral control (Cooper Rating of 5 to 6). Pilot rating went up to C.R. = 4 1/2 to 5 for $\dot{\delta}_w$ = 200 DEG/SEC. Unlimited surface rate capability was rated "acceptable" in very heavy turbulence.

Rapid surface rate capability on the order of $\dot{\delta}_w$ = 200 DEG/SEC has been incorporated into the design of the augmentor wing flight test vehicle lateral control system. Full deflection of the lateral control surfaces is possible within 1/2 second.

3.1.5 Feel and Trim Design

Before the simulator study, it was realized that pilot control forces must be low enough for one-hand operation. Positive system centering was also deemed quite important. The nominal lateral feel-and-centering program used in the study is presented in Figure 3.1-3. Maximum wheel force was $F_{w_{max}} = 14$ lbs; static friction was ±.2 lbs. similar to the airplane. With this force level one-hand precision control was difficult, and the pilots preferred a lower force gradient of $F_{w_{max}} = 9$ lbs (Figure 3.1-3). Positive centering was also emphasized by the pilots.

Reduced control forces were necessary to permit tighter control of the airplane in turbulent or engine-out conditions when the pilot used his other hand to modulate throttles, thrust vector controls and flap selector. Pilots also found that with high force levels it was difficult to release the thumb to operate the trim switch and still maintain precise control. Reduced lateral wheel forces relieved this problem.

Lateral trim was provided by offsetting the feel-and-centering spring through a simulated trim actuator of specified rate capability. Lateral trim rate was evaluated at $\dot{\delta}_w = 3.25$ DEG/SEC, 5 DEG/SEC and 6.5 DEG/SEC. The lower value was deemed too slow by the pilots. The fastest rate was preferred at landing speeds but was considered too fast for cruise flight conditions. The simulator study led to the $\dot{\delta}_w = 5$ DEG/SEC lateral trim rate specification for the airplane.

Directional trim was also evaluated on the simulator. The basic Buffalo trim rate of $\dot{\delta}_R = .8$ DEG/SEC was judged too slow for the STOL flight condition. A trim rate of $\dot{\delta}_R = 1.6$ DEG/SEC was tried and found more to the pilot's liking. It should be noted that changes to the rudder feel and trim unit are not planned as part of the C-8A Modification Program. Changes to the rudder trim rate in light of the limited rudder trim authority ($\delta_R = \pm 6^\circ$) are not warranted.

3.1.6 Effect of Aileron Droop Angle

One pilot flew a configuration with an aileron droop angle of 45°. The airplane weight was 40,000 lbs with I_{XX} = 335,000 SLUG-FT^2 and 20% increased roll sensitivity (factored wheel signal). Lateral control characteristics on landing approach were evaluated.

The aerodynamic data indicate that a droop angle increase from 30° to 45° is accompanied by a 20% reduction in $C_{\ell_{\delta a}}$ and a 100% increase in $C_{n_{\delta a}}$ (at α_w=0), but spoiler effectiveness is increased due to the increased aileron deflection. The overall effect on the airplane is a small increase in wheel sensitivity ($C_{\ell_{\delta w}}$) coupled with a slight increased adverse yaw from the lateral controls (C_n/C_ℓ from -.01 to -.10). Along with these changes, the lift and pitching moments due to the lateral control deflection are decreased.

The time histories of the evaluation runs for 30° and 45° droop angles (Figs. 3.1-4 and 3.1-5) show little difference in wheel activity for rapid heading changes. Computing the lateral cross-coupling factor $\Delta\beta/\Delta\phi$ for the two cases, shows an increase from about .17 (SAS-on) to about .22 (SAS-on) for the 45° droop angle (which agrees with expected results from the aerodynamic data analysis above). However, attempting to measure the effective lag in heading response, T_ψ, from these time histories showed a lot of scatter and, on the average, a decrease from 3.2 seconds to 2.5 seconds for the 45° droop angle. This result is the reverse of what was expected and is presumably due to the inaccuracy of measuring T_ψ from piloted data.

The digital print-out of pilot workload taken during the two runs shows an increase of 2 lbs. - 3 lbs. in the maximum wheel forces used for the 30° droop case, which agrees with the observed fact that slightly higher roll rates were used during that evaluation and the expected increase in $C_{\ell_{\delta w}}$ for 45° droop.

The only real observable difference between the two sets of analog traces is that for the 30° droop angle case the sideslip maneuver was accompanied by a considerable adverse sideslip (±6° β for ±19° ϕ) whereas the 45° droop case showed very little (±1° β for ±16° ϕ). A closer examination of the two cases reveals that the sidestep maneuvers were conducted at considerably different angles of attack (α = -3° for 45° droop, and α = +4° for 30° droop). Further examination of previous cases for 30° droop showed that there is a strong effect of α on the sideslip generated in the sidestep maneuver. One case was found where the sidestep maneuver was conducted at α = +1° and the sideslip generated was ±4°. The main mechanism for these changes is the cross-coupling term $p \cdot \alpha$ in the sideforce equation, large roll rates at high angles of attack generating large $\dot{\beta}$ values. A secondary effect is the increased adverse yaw generated by the roll control at higher α TRIM. The trim condition for 45° droop was arbitrarily chosen in the simulator with the same power set as for the 30° droop configuration. This gave a trim angle of attack of 2° more negative for a 45° droop angle which helped to contribute to the observed differences in the evaluation maneuvers. In practice it is more likely that in a 45° droop configuration advantage would be taken of the improved L/D and the airplane would be trimmed at the same angle of attack as the 30° droop configuration but with less power.

In summary, the observable differences in handling qualities between a 30° droop and 45° droop at 75° flaps are extremely small. If a requirement for data should dictate the necessity of a flight test at an increased droop angle of 45°, there should be only a small degradation in flying qualities.

3.1.7 Engine-Out Lateral-Directional Response

An engine failure on the modified Buffalo results in a combination of rolling and yawing moments on the airplane due to the vectored hot thrust from the remaining engine. Early simulator work conducted by NASA and deHavilland showed that engine-out control was a very serious problem on the airplane. With the hot thrust vectored down, the engine-out rolling moment was on the same order as the lateral control capability. Some form of built-in rolling moment compensation was deemed necessary to alleviate the engine-out control problem.

Blowing air from each engine was distributed to the flap panels and fuselage BLC and to the aileron on the wing opposite the engine. Asymmetric blowing was produced by this system. The resulting airplane duct configuration produced a nominal blowing distribution sketched in Figure 3.1-6. The flaps were blown by both engines; however, each engine delivered 44% of its cold thrust (C_j) to the flaps on the opposite wing compared with 40% to the flap on its own side. Each aileron was only blown with 10% of the cold thrust from the engine on the opposite side of the airplane. The remainder of the blowing air was distributed to the fuselage BLC.

Asymmetric lift is inherently produced upon an engine failure (see Figure 3.1-6). This lift distribution produces an aerodynamic rolling moment which opposes the unbalanced hot thrust from the operative engine, thereby reducing the lateral control input required from the pilot. Figure 3.1-6 illustrates how rolling moment is generated by one blown and one unblown aileron. This compensating moment is generated with drooped ailerons and requires no pilot input.

Figure 3.1-7 shows the net engine-out rolling moment due to an engine failure at landing flaps with the thrust vector pointed downward ($\nu = 90^\circ$). At approach power the engine-out transient is significantly reduced by asymmetric blowing. As power is increased to the emergency setting on the remaining engine to recover a portion of the powered lift lost after an engine failure, the engine-out rolling moment rises to an appreciable percentage of the lateral control capability. Figure 3.1-8 illustrates the lateral control situation presented to the pilot in the simulator under this condition. Lateral control capability is degraded under engine-out conditions. Without inherent roll compensation, the airplane could not be controlled at STOL speeds as shown by the "raw hot thrust" curve in Figure 3.1-8.

If the pilot rotated the Pegasus nozzle aft, the hot thrust rolling moment was reduced and turned into yawing moment. Figure 3.1-9 illustrates the phenomena at 60 knots. For a go-around ($\nu = 18^\circ$) almost full rudder is required for control. With the hot thrust rolling moment reduced, the built-in aerodynamic asymmetric blowing results in reversed wheel deflection to balance the airplane.

Asymmetric blowing also is effective at takeoff and climbout conditions. Even at thrust vector of $\nu = 18^\circ$ an appreciable rolling moment is generated ($\sin 18^\circ = .31$). The net, compensated rolling moment at takeoff flaps ($\delta_F = 30^\circ$) and climbout speeds ($V_e \doteq 75$ knots) is very nearly zero. Figure

3.1-10 illustrates this characteristic. Note that with roll compensation the airplane has sufficient lateral control even at $\nu = 90^{\circ}$.

3.1.8 Pilot Techniques for Engine-Out Control

The pilots were subjected to engine failures via a remote switch on the test engineer's console. The three pilots who flew the simulator conducted between them nearly one hundred engine failure conditions. Upon gaining control of the situation the pilots either continued the approach or made a go-around. Adequate control power existed in the simulation to counter the engine failure. Engine failures on approach resulted in some increase in speed which tended to further reduce the amount of lateral control input. The pilots were able to gain control of the airplane using approximately $\delta_W = 35^{\circ} - 45^{\circ}$ ($\delta_W = 75^{\circ}$ max) and less than $\delta_R = 5^{\circ}$ ($\delta_R = 25^{\circ}$ max). The initial bank angle upset was on the order of $\phi = 7^{\circ}$. In the recovery (added power, changed thrust vector) bank angles of $\phi = 10^{\circ} - 12^{\circ}$ were used to return to the runway centerline. Use of rudder with the Pegasus nozzle aft ($\nu = 18^{\circ}$) increased the required levels to $\delta_R = 10^{\circ} - 15^{\circ}$.

Engine-out during takeoff climbout conditions produced a more or less conventional airplane response. Very little wheel input was required due to the inherent compensation. Lateral control for engine failure on takeoff was definitely not a problem.

Even though adequate lateral control was available to trim the hot thrust moment, the engine-out condition was by no means insignificant. The change in sign in rolling moment with thrust vector coupled with the lack of dihedral effect produced a confusing situation to the pilots. The movement of the thrust vector introduced yawing and pitching moment changes, further complicating the problem. These confusing characteristics, coupled with marginal single-engine performance, make the engine failure condition on approach quite taxing even with adequate control power.

Due to the unconventional engine-out characteristic, the pilots spent considerable time working out techniques for dealing with an engine failure. Two distinct techniques finally emerged from this process:

Technique A

The technique favored by two of the pilots was to immediately react to the engine failure with an increase in the thrust of the remaining engine. If the landing was to be continued the nozzles and flaps were left in the landing configuration and rate of descent was varied using power and elevator control. If a go-around was to be made, the nozzles were raised and the flaps selected to 30° as the speed built-up, as shown in Figure 3.1-11.

This technique has the advantage of giving a quick recovery of most of the lift lost when the engine failed. However, the rolling moment builds up sharply from the live engine and requires a greater amount of lateral control. Adjustment of rate of descent with power with the nozzles at 90° gives only very small pitching and yawing moment changes, but large changes in rolling moment.

The initiation of the go-around also occurs at high power setting. Rotating the nozzles in this condition gives a very large nose-up pitching moment and a transfer of large out-of-balance moments from roll to yaw. The last effect tends to be very confusing to the pilot requiring a great deal of control coordination to keep the wings level and the airplane pointing towards the runway.

Technique B

The remaining pilot preferred to raise the nozzles immediately after engine failure, and then increase the thrust. This technique minimizes the control required to balance the failed engine, and was adopted by this pilot because of his intense dislike of the confusion resulting from the particular set of cues generated by the engine failure. Go-around was then initiated by flap retraction at the correct speed. Figure 3.1-12 presents a "Technique B" go-around.

For a continued landing, rate of descent was controlled by elevator and power changes with the nozzles fully up - a condition more in line with current airplanes. Power application causes almost pure yawing moment and a fairly strong nose-up pitch. However sufficient rate of sink may not be available at reasonable power settings to maintain a desired glide slope with the nozzles fully up. In this case the nozzles must also be used to maintain the glide path. Moving the nozzles caused sudden large rolling moments on the airplane generally close to the ground making the landing difficult.

3.1.9 Engine-Out Control with Reduced Roll Compensation

The larger portion of all landings and go-arounds on one engine were made with the nominal asymmetric blowing levels. Data available at the time of the simulator investigation gave the following asymmetric blowing conditions for the approach power setting -

- One engine T_{HOT} = 3760 lbs.
  T_{COLD} = 2600 lbs.
- Flap cross-over duct thrust - at flap nozzle = 1142 lb/engine (44%)
- Flap straight-back duct thrust - at flap nozzle = 1040 lb/engine (40%)
- At each aileron nozzle, thrust = 260 lbs. (10%)
- Upper fuselage blowing, thrust = 158 lbs. (airplane ℄) (6%)

Thus the cold thrust unbalance to the wing flaps was 102 lbs. in opposition to the engine hot thrust. A reversed condition was flown, with the aileron blowing unchanged but with the flap blowing reversed at 102 lbs. asymmetry adding to the engine hot thrust moment. This reversed flap blowing condition resulted in the engine-out situation shown in Figure 3.1-13. At 60 KTS, landing flaps and emergency power, approximately 75% of the available roll control was required to statically balance the engine-out condition.

For an individual pilot in a number of engine-out conditions there was considerable data scatter in the results - both because of technique and speed and attitude variation between each run. However, the general trend of events showed that a reversed asymmetric blowing required considerably more roll control to balance (50° to 70° δ_W compared to 35° to 45° δ_W, technique A). The initial roll excursions often required full wheel to arrest the motion, and in several instances this saturated control system condition led to a hazardous pilot induced oscillation being set up near the ground. Also the evidence suggests that with reversed blowing there was a larger lateral offset from the runway centerline before the engine failure could be brought under control. In some instances this offset was larger than 200 feet, which would make a landing from an engine failure at low altitude extremely difficult. One pilot's comments were: "Listen, it would be pretty hazardous to try to continue the approach, I think. I felt I was quite marginal on control trying to land the airplane".

Technique B of course showed little difference between normal and reversed blowing. However during the landing with $\nu = 18°$, this pilot on two occasions tried to use the nozzle to control rate of descent. The large rolling moments induced with this level of reversed blowing immediately set up the lateral P.I.O and in both cases the pilot was unable to control the airplane to wings level for the touchdown, as shown in Figure 3.1-14.

Reversed blowing asymmetry brings the lateral control required to balance the engine failure close to the maximum available. This increases the probability of setting up an oscillation saturating maximum control which could be hazardous. Raising the thrust vector nozzles could alleviate the roll problem in exchange for control problems about the other axes. The simulator study showed the reversed flap blowing which works against engine-out control is unacceptable.

The simulator study verified that proper asymmetric blowing was helpful for engine failure conditions. Engine-out conditions can be controlled at STOL operational speeds. Considerable design effort has been devoted to assuring that the air ducting system delivers the proper blowing distribution.

3.1.10 Hydraulic System Failures

The lateral control system is powered by two hydraulic systems with manual reversion to the ailerons as a back-up system. A single hydraulic system failure results in the loss of either the chokes or spoilers (but not both). According to the way the three control surfaces were programmed with the pilot's wheel, the loss of the spoiler hydraulics presents the largest degradation in roll control for small wheel inputs. Figure 3.1-15 illustrates this characteristic at the approach condition. The aerodynamic data used in the simulator gave a reduced wheel sensitivity (50% reduction at 20° δ_W, 30% reduction at 40° δ_W), and slightly greater adverse yaw (but still near zero) with the spoilers not operation.

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This condition was flown SAS on, with an engine failure and in gusts, and also SAS off at STOL approach conditions ($\delta_F = 75°, V_{APP} = 60 kts$). Examination of the records shows no apparent degradation in control capability except for slightly larger wheel inputs required to generate acceptable roll rates. Pilots evaluated the airplane using different size wheel inputs in order to be sure that the non-linear control effectiveness with the spoilers inoperative was not a handling qualities problem.

Pilot comments were that non-linearities in rolling moment caused no problems. The airplane was still quite manageable without spoilers, and there seemed to be adequate control power for a failure condition. The pilots would not hesitate to land the airplane in this condition.

The manual reversion mode was assumed to have been caused by the total loss of both hydraulic systems. Therefore operation of the lateral controls produced aileron motion only - the chokes and spoilers being locked in the down position. It was assumed that rudder hinge moments were so high that the poor mechanical advantage that the pilot has would produce no rudder deflection at all, and no rudder trim capability. Friction forces in the lateral control system were judged high enough to prevent lateral trim from working the controls. The SAS actuators were assumed to be the hydraulically operated type that lock to zero displacement when supply pressure is lost at the actuators.

The simulator cab force-feel system was not capable of producing the full characteristics of the force versus wheel displacement curves as laid out in the simulator spec. A much simplified force simulation without friction was therefore used and is shown in comparison with the predicted airplane characteristics in Figure 3.1-16. The feel forces simulated were independent of airplane speed and configuration. The fidelity of the manual reversion feel force simulation was thus not as good as hoped; however, the force level was made as high as anticipated in the augmentor wing flight test vehicle.

Two pilots investigated manual reversion landings at various flap configurations and speeds in both calm air conditions and with a moderate level of simulated turbulence. Both pilots were first asked to attempt a landing in the STOL configuration, flap 75°, 60 knots approach speed. The airplane was simulated with the lower than nominal values of moments of inertia for all three tests.

The analog traces revealed a distinct difference between the $C_{\ell_\beta} = 0$ and $C_{\ell_\beta} = -.25$ cases for the STOL configuration. The very unstable spiral mode associated with $C_{\ell_\beta} = 0$ requires full control authority to keep the wings level. Generally the pilots used pulse-type wheel inputs to keep the bank angle excursions below about ten degrees. Approximately 40° - 60° δ_w was required to do this with at least one full wheel input on each approach. Peak roll rates used were of the order 5 to 6°/sec, with peak roll accelerations of .15 rad/sec^2. Close to the ground larger wheel inputs were required to ensure wings level at touchdown.

The more stable spiral condition at $C_{\ell_\beta} = -.25$ was much easier to control, there being no apparent tendency to excite the less damped dutch-roll oscillation. On the average, wheel inputs for control were of the order 20° to 40° δ_w, with occasional use of 60° δ_w in maneuvering. Roll rates were generally between 2° and 4°/sec with roll accelerations of about .10 rad/sec^2.

Flying either of these conditions in mild turbulence created a heavy pilot workload and Cooper ratings for these conditions were in the region of 8 to 9.

To evaluate whether the high wheel forces were inhibiting landings in manual reversion, several landings were made with the low boost-on forces but with only ailerons for lateral control. It was judged that lack of adequate control with only ailerons was the largest factor in precluding manual reversion landings at STOL speeds.

The pilots were then asked to search for an approach configuration with reasonable handling qualities in the manual reversion condition simulated. Wheel forces shown to the pilots were independent of airplane speeds and configuration - airplane dynamics were the varying features in these runs.

Very little improvement was found at the flap 50° configuration. Approaches were conducted at 90-110 knots with flaps 30° and the improvement there was quite noticeable. With $C_{l\dot{\beta}}$ =0 the approach was accomplished using $30^\circ \delta_w$ maximum, with roll rates of 2°/sec and roll accelerations of .04 rad/sec^2. Turbulence increased these values a little but the task was still accomplished within a reasonable level of pilot workload. At the higher approach speeds there was a tendency to induce pitch oscillations when nearing touchdown and both pilots finally settled on touchdown speeds of 90-95 knots as being <u>satisfactory</u> for this condition. After the <u>first</u> system failure on the airplane, the pilots will be instructed to seek this <u>emergency configuration</u> (flaps 30°) for landing at 90-95 knots. The emergency configuration provides the highest level of safety attainable in the event of a second system failure.

One of the significant problems associated with manual operation was the lack of rudder control. The pilots complained about this deficiency. The present

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rudder design does not include manual reversion capability. Changes to the rudder system are beyond the scope of the basic modification program, and no action in this area is anticipated.

3.1.11 Burst Air Duct

Engine fan air was supplied to the flap, aileron and fuselage via two air ducts from each engine. One duct delivered 40% of the cold thrust to the flap panels aft of the engine. The other line distributed 60% of the cold thrust through a fuselage crossover duct to the body BLC and flap and aileron blowing on the opposite wing. If this "60%" crossover duct were to break inside the body and if the other "40%" line were to continue to operate normally, then a significant asymmetric blowing condition would result. At the time of the simulator study, such a duct failure was deemed possible. Interpretation of the system design at the time of the simulation indicated that the remaining 40% line would continue to operate normally, an assumption which turned out to be more severe than the actual situation. Figure 3.1-17 illustrates the burst duct blowing distribution and resulting rolling moment placed on the airplane. Very nearly 100% of the existing lateral control capability would be required to statically trim this condition at STOL speeds. This burst duct situation was simulated and shown to one of the pilots.

The failure simulated was an instantaneous bursting of the 60% flow crossover duct. The asymmetric lift and drag (causing rolling and yawing moments) due to the failure were faithfully simulated. Lift loss and changes in drag and pitching moment from the ailerons and changed lateral control effectiveness were simulated as well. The simulation did not include the loss of lift and

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drag and the nose-up pitching moment changes due to overall loss in C_j, on the two wings taken together. The simulation assumed that the center of pressure of asymmetric lift was at the geometric center of the flap span. Statically, for the condition simulated ($\delta_F = 75^\circ$, 60 Kts, $C_j = .49$, $\gamma = -7.5^\circ$), the out-of-balance rolling moment required nearly 100% of the available lateral control to trim.

In the dynamic situation with the resulting upset in bank angle, the pilot was unable to recover the airplane at 60 kts. By deliberately pushing the nose down and increasing speed, the pilot was able to recovery; however, 950 feet altitude loss occurred in the maneuver. The height loss sustained before the bank angle could be brought back to level was 700 feet.

Since the time of the simulator study several steps have been taken to alleviate the burst duct problem. "Flow limiters" in the ducts have been deleted from the design. If a burst duct should occur, airflow to the remaining duct would be lessened considerably thereby reducing the rolling moment. Analysis of the burst duct condition was also revealed that shut down of the "bad" engine or throttling up the "good" engine would tend to balance the situation. Increased emphasis has been placed on safe-life design and the use of dual load path where feasible. The possibility of a duct burst is being made extremely remote by design and by specifying frequent inspection for possible cracks in the ducting.

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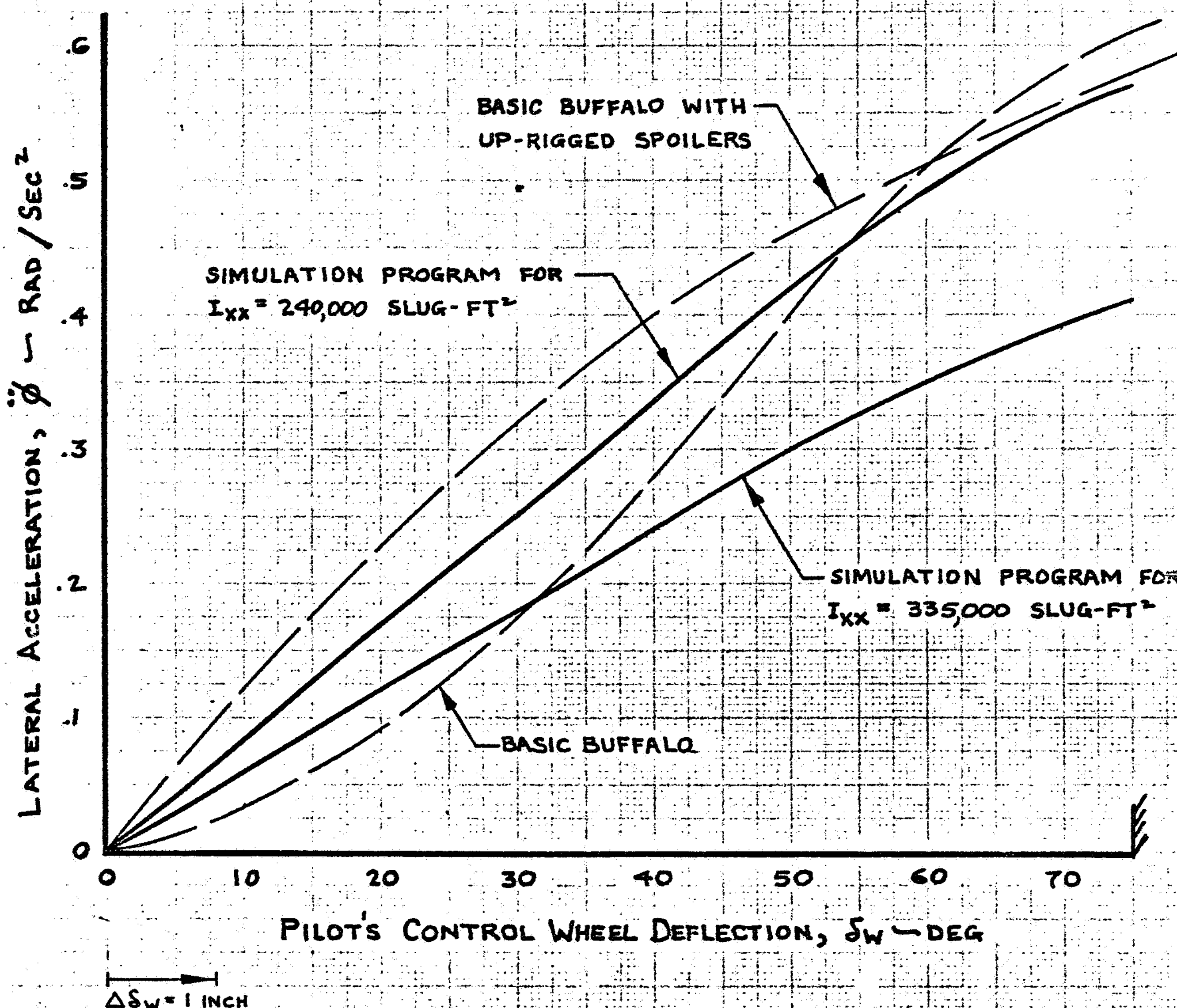
ROLL ACCELERATION CAPABILITY AT LANDING APPROACH

NOTE: 1. AUGMENTOR WING F.T.V. FLIGHT CONDITION: $\delta_F = 75°$, $V_e = 60$ KTS, $\alpha_F = 0°$

2. ROLL ACCELERATION CALCULATED AT ZERO ROLL RATE



CALC	SPITZER	1-14-71	REVISED	DATE	ROLL ACCELERATION CAPABILITY WITH WHEEL DEFLECTION AT LANDING APPROACH	D6-24806I
CHECK						FIG 3.1-1
APR					THE BOEING COMPANY	PAGE 3.22
APR						

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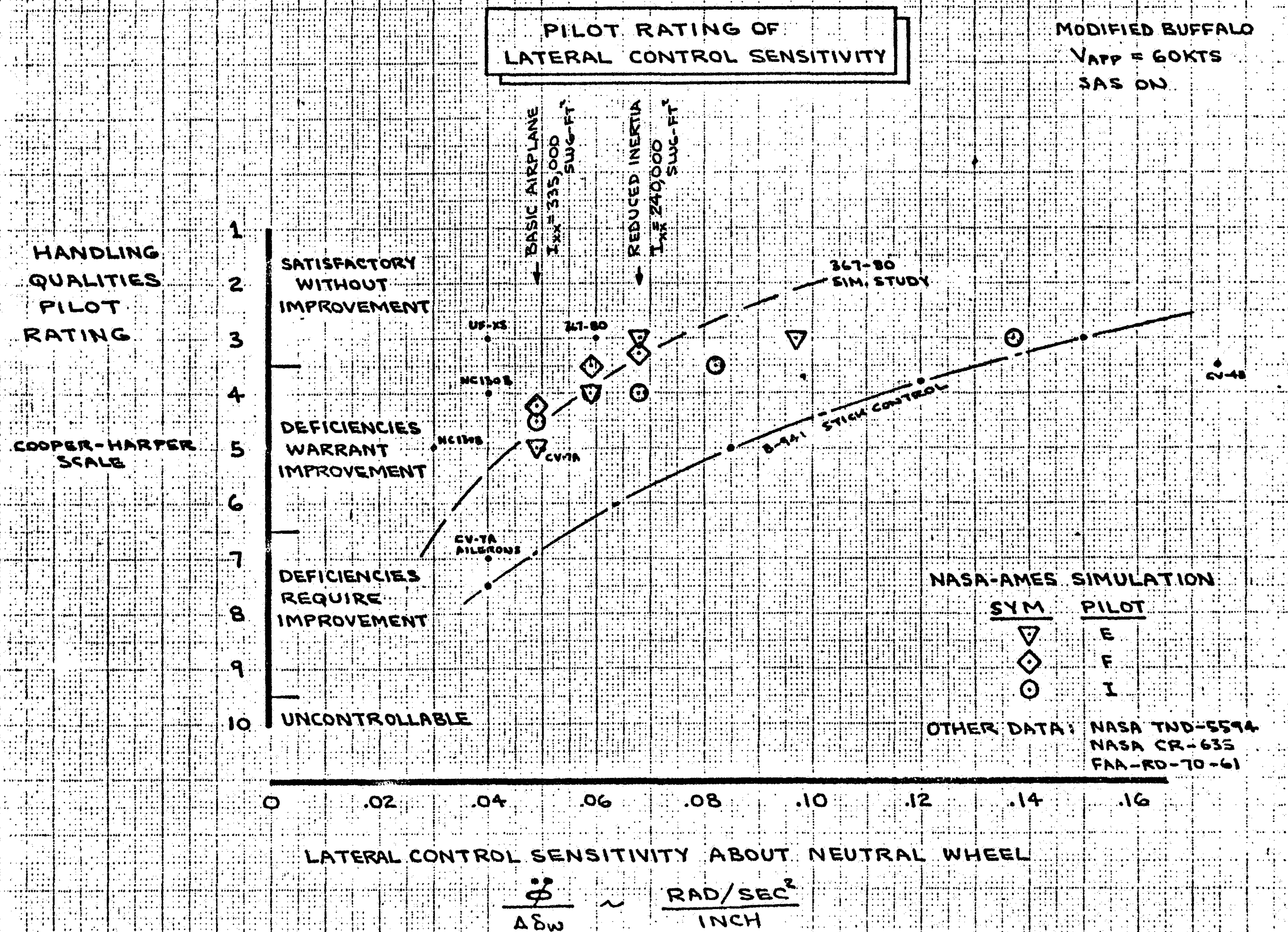

CALC	SPITZER	12-11-70	REVISED	DATE
CHECK				
APR				
APR				

PILOT RATING OF LATERAL CONTROL SENSITIVITY

THE BOEING COMPANY

D6-24806

FIG 3.1-2

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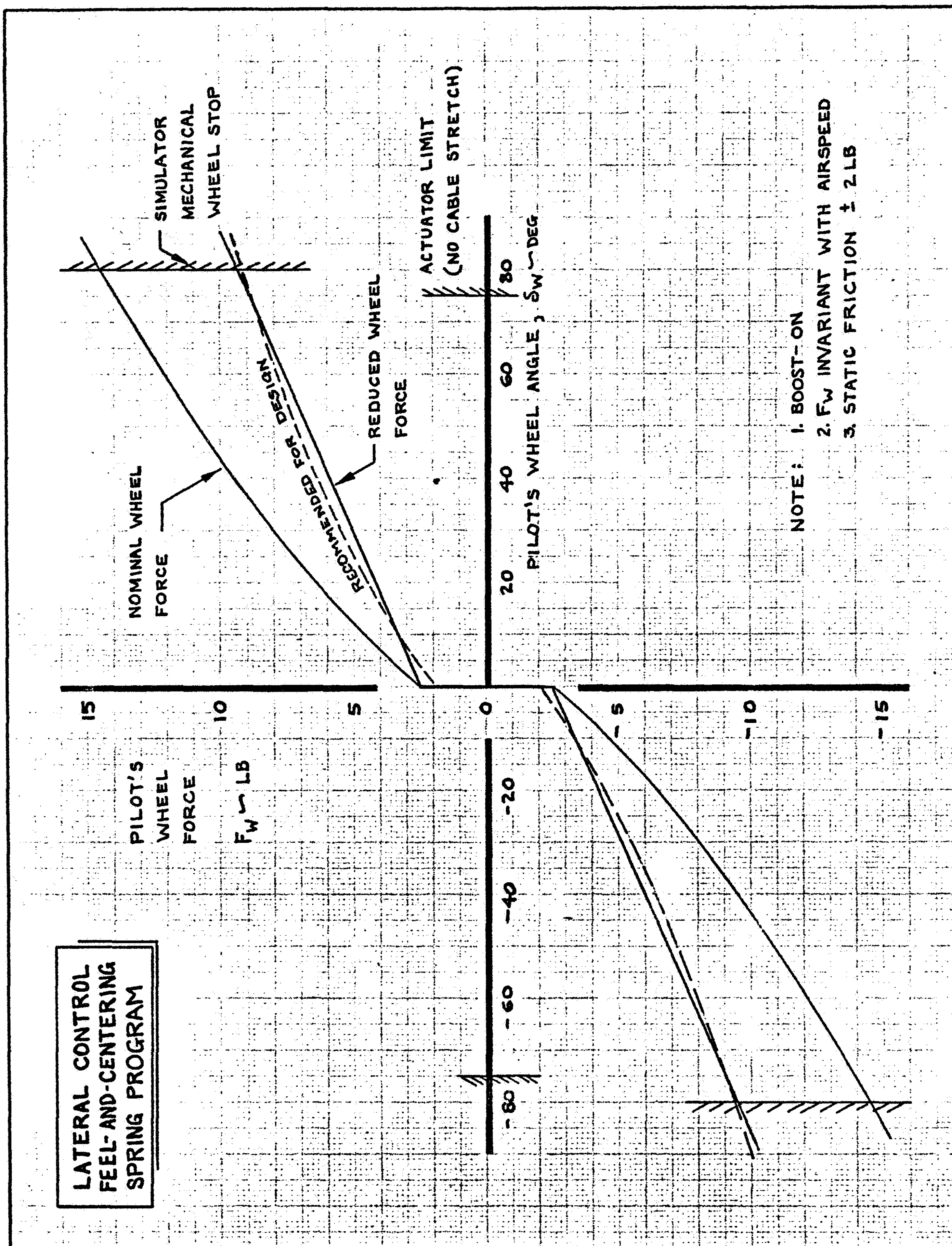

CALC	SPITZER	1-14-71	REVISED	DATE	LATERAL CONTROL FEEL-AND-CENTERING SPRING PROGRAMS USED IN THE SIMULATION	D6-24806
CHECK						FIG 3.1-3
APR					THE BOEING COMPANY	PAGE 3.24
APR						

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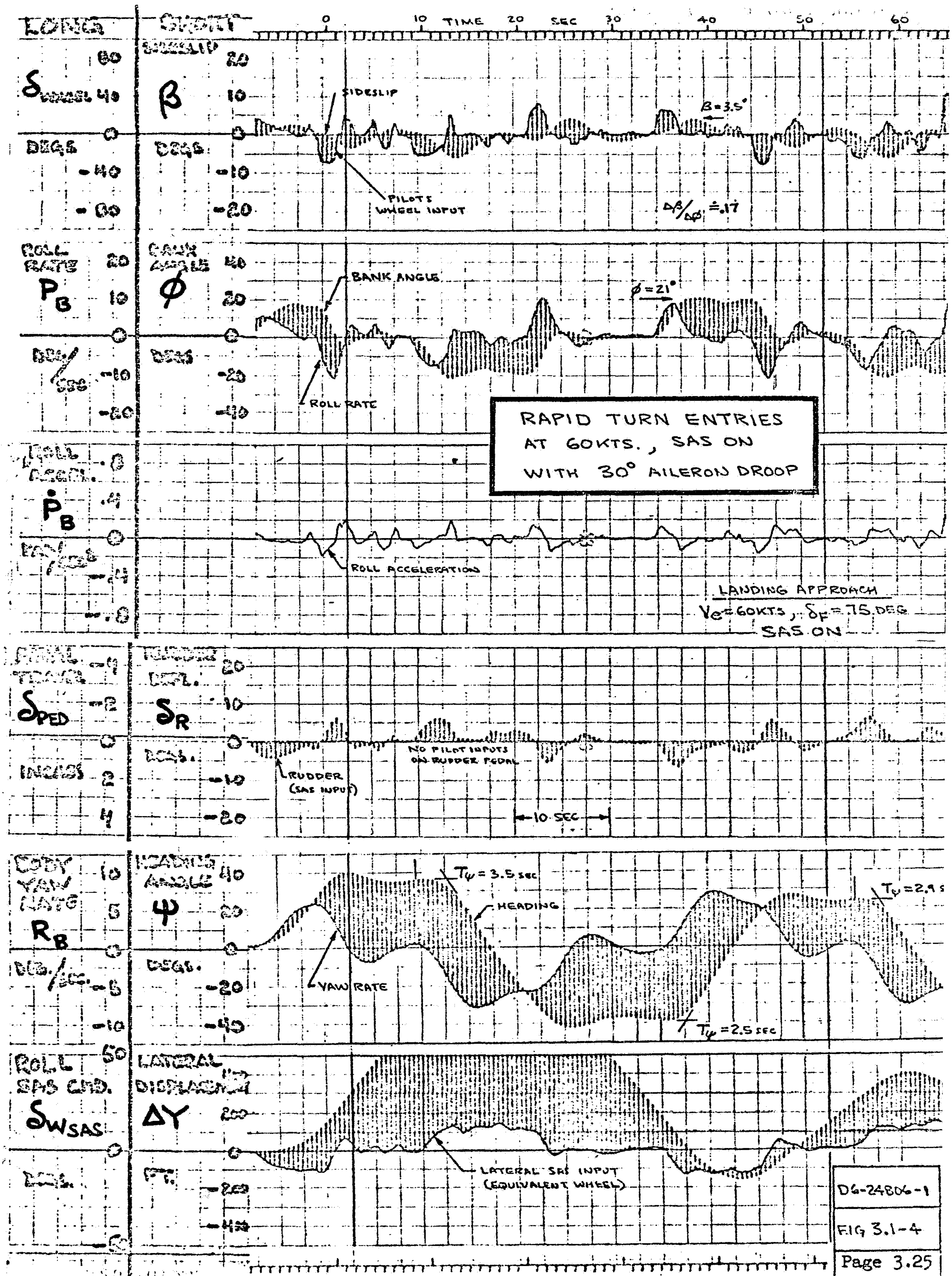
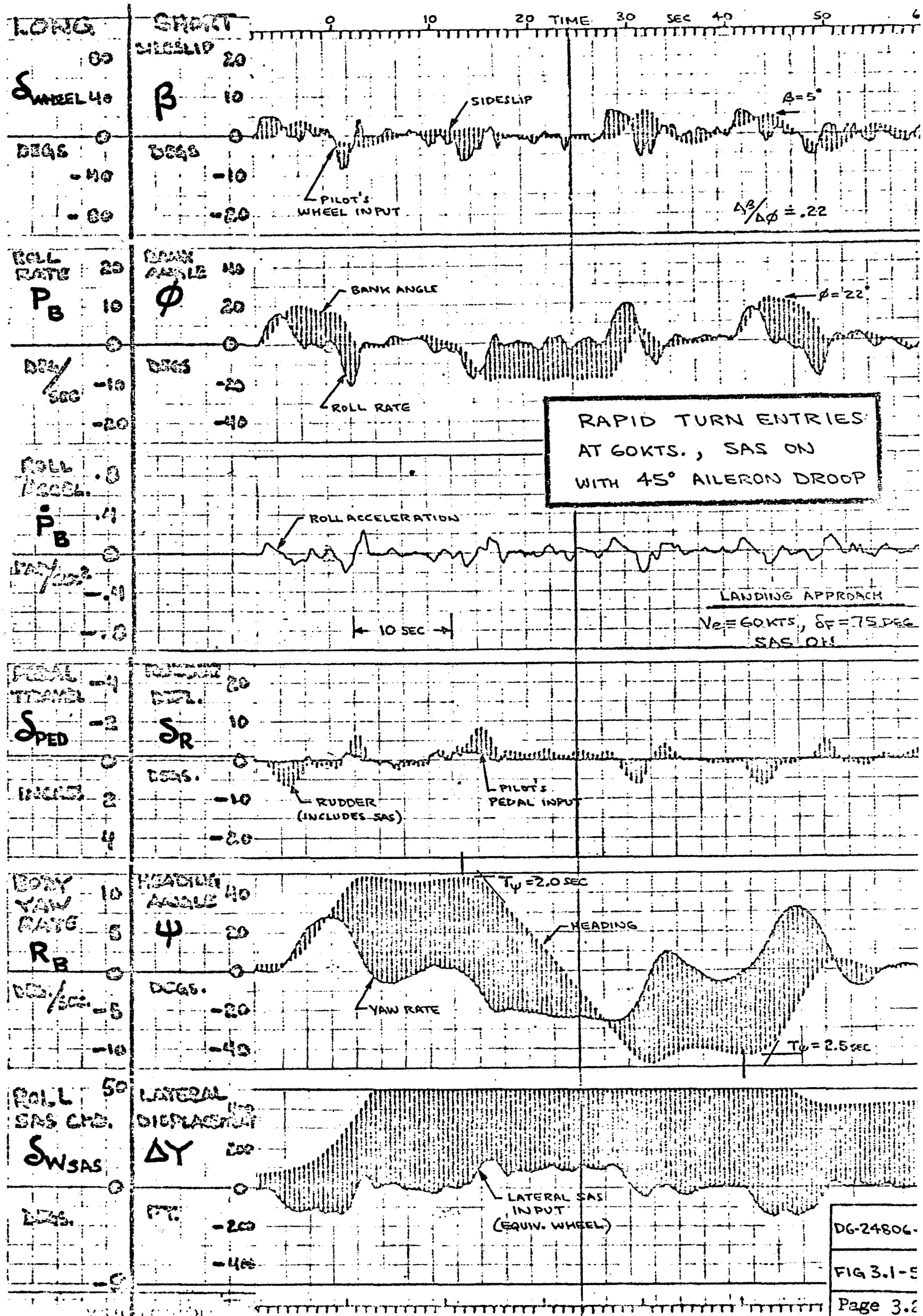
K+E ALBANENE 1956 6.5 G-6000 TRACING PAPER



D6-24806-1

FIG 3.1-4

31



DG-24806-

FIG 3.1-5

Page 3.2

LATERAL CONTROL INFLUENCE ON BLOWING DISTRIBUTION

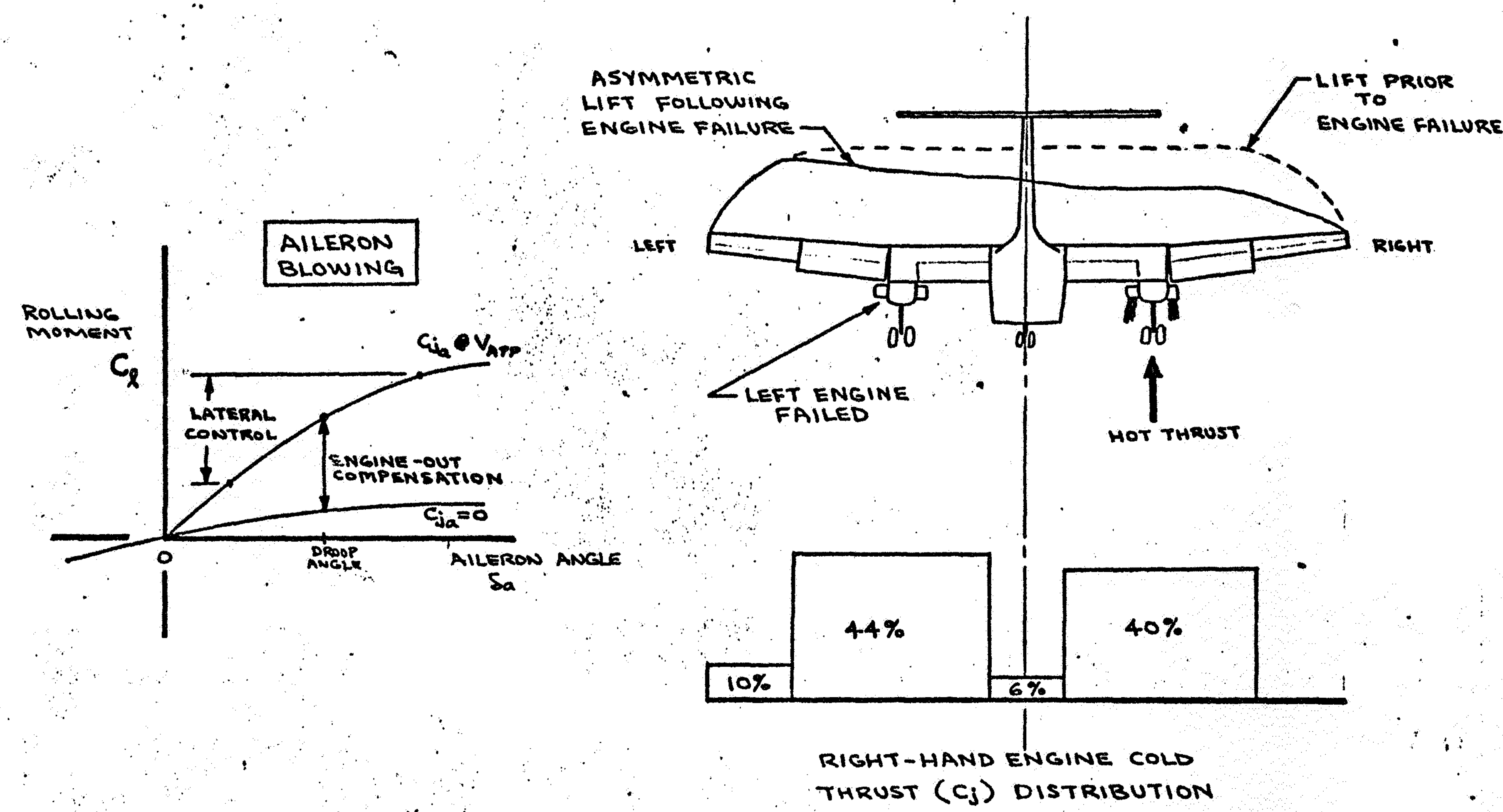
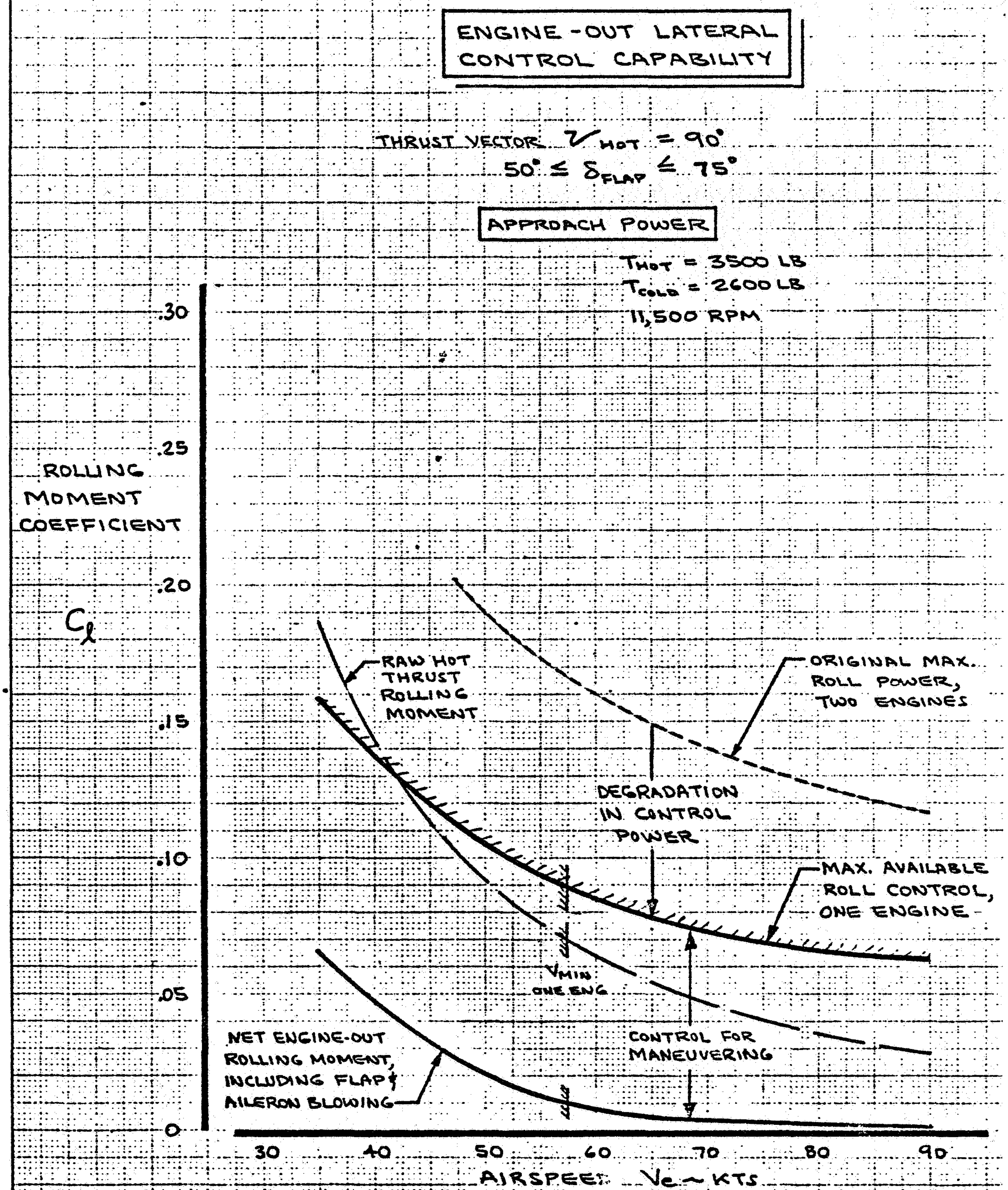


FIG 3.1-6

39



CALC	SPITZER	9-28-70	REVISED	DATE
CHECK				
APR				
APR				

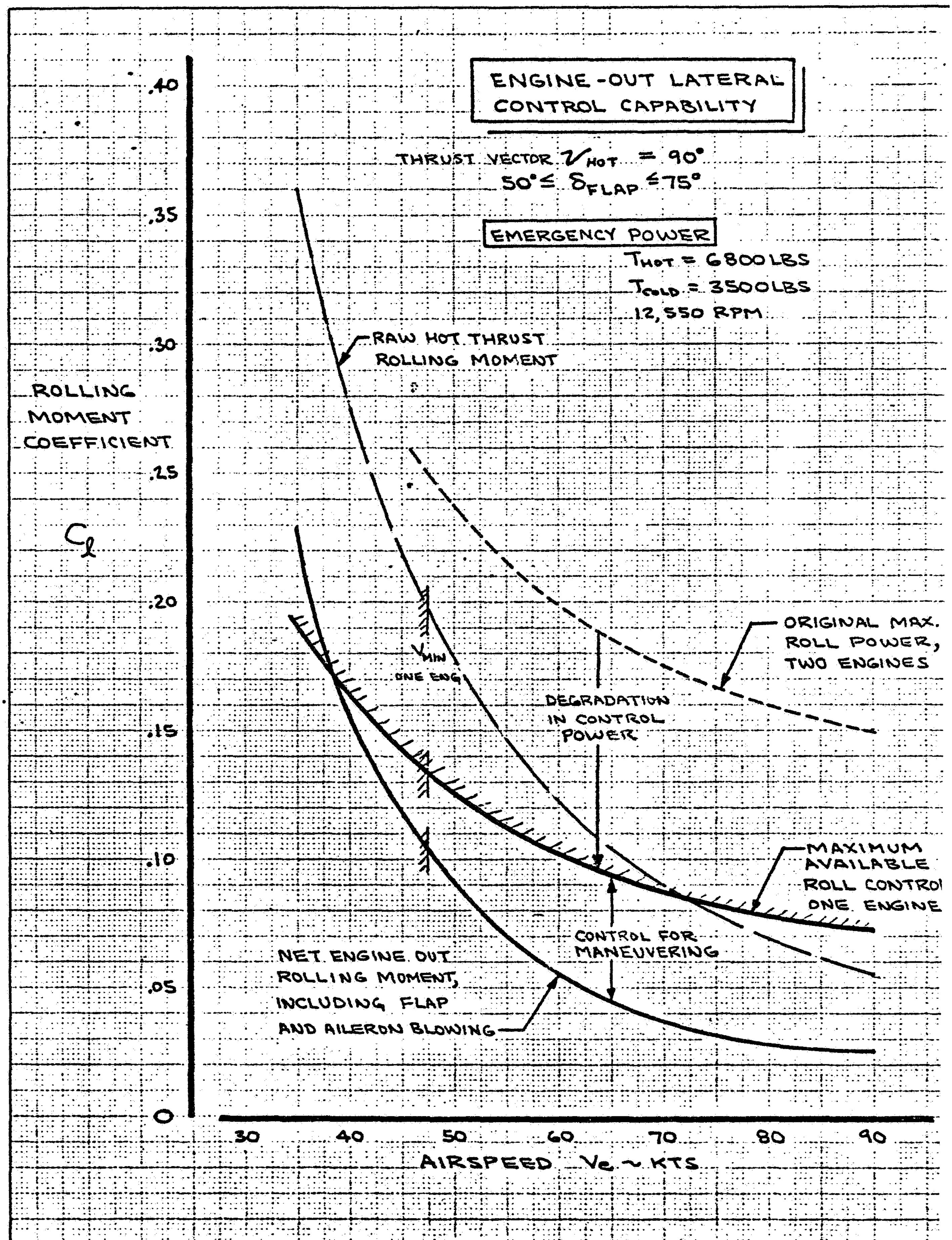
LANDING APPROACH ENGINE-OUT LATERAL CONTROL CAPABILITY AT APPROACH POWER SETTING

THE BOEING COMPANY

D6-24806-

FIG. 3.J-7

40



CALC	SPITZER 9-28-70	REVISED	DATE	LANDING APPROACH ENGINE-OUT LATERAL CONTROL CAPABILITY AT EMERGENCY POWER SETTING	D6-24806
CHECK					FIG 3.1-1
APR				THE BOEING COMPANY	PAGE 3.29
APR					

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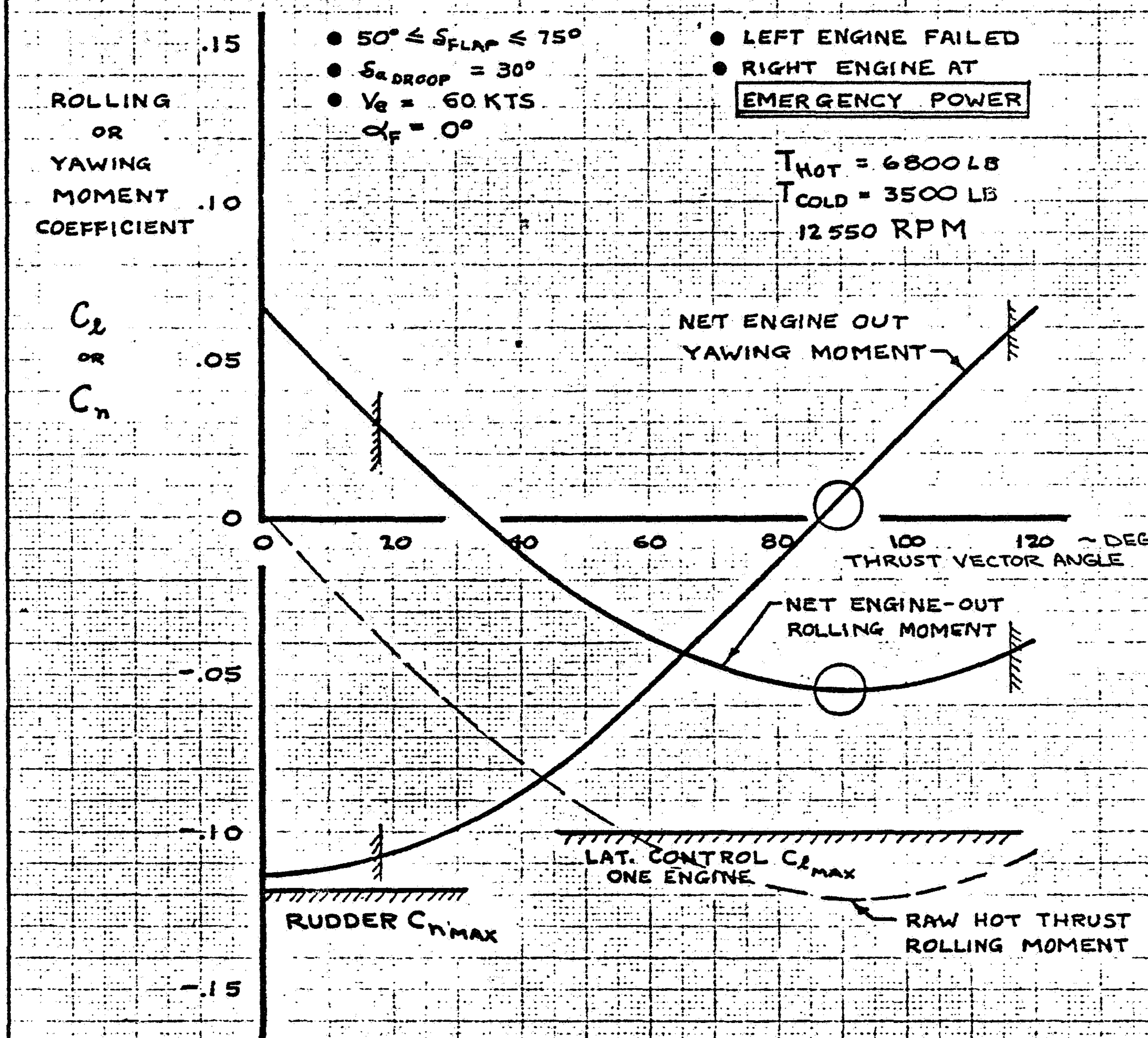
EFFECT OF THRUST VECTOR ON ENGINE-OUT CONTROL AT LANDING FLAPS



CALC	SPITZER	12-11-70	REVISED	DATE	EFFECT OF THRUST VECTOR ANGLE ON ENGINE-OUT CONTROL AT EMERGENCY POWER SETTING	D6-24806-1
CHECK						FIG 3.1-9
APR					THE BOEING COMPANY	PAGE 3.30
APR						

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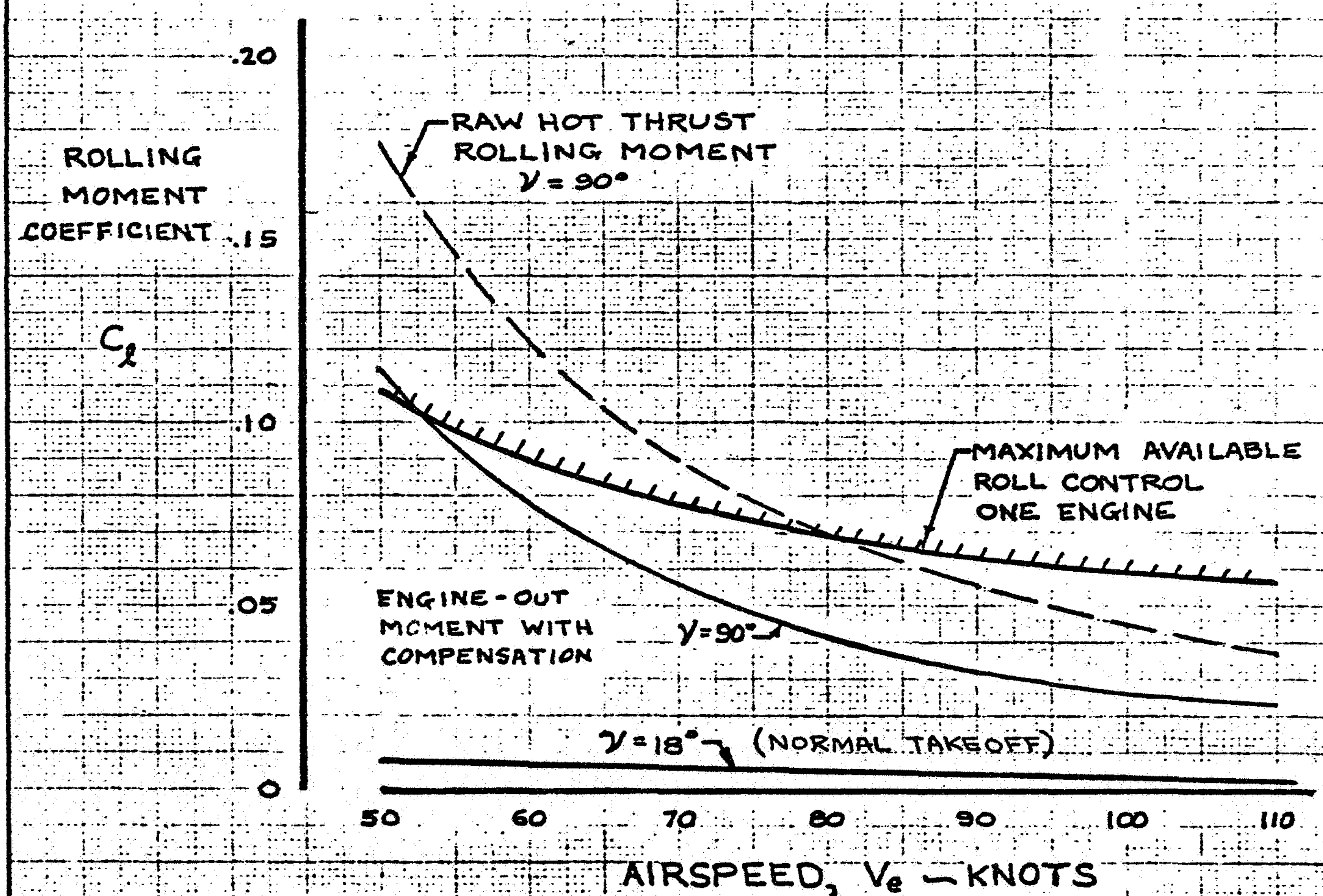
ENGINE – OUT LATERAL CONTROL CAPABILITY AT TAKEOFF FLAPS

$\gamma = 18° \& 90°$

$\delta_{FLAP} = 30°, \delta_{a_{DROOP}} = 20°$

EMERGENCY POWER

$T_{HOT} = 6800$ LB
$T_{COLD} = 3500$ LB
12550 RPM



CALC	SPITZER	12-11-70	REVISED	DATE	ENGINE-OUT LATERAL CONTROL CAPABILITY AT TAKEOFF FLAPS AND EMERGENCY POWER SETTING	D6-24806
CHECK						FIG 3.1-10
APR					THE BOEING COMPANY	PAGE 3.31
APR						

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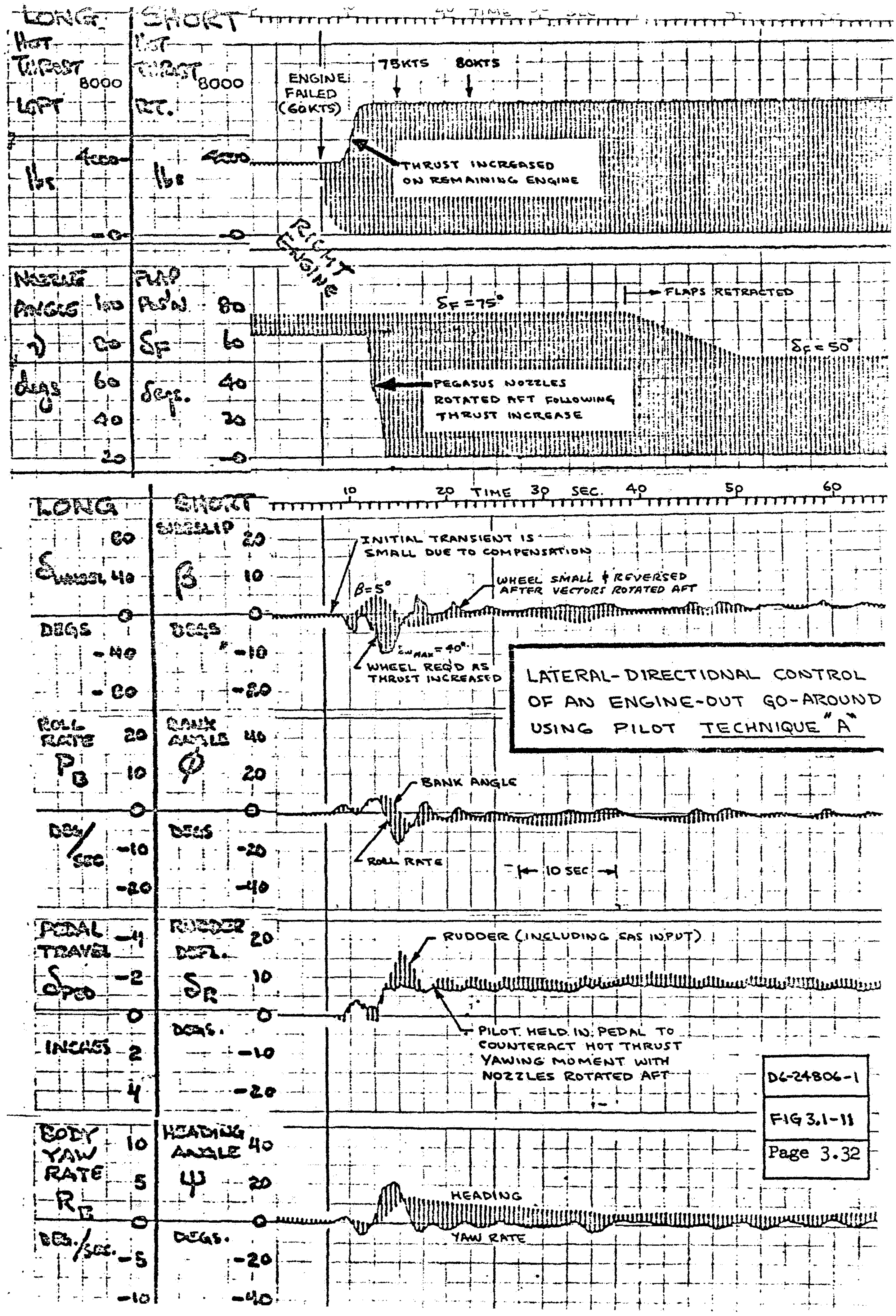
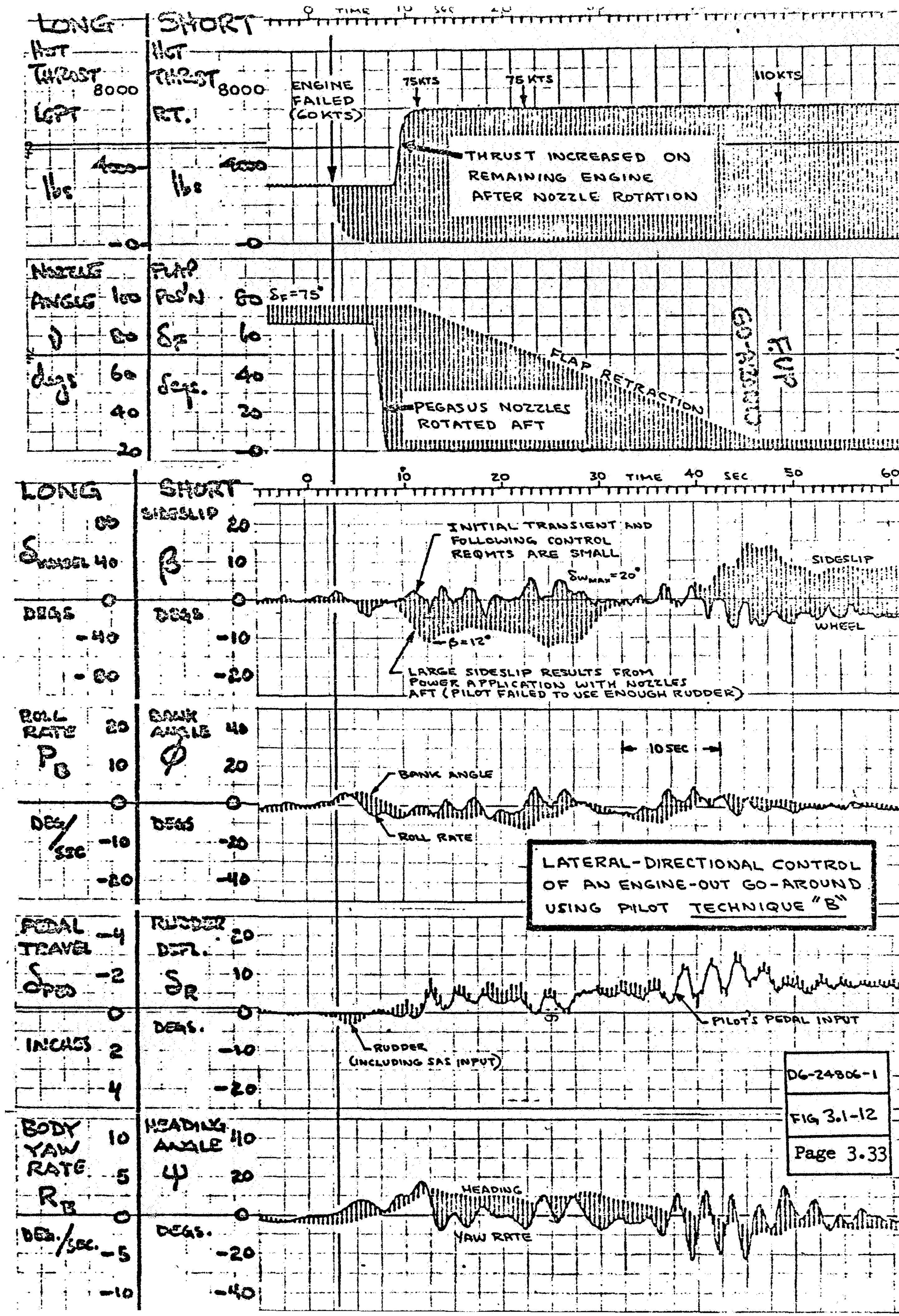


LATERAL-DIRECTIONAL CONTROL OF AN ENGINE-OUT GO-AROUND USING PILOT TECHNIQUE "A"

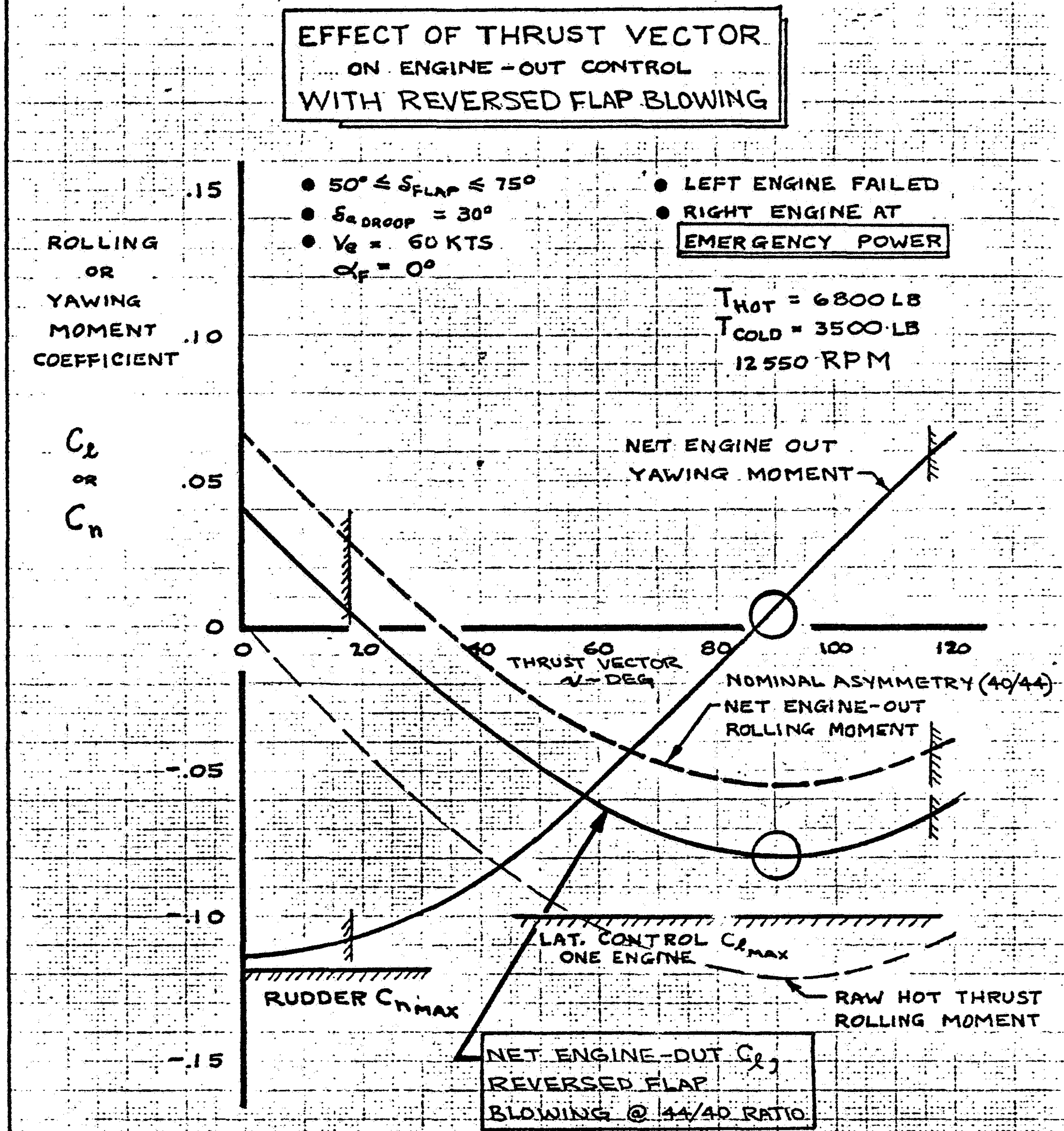
4



LATERAL-DIRECTIONAL CONTROL OF AN ENGINE-OUT GO-AROUND USING PILOT TECHNIQUE "B"

D6-24806-1

FIG 3.1-12



CALC	SPITZER	12-11-70	REVISED	DATE	LANDING APPROACH ENGINE-OUT LATERAL CONTROL CAPABILITY WITH REVERSED FLAP BLOWING ASYMMETRY	D6-24806-1
CHECK			SPITZER	1-14-71		FIG 3.1-13
APR					THE BOEING COMPANY	PAGE 3.34
APR						

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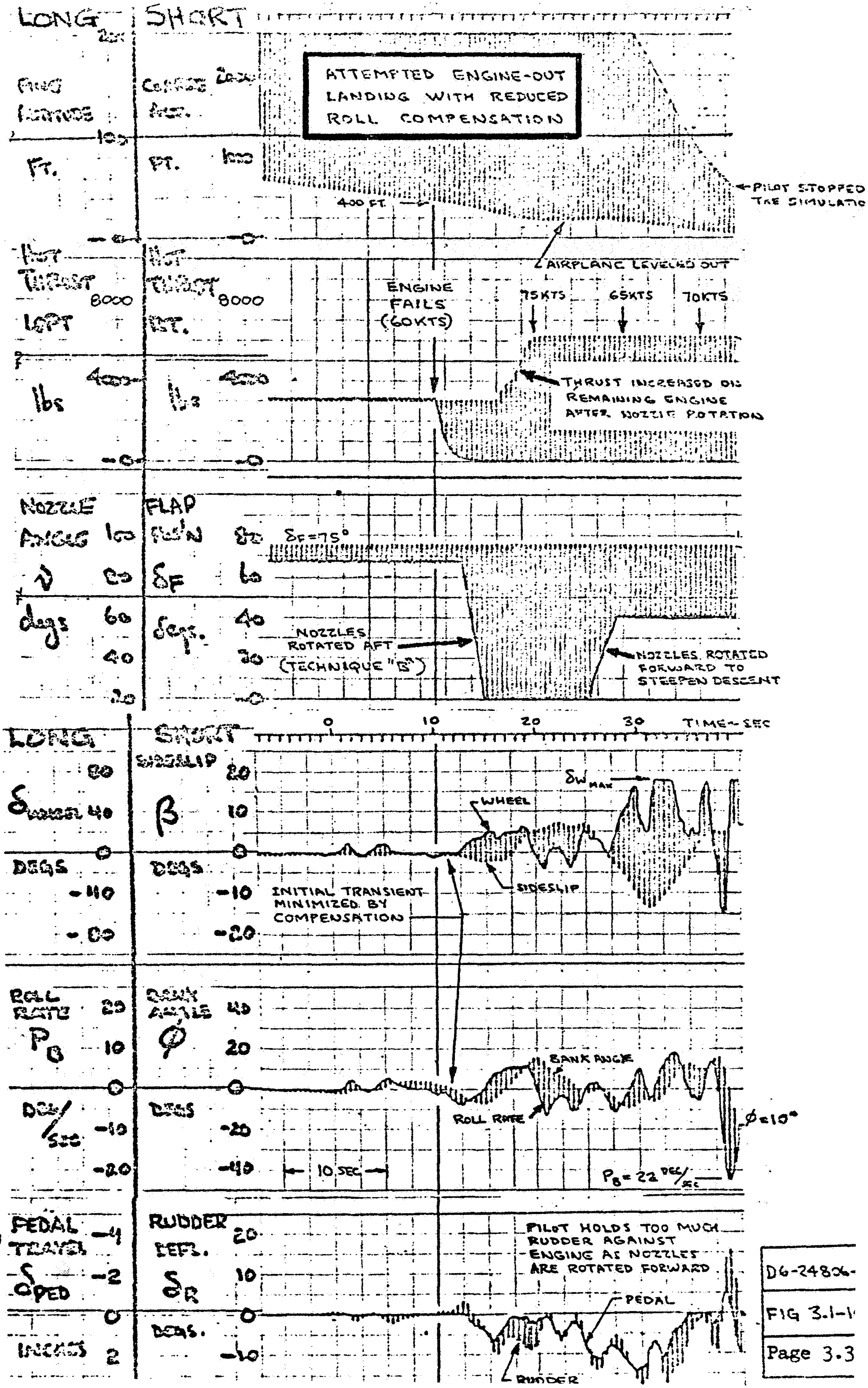

D6-24806-

FIG 3.1-1

Page 3.3

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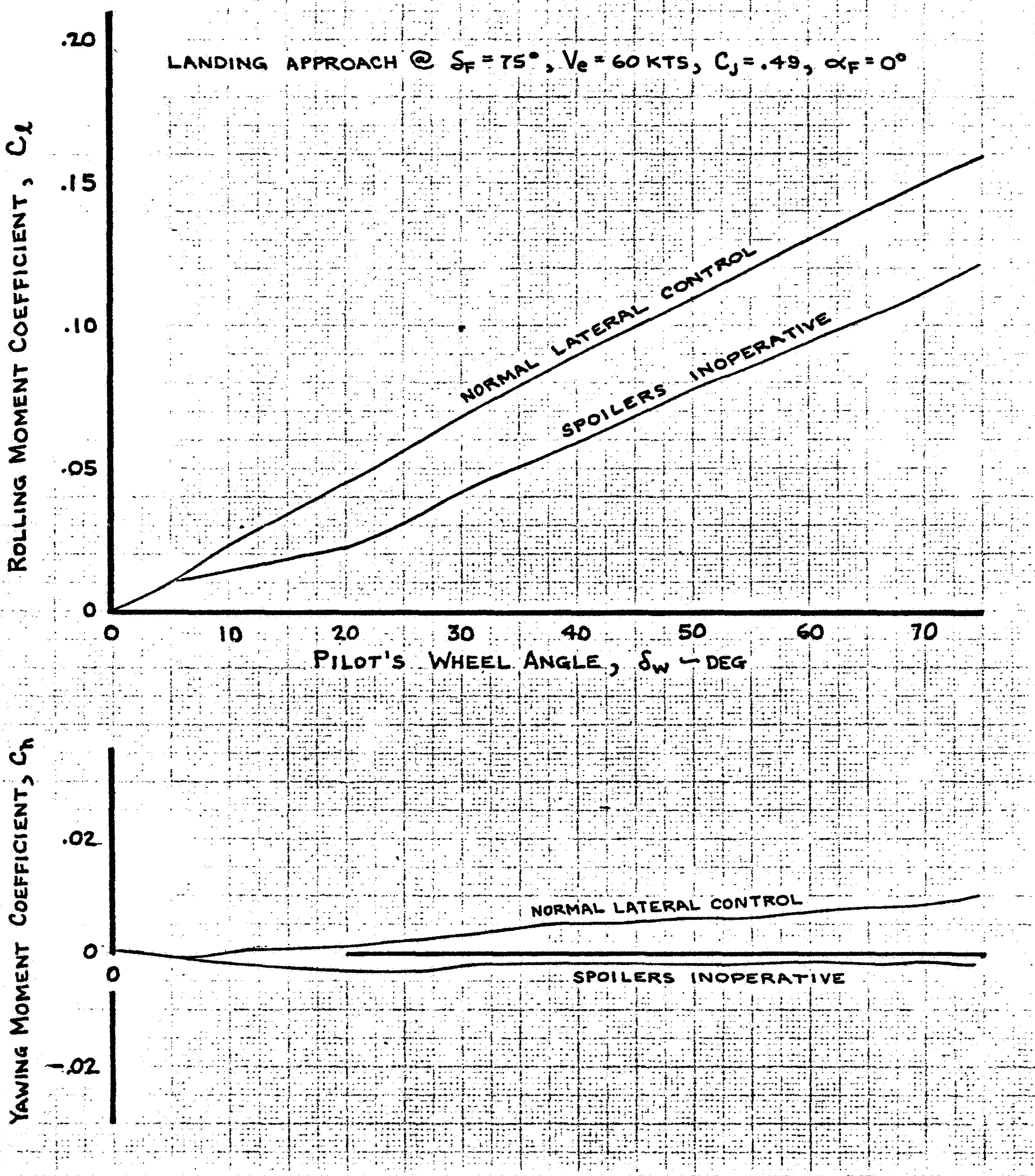
EFFECT OF SPOILER HYDRAULIC SYSTEM FAILURE ON LATERAL CONTROL



CALC	RUMSEY	12-15-70	REVISED	DATE	CHANGED LATERAL CONTROL EFFECTIVENESS WITH INOPERATIVE SPOILERS	D6-24806-1
CHECK			SPITZER	1-14-71		FIG 3.1-15
APR					THE BOEING COMPANY	PAGE 3.36
APR						

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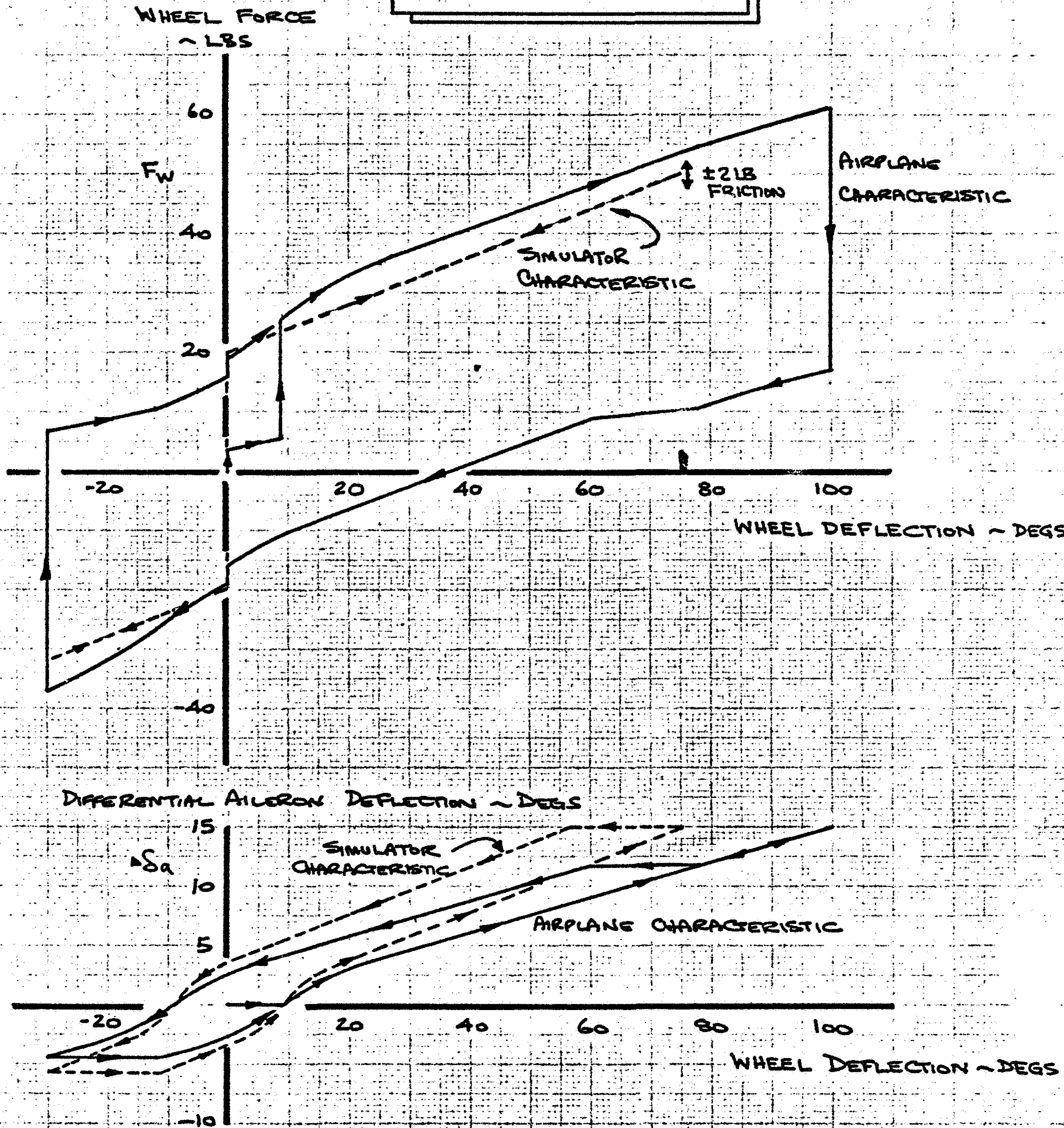
SIMULATOR MANUAL REVERSION CHARACTERISTICS

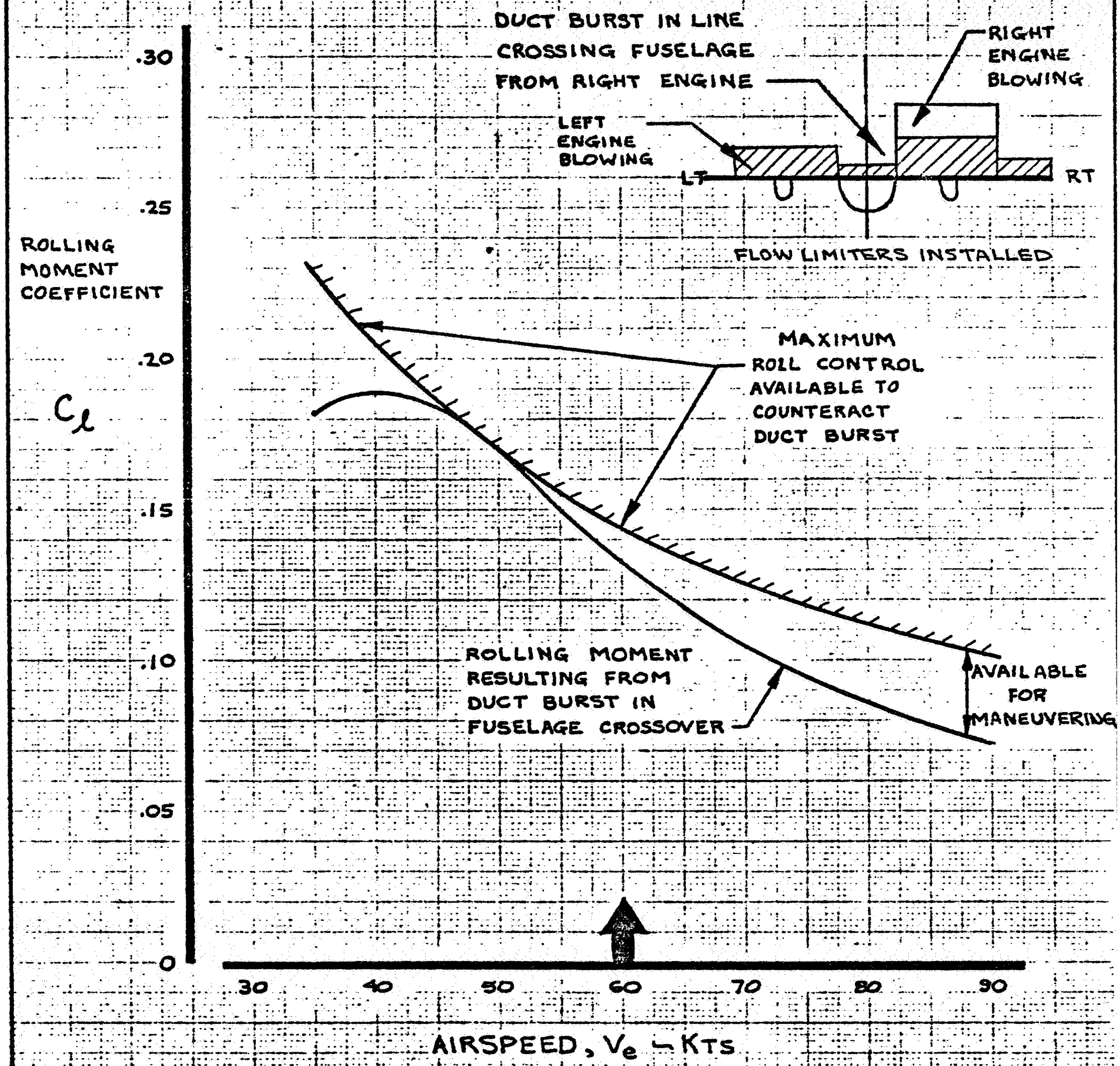


CALC	RUMSEY	12/30/70	REVISED	DATE	SIMULATION OF MANUAL REVERSION CONTROL SYSTEM CHARACTERISTICS	D6-24806-1
CHECK						FIG 3.1-16
APR					THE BOEING COMPANY	PAGE 3.37
APR						

TD 461 C-R4

ROLLING MOMENT DUE TO FAILURE IN "60%" CROSSOVER AIR DUCT
BOTH ENGINES AT APPROACH POWER

$50° \leq \delta_F \leq 75°$
$\delta_{a_{DROOP}} = 30°$
$\alpha_W = 6°$
$T_{HOT} = 3500$ LB/ENG
$T_{COLD} = 2600$ LB/ENG
11500 RPM



CALC	SPITZER	12-11-70	REVISED	DATE	ROLLING MOMENT PRODUCED BY BURST AIR DUCT ON LANDING APPROACH	D6-24806I
CHECK						FIG 3.1-17
APR					THE BOEING COMPANY	PAGE 3.38
APR						

3.2 LONGITUDINAL CONTROL SYSTEM DESIGN

3.2.1 Simulation and Longitudinal Characteristics

Longitudinal aerodynamic characteristics were built up in the simulator computations from tabulated wing-body data, an analytical downwash model and linearized horizontal tail characteristics. Aerodynamic data were a function of angle of attack, flap deflection, aileron droop and cold thrust blowing coefficient (C_j). Hot thrust effects were included directly into the equations of motion including the pitching moment variation produced by thrust vectoring. The control column gearing to elevator was simulated along with limitations on maximum surface deflection. The stick forces associated with the Buffalo spring tab system were presented to the pilot. At 60 KTS maximum stick force for full deflection was approximately F_S = 40 lbs. Stick force varied with airspeed as in the real airplane. Certain simplifications were made in the simulation, and no attempt was made to actually math model the spring tab system. Control column dynamics were adjusted until deemed realistic by the pilots.

All flying was done along the available best estimate of the center of gravity schedule with weight. The CG at OEW = 32500 lbs. was located at 23.5% $\bar{C}$. As fuel was added to increase weight the CG moved aft. At 40000 lbs., where most flying was done, the CG was approximately 26% $\bar{C}$. Weight and CG remained fixed during any particular test run.

Longitudinal characteristics in the STOL flight regime are indicated by the check-out data in Section 7.2. Trim at landing approach ($\delta_F = 75^\circ$, 60 KTS) lies on the "backside" of the drag curve resulting in mildly divergent flight path stability. Operation on the "backside" of the flight path-speed curve required modulation of the thrust vector levers as well as control column during the approach. At all flap settings the airplane had positive static longitudinal stability in terms of elevator-to-trim versus speed. Static stability was degraded by increased C_j and by the nose-up pitching tendency at low thrust vector angles ($\nu = 18^o$). Stall characteristics were not objectionable, but lack of precise data at high angles of attack precluded using anything except smooth pitching moment characteristics in the simulation.

Even though the airplane had positive static stability and more-or-less conventional damping derivatives, longitudinal dynamic response was degraded by the low flight speed. Response was characterized as overdamped and sluggish. Elevator step response is shown in Figure 3.2-1. Airspeed changes occurred almost immediately with control input producing a combined short-period and phugoid response.

One of the most significant aspects of the airplane was the effect of rapid changes in throttle setting and Pegasus nozzle angle. Figure 3.2-2 presents the "hands off" response to sudden power application at 60 KTS in the landing configuration. Increased thrust added immediate lift to the airplane in two ways, direct vertical hot thrust and an increase in wing lift due to cold thrust blowing. The load factor trace shows an instantaneous .2g increase in load factor caused by running both engines up to emergency power. Without pilot input the airplane reduced angle of attack by about $\Delta\alpha = -9^o$ and went into its 18 sec phugoid. Average airspeed was actually less than the original trim value. Power application on landing approach (vertical hot thrust) produced considerably different response than seen in a conventional airplane.

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The Pegasus nozzles were located 30 inches below and slightly aft of the center of gravity. On landing approach with the nozzles pointed downward, the pitching moment produced by hot thrust was quite small. Rotating the Pegasus nozzles aft then produced a substantial nose-up pitching moment transient. The pilots disliked this effect and universally complained about it. Figure 3.2-3 illustrates the airplane's response to a rapid nozzle rotation from $\nu = 116^{\circ}$ to $\nu = 18^{\circ}$. The immediate effect of the pitching moment was an increase in angle of attack and pitch rate. After a delay load factor increased and the airplane began to climb. (In cases where $\nu = 90^{\circ}$ initially, load factor actually decreased momentarily upon nozzle rotation as the hot thrust lifting force was taken away.) For the flight condition shown in the Figure, the pilot applied down elevator to prevent an excessive pitch attitude increase. (The airplane, if left unattended, would over-rotate, slow down and stall). With nozzle rotation the airplane had sufficient thrust to increase angle of attack and airspeed simultaneously.

3.2.2 Pilot Induced Oscillations

Very early in the study pilot induced oscillations were experienced in the pitch plane. This P.I.O. phenomena was unforseen prior to the study. All pilots experienced it during early familiarization flying and at times later on in the study. Any rapid pitching moment input or change in flight condition could set off P.I.O. In particular, Pegasus nozzle rotation was most often the causing factor. Figure 3.2-4 presents a typical example of a P.I.O. condition which occurred during an engine-out go-around maneuver. The P.I.O. period was approximately 3 seconds, and the amplitude very often would remain relatively constant. P.I.O. was occasionally encountered near the ground as the pilot prepared to flare. In these cases the amplitude would sometimes increase.

Some attempt was made to reduce the P.I.O. tendency. It was definitely determined that pitch oscillations could not be sustained without the pilot in the loop. The motion system response was reviewed to determine if some motion cue could be a contributing factor. To provide the "feel" for a long-term longitudinal acceleration, the motion cues provided fore and aft cab tilt. Rapid changes in axial acceleration occur on the airplane due to rotation of the Pegasus nozzles. These changes then commanded a rapid change in cab tilt angle. It was reasoned that this angular change could have introduced spurious angular pitching accelerations which confused the pilot. The cab tilt signal was deleted from the motion system drive, and the tendency to induce P.I.O. declined after this change. Unfortunately, P.I.O. continued to occur at times throughout the simulation testing. Whether pilot induced oscillations will occur on the actual augmentor wing flight test vehicle is an open question.

3.2.3 Longitudinal Stick Forces

It is a requirement that the pilot be able to control the airplane in pitch and roll using one hand. The pilots felt that the longitudinal stick forces at 60 KTS were too high for good one-hand operation. High stick forces prevented the pilot from freeing his thumb to operate the trim switch and still maintain precise control. The stick forces were arbitrarily reduced by 50% (max. F_S = 20 lbs. at 60 KTS) to evaluate their effect. (Combinations of reduced lateral and longitudinal forces were tried at this time.) The pilots very much preferred the lower stick forces, and the tendency for P.I.O. was judged to be further reduced.

Reduced longitudinal stick forces require changes to the elevator control system beyond the scope of the basic augmentor wing flight test vehicle program. Action in this area is not anticipated at this time.

3.2.4 Longitudinal Trim

A thumb switch activated electrical trim system was provided in the simulation in keeping with the airplane design. Longitudinal trim rates were evaluated as follows:

EQUIVALENT ELEVATOR TRIM RATE $\dot{\delta}_e$ DEG/SEC	PILOT COMMENTS
1.75	Too slow!!
2.80	Better
3.50	OK for 60 KTS, too fast for 140 KTS

As a result of the simulation, the trim tab motor in the modified C-8A will move the elevator at $\dot{\delta}_e$ = 2.80 DEG/SEC. This faster trim rate should alleviate the high stick force problem to a large extent.

The flap/longitudinal trim interconnect program was not included in the simulation. The elevator angle required for trim throughout the flap transition maneuver was quite small. With adequate trim rate, the trim interconnect was not needed for the transition maneuver. Other considerations, such as excessive nose down trim capability at cruise, will establish the trim interconnect program for the actual flight test vehicle.

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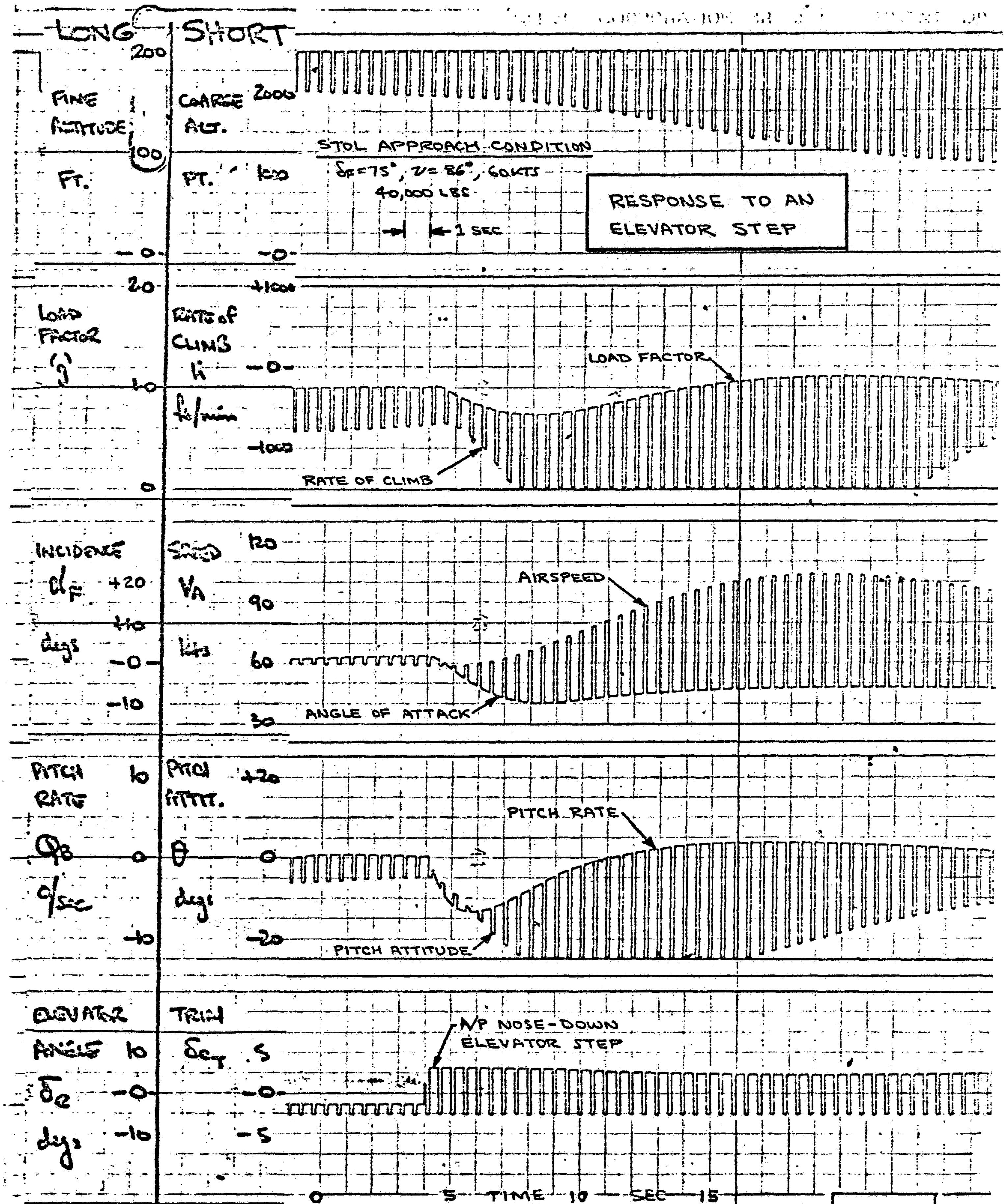
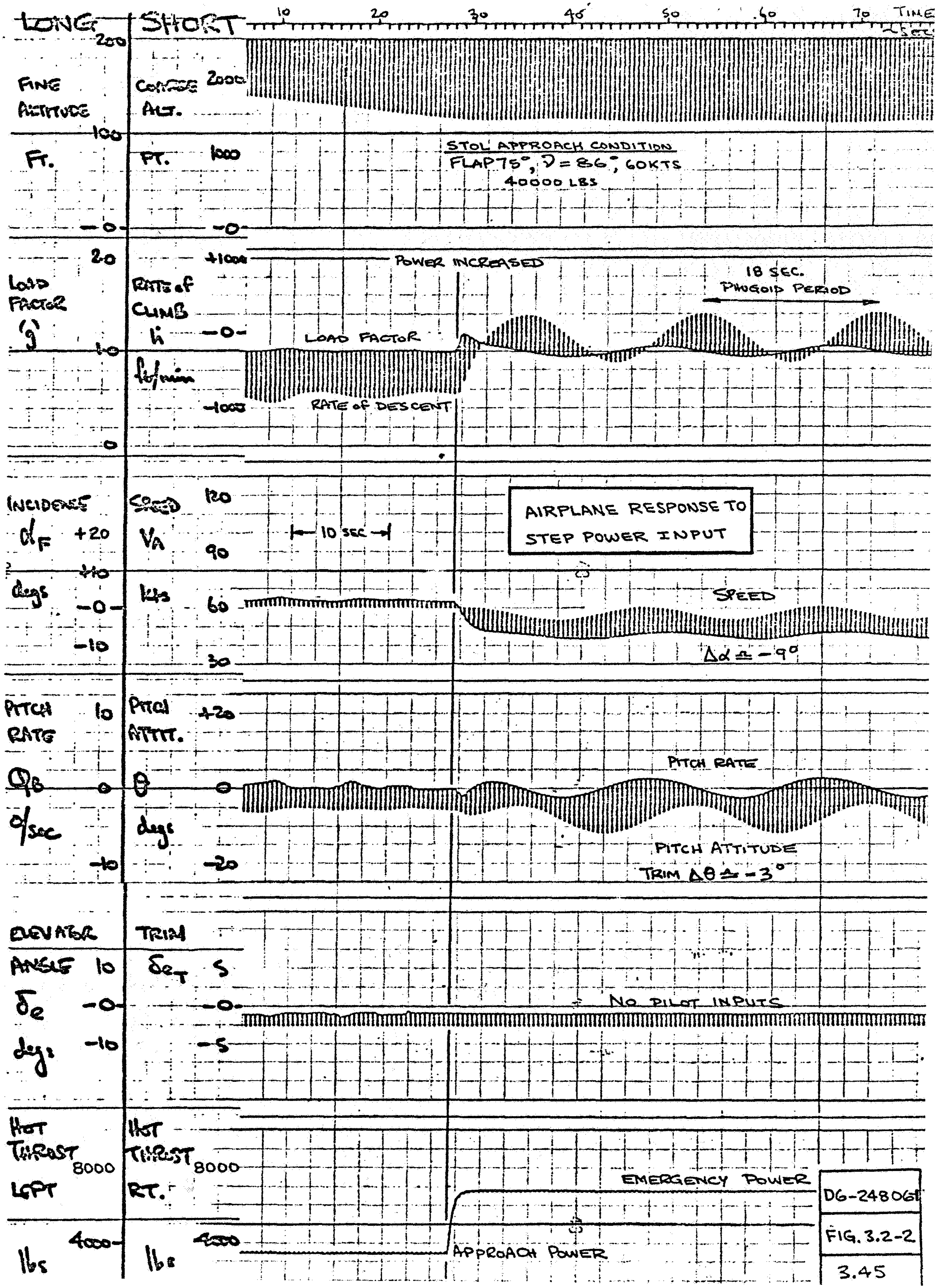


NOTE: EXPANDED TIME SCALE

D6-24806

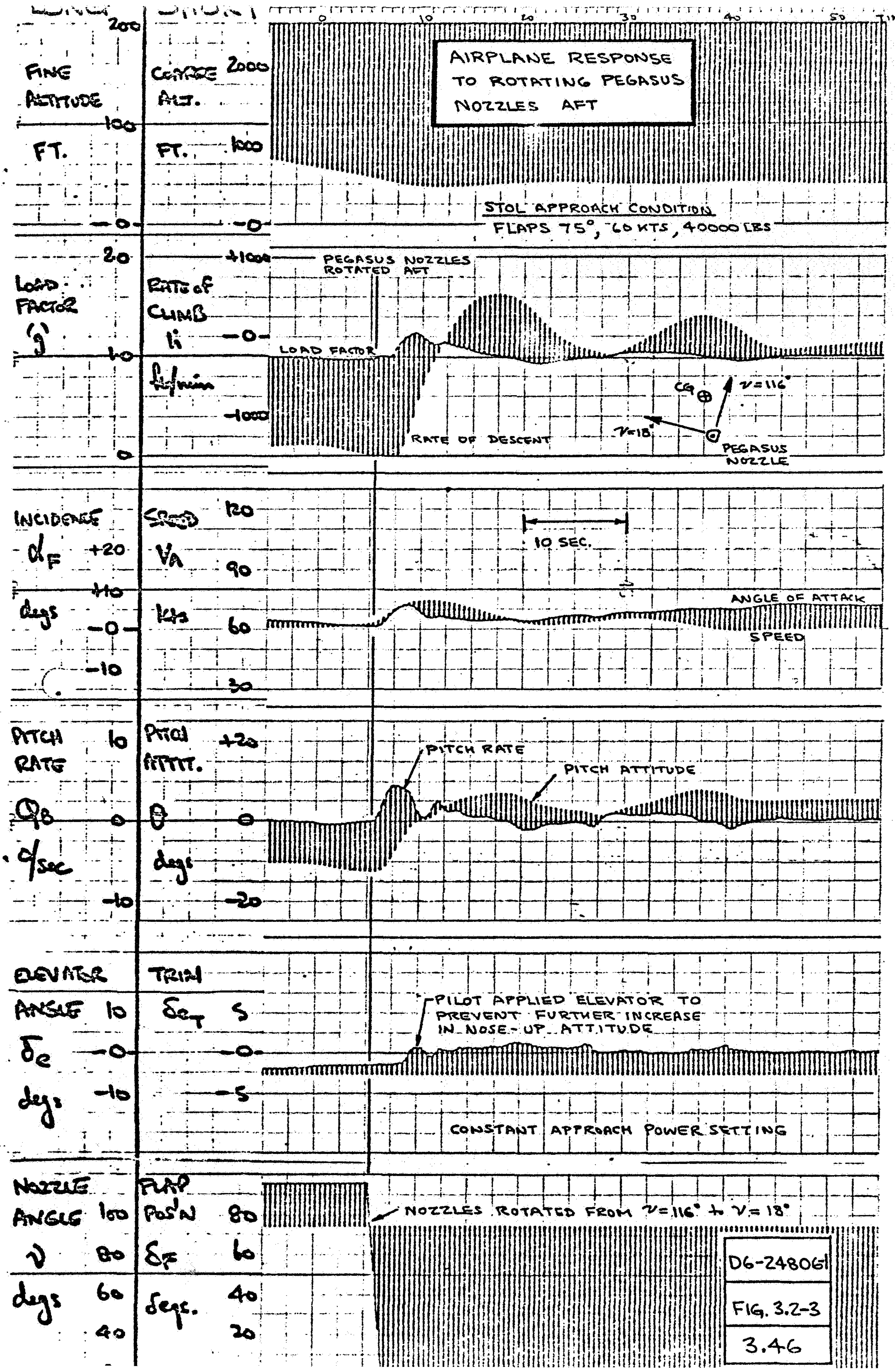
FIG 3.2-1



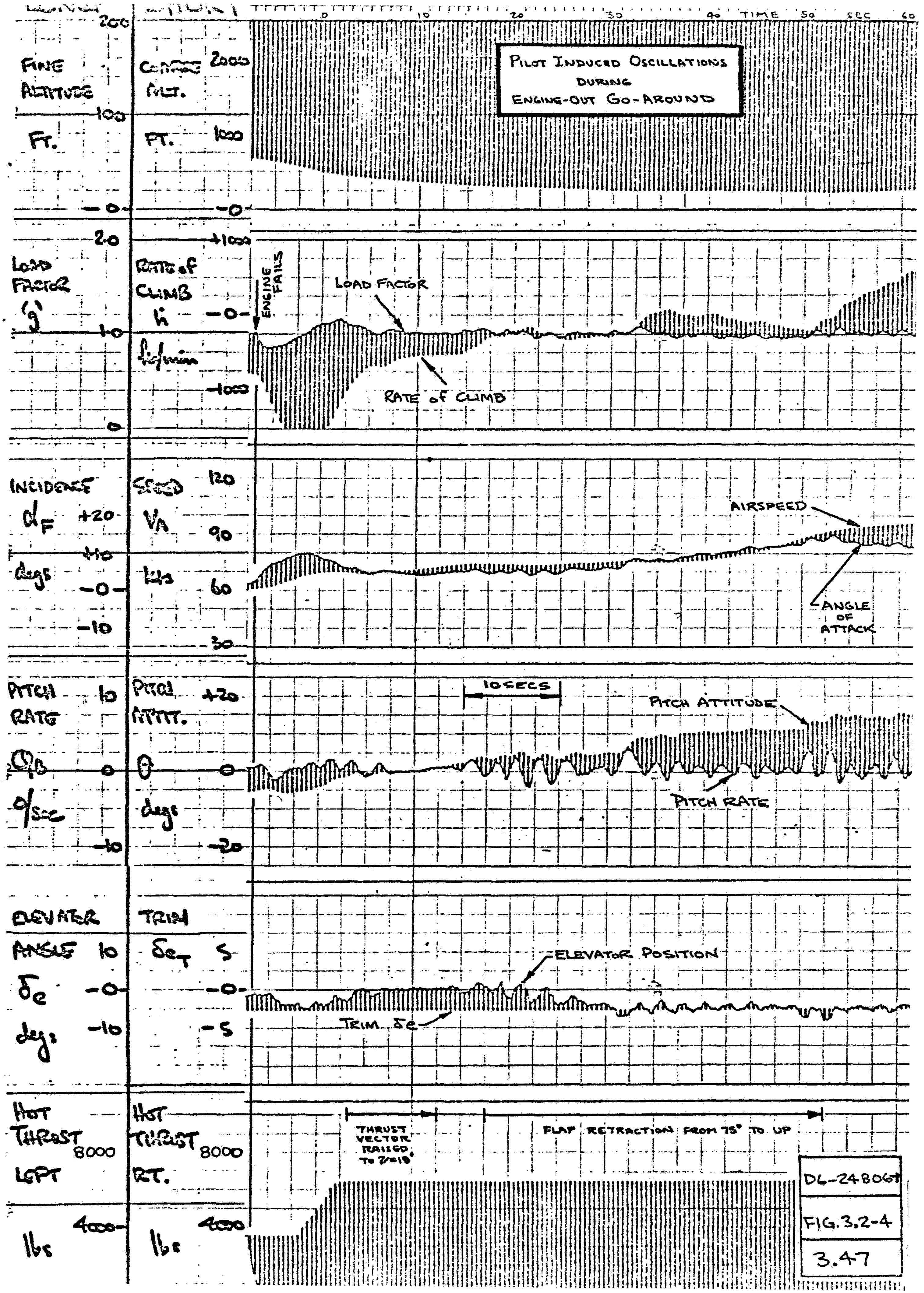
D6-24806I

FIG. 3.2-2

3.45



AIRPLANE RESPONSE TO ROTATING PEGASUS NOZZLES AFT
TIME ~ SEC
FINE ALTITUDE FT.
COARSE ALT. FT.
STOL APPROACH CONDITION
FLAPS 75°, 60 KTS, 40000 LBS
LOAD FACTOR 'g'
RATE of CLIMB ḣ ft/min
PEGASUS NOZZLES ROTATED AFT
LOAD FACTOR
RATE OF DESCENT
CG
ν=116°
ν=18°
PEGASUS NOZZLE
INCIDENCE α_F degs
SPEED V_A kts
10 SEC.
ANGLE OF ATTACK
SPEED
PITCH RATE Q_B °/sec
PITCH ATTIT. θ degs
PITCH RATE
PITCH ATTITUDE
ELEVATOR ANGLE δ_e degs
TRIM δe_T
PILOT APPLIED ELEVATOR TO PREVENT FURTHER INCREASE IN NOSE-UP ATTITUDE
CONSTANT APPROACH POWER SETTING
NOZZLE ANGLE ν degs
FLAP POS'N δ_F degs.
NOZZLES ROTATED FROM ν=116° to ν=18°
D6-24806I
FIG. 3.2-3
3.46



D6-24806T

FIG. 3.2-4

3.47

3.3 LATERAL-DIRECTIONAL STABILITY AUGMENTATION SYSTEM EVALUATION

3.3.1 Introduction

The lateral-directional stability augmentation system for the modified C-8A was evaluated during the week of November 3 through November 6, 1970. The primary purpose of the evaluation was to verify:

- The controllability of the airplane under VFR conditions at the "Approach" flight condition without stability augmentation.
- The control laws and gains derived in the studies at "Boeing".
- The proposed system authority limits.

The evaluation was dome primarily at the following conditions:

FLIGHT CONDITION

Approach - V_e = 60 Kts., F75°, $\gamma = -7.5^\circ$

- V_e = 50 Kts., F50°, $\gamma = -7.5^\circ$

Hold - V_e = 75 Kts., F30°, $\gamma = 0^\circ$

Takeoff - V_e = 75 Kts., F30°, $\gamma = 18.5^\circ$

Cruise - V_e =150 Kts., F4.5°, $\gamma = 0^\circ$

- V_e =160 Kts., F4.5°, $\gamma = 0^\circ$

AIRPLANE CONFIGURATION

Two sets of inertias - Nominal, and reduced roll and yaw inertias

Varying dihedral effect (C_{ℓ_β} = .0 and -.25/Rad)

ATMOSPHERIC CONDITIONS

Calm Air

"Moderate" Turbulence (4'/Sec RMS turbulence with random discrete gusts giving 8° bank angle change in 1 sec.)

Performance evaluation tasks included the following:

- Rapid turn entry to a given bank angle with roll-out to a specific heading.
- 200 foot sidestep maneuver initiated at 300 ft. altitude to a landing.
- Localizer tracking.

All tasks were done under VFR conditions only.

Following are the major results from the simulator evaluation.

3.3.2 Criticality of SAS

The airplane was found to be landable by all pilots at the 'approach' flight condition in moderate turbulence with no lateral-directional stability augmentation. Total loss of the augmentation system is therefore acceptable from a safety viewpoint. However, the pilots commented that they would not knowingly transition into the 'approach' flight condition without stability augmentation, but would land at a higher approach speed to take advantage of the improved handling qualities. (See Section 4.3)

3.3.3 SAS Control Law Evaluation

The simulator check-out and investigations prior to the SAS evaluation period started with the system shown in Figure 3.3-1. This was modified for the SAS evaluation to reflect the latest system configuration shown in Figure 3.3-2. This later system uses gain switching as a function of flap position (instead of a more complex airspeed gain control) to provide a more uniform response over the total flight envelope. A subsequent change in the trim condition from Flaps 50° to Flaps 30° for the 'takeoff' and 'hold' flight conditions therefore had little effect, with F50 gains being used at F30. Figure 3.3-3 shows the final system configuration arrived at during the simulator evaluation. These gains and control laws were found acceptable by all pilots.

The majority of the work was done at the approach flight condition, which exhibited the poorest handling qualities. Little difference was noted in the airplane response with C_{l_β} = .0 and C_{l_β} = -.25/RAD, SAS on; changes in airplane inertias were apparent however, with the nominal inertia configuration being down rated due to the reduced roll damping.

Roll control was the biggest single problem, with all three pilots commenting on the difficulty of precisely controlling roll attitude without continual wheel inputs. Increasing the roll rate feedback gain (δ_w/P) from -1.2 to -2.0 was considered a definite and desirable improvement by one pilot; a second pilot liked the increased damping but objected to the reduced wheel sensitivity and preferred the lower gain. Another pilot felt that roll control deteriorated with increasing bank angles, control being good at 10 degrees, fair at 20 degrees and moderately difficult at 30 degrees.

The only augmentation considered desirable at cruise flight conditions was a yaw rate damper. The $\hat{\beta}$ damper* was not as satisfactory, presumably due to the roll attitude feedback gain which was set for the 'approach' condition.

The $\hat{\beta}$ damper was evaluated at the 'approach' condition with and without the bandpass filter, with no differences noted.

3.3.4 SAS Authority Requirements

Authority requirements were evaluated primarily at the approach flight condition; with the pilots being asked to fly roll angle maneuvers that they considered reasonable.

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*Directional SAS using pseudo rate-of-change of sideslip feedback

Typical maneuver magnitudes and maximum SAS inputs with moderate turbulence are as follows:

MANEUVER	MAXIMUM SAS INPUTS			
	δ_W^{SAS}	$\dot{\delta}_W^{SAS}$	δ_R^{SAS}	$\dot{\delta}_R^{SAS}$
$\phi = 10^\circ \rightarrow 15^\circ$ $P_B \leq 8^\circ$/Sec	18°	35°/Sec	7°	14°/Sec
$\phi = 15^\circ \rightarrow 20^\circ$ $P_B \leq 12^\circ$/Sec	21°	20°/Sec	10°	10°/Sec
$\phi = 20^\circ \rightarrow 25^\circ$ $P_B \leq 14^\circ$/Sec	23°	20°/Sec	10°	20°/Sec

It should be noted that the SAS inputs, even though peak values, are less than the authorities used for the failure evaluation. Figure 3.3-4 shows a typical time-history of the airplane and SAS response in calm air at the 'approach' flight condition.

The effect of position limits on the SAS performance was investigated. In the directional axis, the minimum satisfactory authority was found to be $7.5^\circ\ \delta_R$. Authorities less than this resulted in degraded turn coordination. In the lateral axis, an authority limit of $15^\circ\ \delta_W$ appeared satisfactory. Authorities less than this resulted in an increased pilot workload. Although reduced position authority limits degrade performance and increase pilot workload, they do not cause control or instability problems beyond those of the free airplane.

Minimum rate limits for the SAS configuration shown in Figure 3.3-3 were not specifically determined, with the system rate limits considered satisfactory for the proposed SAS configuration. However, the 30 deg/sec aileron rate limit in the lateral axis was found to be too low for a control wheel steering system that was briefly investigated, causing P.I.O.'s with two evaluation pilots.

3.3.5 SAS Failure Transients

Flaps Down

Hardover and oscillatory failures were evaluated at the takeoff and approach flight conditions using the following authority limits:

Axis	Position Limit	Rate Limit
Lateral	$25^\circ\ \delta_w$	$30^\circ/\text{Sec}\ \dot{\delta}_w$
Directional	$12.5^\circ\ \delta_R$	$15^\circ/\text{Sec}\ \dot{\delta}_R$

After a failure in the approach condition the pilots were asked to fly to a landing; after a failure during takeoff, the pilots were asked to maintain a constant heading. No pilot delay prior to recovery action was used. Failures were inserted at altitudes ranging down to 60 ft.

Recovery from single axis hardover failures, either lateral or directional, was no problem at the approach flight condition, with a successful landing made in all cases. Comparable results were found with hardovers during takeoff. Oscillatory failures also posed no recovery problem. A simultaneous hardover in the lateral and directional axis during the approach was felt to be the most difficult failure, but still considered acceptable.

Because the SAS quick-disconnect switch was not available in the simulator, pilots did not switch out the failure until they had the initial transient under control. This switch will be situated on the control wheel in the actual airplane and would normally be used to disconnect the SAS in the event of a hardover. Typical maximum values following a hardover failure at the approach flight condition are as follows:

	ROLL ANGLE	SIDESLIP ANGLE	RUDDER PEDAL	CONTROL WHEEL
DIRECTIONAL HARDOVER	6°	15°	2.5"	-
LATERAL HARDOVER	10°	5°	.6	57°
DIRECTIONAL AND LATERAL HARDOVER	12°	12.5°	2.1"	65°

Flaps Up

Directional axis hardovers were evaluated at the maximum cruise flight condition (V_e = 160 Kts) by two pilots. The evaluation was started for the first pilot with the authority at 12.5 degrees rudder deflection. This gave an unacceptably large transient. Successive attempts at reducing the transient indicated a maximum permissible authority of 5 degrees rudder deflection. This authority, which was the starting point for the second pilot's evaluation, was rated unacceptable by him. His preference was to have no SAS flaps up rather than be open to such large transients. Further tests were not done since performance requirements at the 'approach' flight condition indicated the desirability of authorities greater than this. As a result of these tests no SAS is provided for the flaps up configuration.

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3.3.6 Control System Resolution

Lateral control system deadzone effects were evaluated at the 'approach' flight condition, with the deadzone inserted downstream of aileron power control unit. Both SAS and pilot inputs were therefore subjected to this non-linearity. Deadzones as low at $1.5^{\circ} \delta_w$ (2% max. pilot authority) were noticable to the pilot, causing an increased pilot workload due to the spiral divergence. Efforts are therefore being made to keep control system deadzones to less than one degree.

3.3.7 Control Wheel Steering (CWS)

A series control wheel steering mode was briefly evaluated as an alternative to the lateral augmentation system shown in Figure 3.3-3. The CWS system provided a roll attitude hold capability, as well as electrical feed forward for response quickening. Pilot commands are transmitted mechanically to the aileron PCU in the normal manner. In addition, a series servo actuator adds or subtracts to the mechanical signal proportional to the roll attitude error. This error signal is derived from the desired roll rate, which is proportional to control wheel deflection, and the actual roll rate and attitude.

A block diagram of the CWS system is shown in Figure 3.3-5. This configuration did not give a totally satisfactory pilot response, partially due to inadequate system check-out time. The evaluation was further compromised by low servo rate limits which tended to cause pilot induced oscillations. Despite these shortcomings, the advantages of such a system, particularly in turbulence, were apparent.

3.3.8 Automatic Speed Control

A simple speed control system was briefly evaluated, using vector control of the Pegasus nozzles to maintain the desired airspeed. Only airspeed hold, not airspeed capture capability was provided. The following control law was used:

$$\nu = K(V - V_{REF}) \text{ , where}$$

ν = nozzle angle,

V = instantaneous airspeed

V_{REF} = desired airspeed

K = system gain, deg/kt.

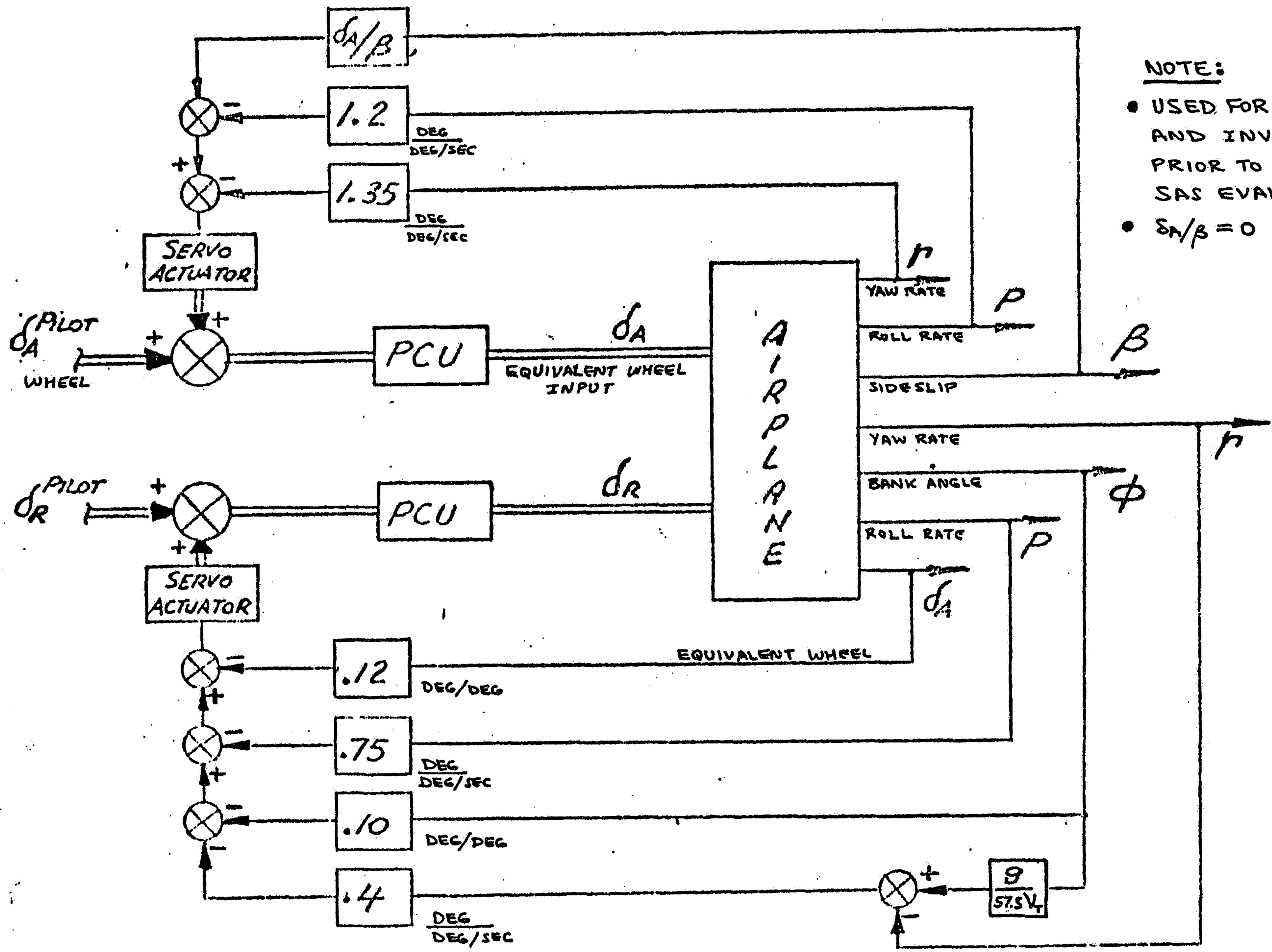
A gain K=15 deg/kt. was considered desirable, keeping the airspeed error within 2 kts. during normal 'approach' maneuvers, including 200 ft. sidesteps. Nozzle deflections varied between $\nu = 50^{\circ}$ and $\nu = 112^{\circ}$, with maximum nozzle rates (in calm air) of approximately 10 deg/sec. No adverse pitching moment or pilot coupling problems were noted with the system operating.

One deficiency of the system was noted during a simulated engine failure. Airplane response to an engine failure without speed control is such that the speed automatically increases toward the "go-around" speed. With the speed control system engaged, however, this speed increase commanded the nozzles fully forward to maintain the normal 'approach' airspeed. This nozzle movement increases the sink rate, and hence the altitude loss following an engine failure, considerably. This made the system unacceptable to the pilot, even though it did reduce the workload during normal operation.

3.3.9 Rudder Induced Rolling Moment

The conventional airplane response to rudder pedal deflection (negative rudder surface) is to roll in the same direction that the pedals are deflected, i.e., right pedal produces right roll. During the SAS evaluation, one pilot noted that the augmented airplane with $C_{\ell_\beta} = 0$ rolled left with right pedal deflection. Roll rate per degree of rudder was approximately 0.2 deg/sec/deg rudder. This reversed response was disturbing to the pilot. The free airplane was found to respond in the conventional manner to rudder inputs.

A subsequent linear analysis indicated that the free airplane responds in the conventional manner only for the first 8 seconds, and then reverses itself. Further analysis showed that to produce right roll with right rudder pedal, the augmented airplane must have a $C_{\ell_\beta} = -.075$/RAD. If the airplane has neutral dihedral effect and it is strongly felt that conventional roll response due to pedal deflection is required, then this can be obtained with a rudder to aileron interconnect, using a gain $\frac{\delta_W}{\delta_R} = -.66$. Such an interconnect is not being incorporated into the design at the present time.

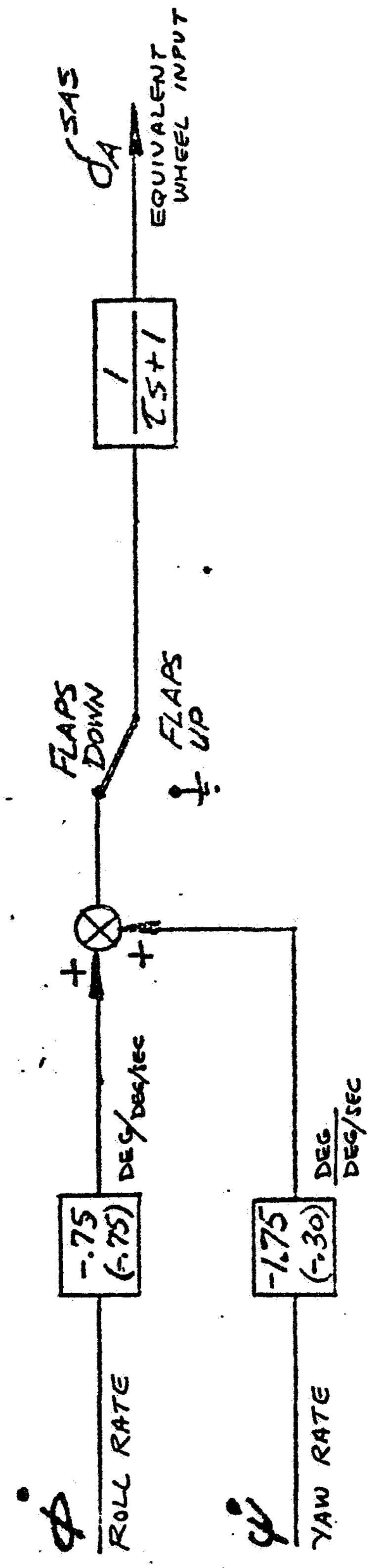


NOTE:

- USED FOR CHECK-OUT AND INVESTIGATIONS PRIOR TO FORMAL SAS EVALUATION
- $\delta_A/\beta = 0$

FIG. 3.3-1 - STABILITY AUGMENTATION SYSTEM (V_e = 60 KTS. C_{ℓ_β} = .0)

69

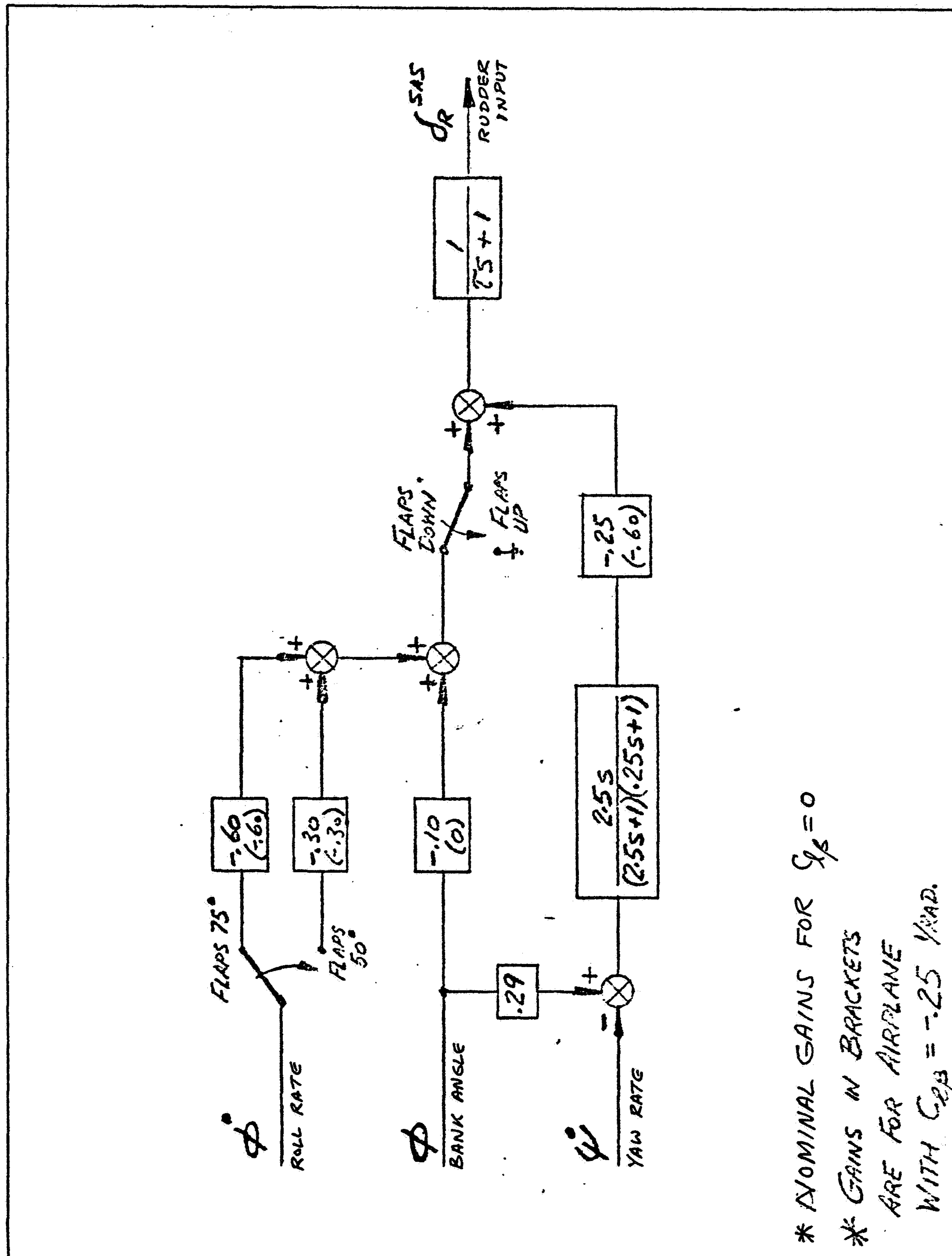


● NOMINAL GAINS FOR $C_{l_\beta} = 0$

● GAINS IN BRACKETS ARE FOR AIRPLANE WITH $C_{l_\beta} = -.25/RAD$

ENGR.	GLENDE	10/28/70	REVISED	DATE	LATERAL SAS USED FOR INITIAL EVALUATION OF GAINS AND CONTROL LAWS	D6-24806I
CHECK						FIG 3.3-2A
APR					THE BOEING COMPANY RENTON, WASHINGTON	3.58
APR						

70



ENGR.	GLENDE	10/28/70	REVISED	DATE	DIRECTIONAL SAS USED FOR INITIAL EVALUATION OF GAINS AND CONTROL LAWS	D6-24806I
CHECK						FIG 3.3-2B
APR					THE BOEING COMPANY RENTON, WASHINGTON	3.59
APR						

71

AD 1546 O

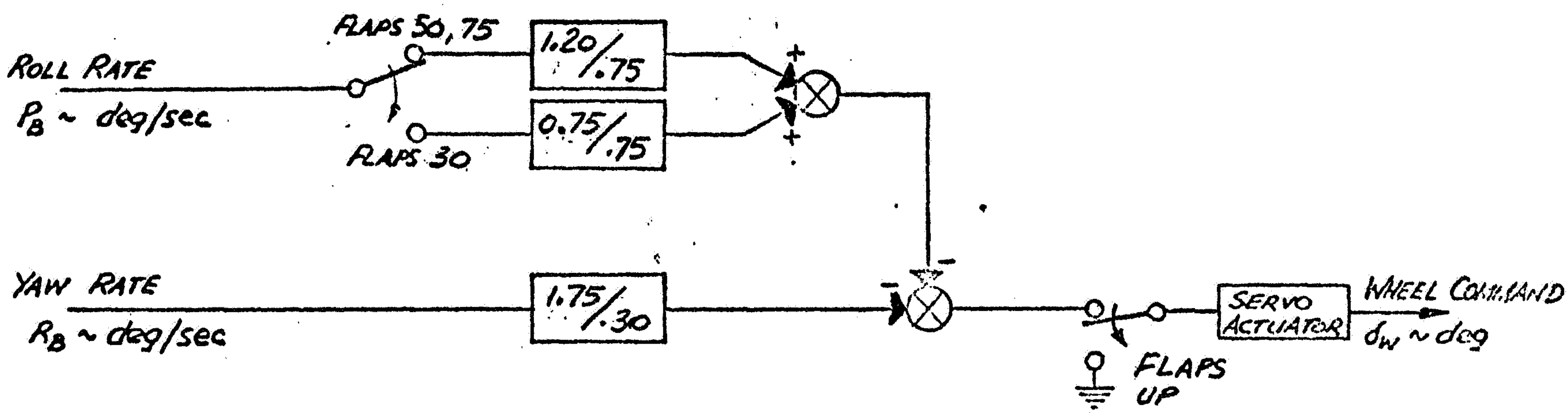


FIG 3.3-3A FINAL LATERAL AXIS STABILITY AUGMENTATION

NOTE: $\boxed{X/Y}$ 'X' IS GAIN FOR A/C WITH $C_{\ell_\beta} = 0$
'Y' IS GAIN FOR A/C WITH $C_{\ell_\beta} = -.25$/RAD

72

AD 1546 D

NOTE: X/Y 'X' IS GAIN FOR A/C WITH $C_{\ell_\beta} = .0$
'Y' IS GAIN FOR A/C WITH $C_{\ell_\beta} = -.25/RAD$

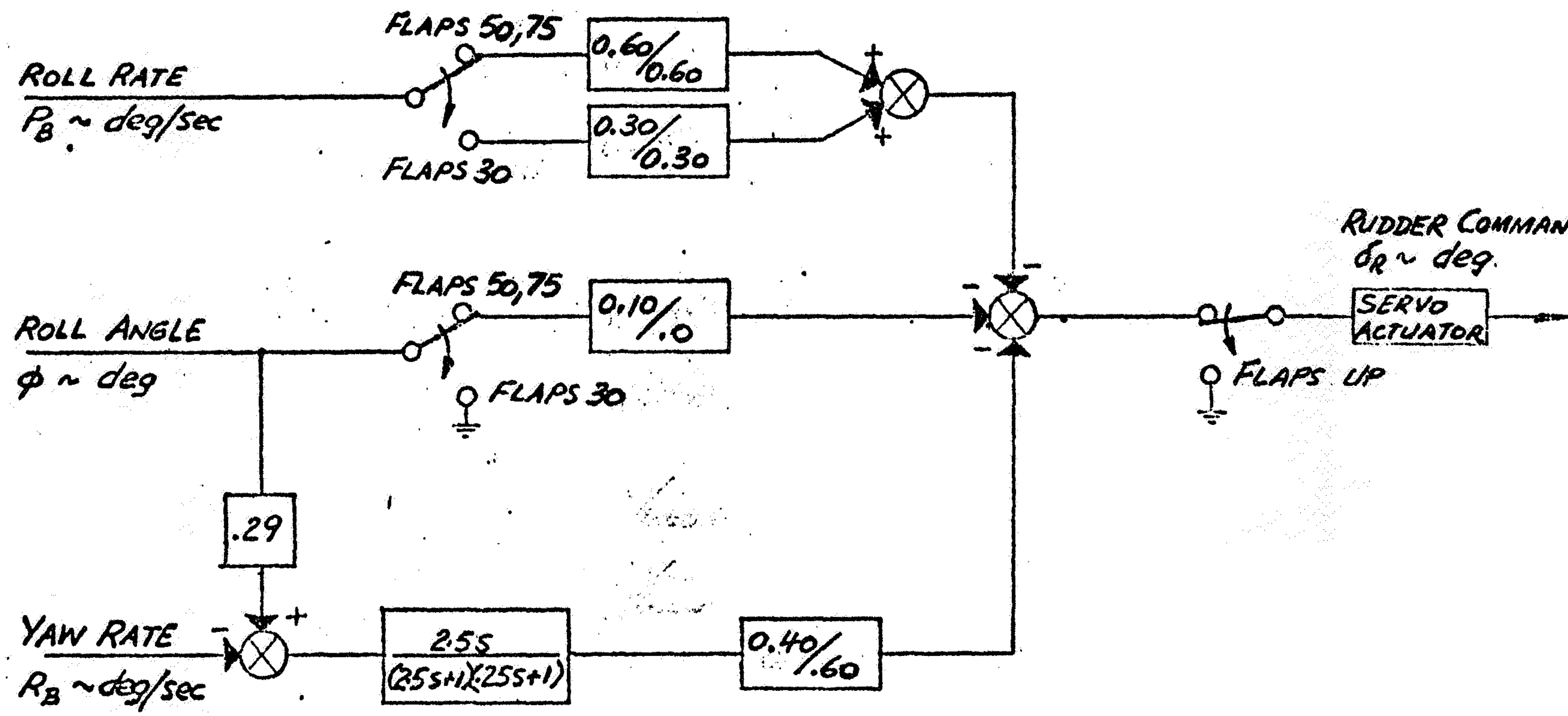
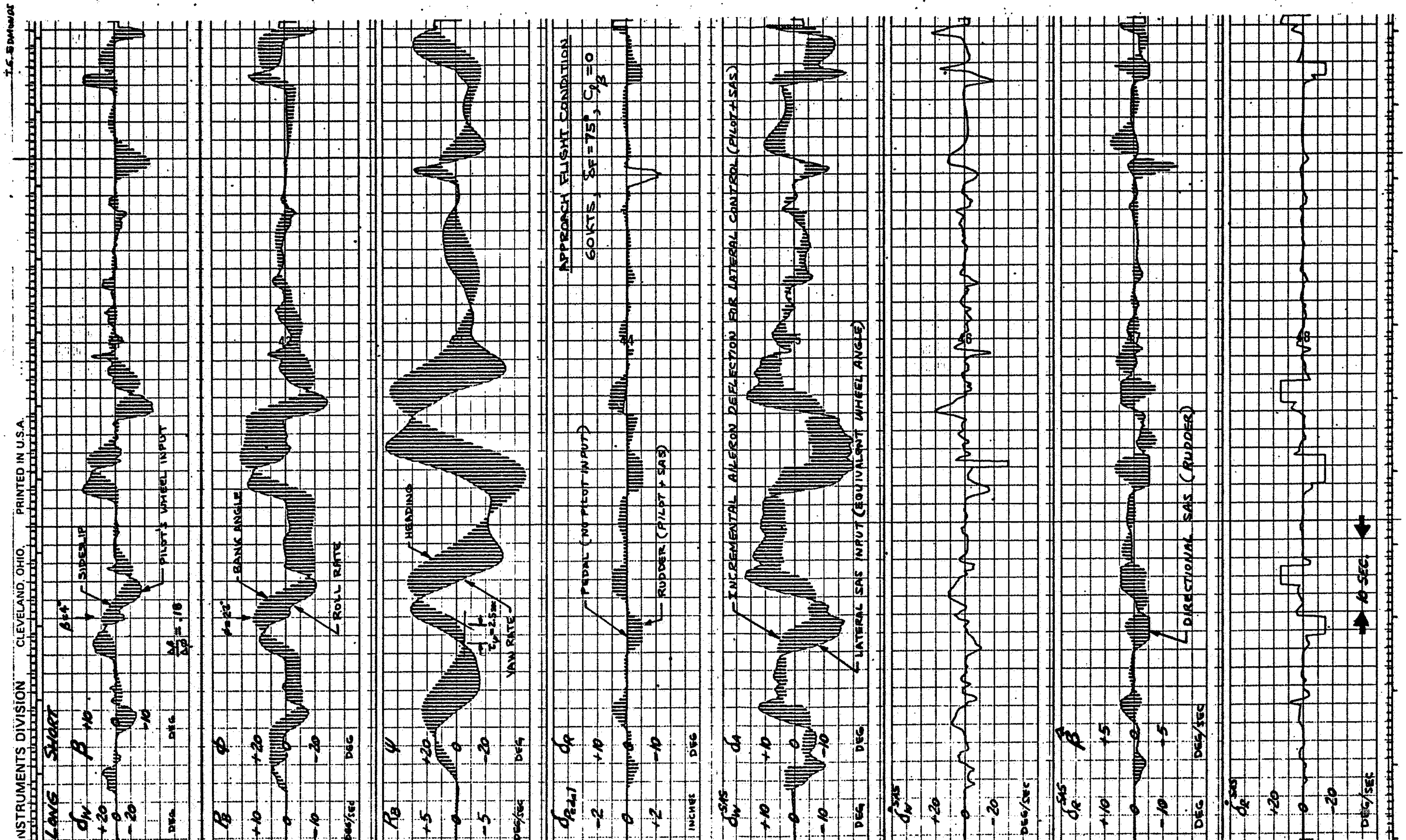


FIG 3.3-3B FINAL DIRECTIONAL AXIS STABILITY AUGMENTATION



TYPICAL RESPONSE TO
RAPID HEADING CHANGES
WITH LAT.-DIR. SAS

D6-24806-1

FIG 3.3-4

3.62

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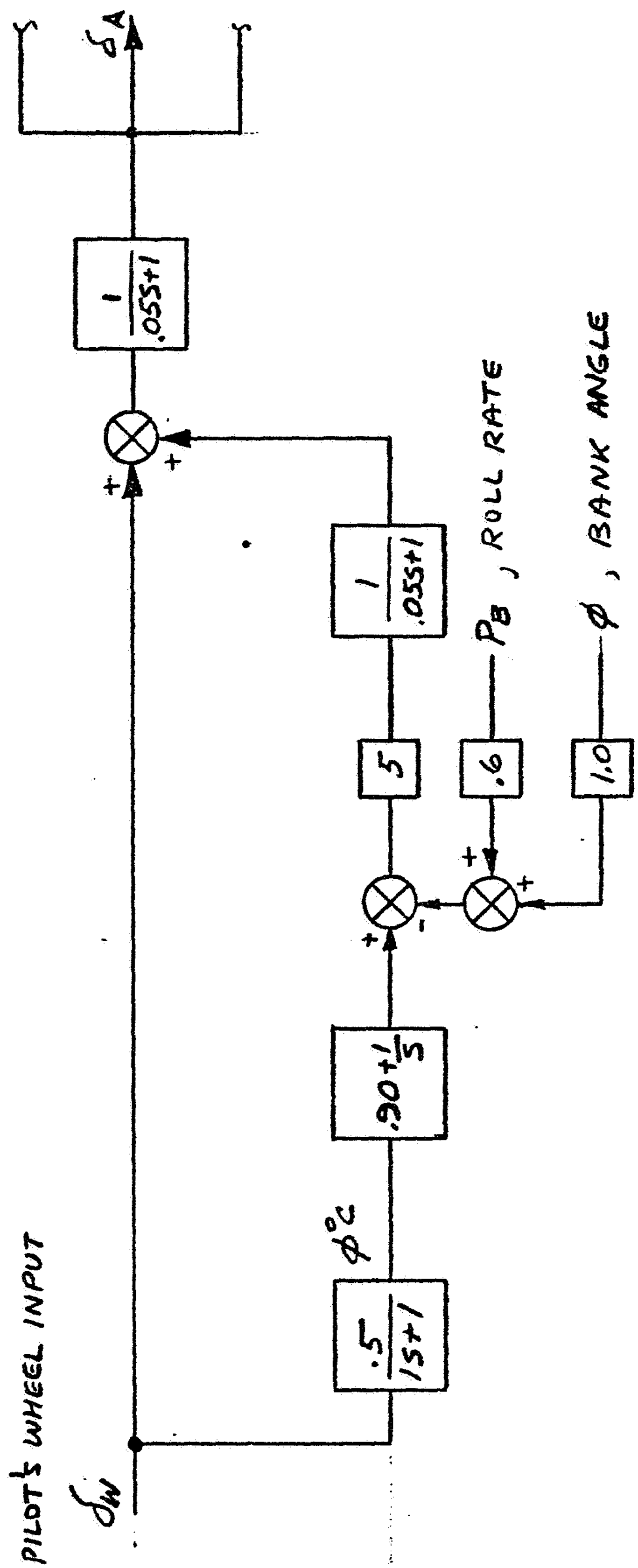

ENGR.	GLENDE	10/28/70	REVISED	DATE	SIMULATED CONTROL WHEEL STEERING SYSTEM	D6-248061
CHECK						FIG 3.3-5
APR						
APR					THE BOEING COMPANY RENTON, WASHINGTON	3.63

AD 1017 R6

75

3.4 ENGINE AND PEGASUS NOZZLE CONTROL SYSTEMS DESIGN

Predicted engine and nozzle characteristics were investigated from the point of view of control, airplane handling qualities, and to gain an insight into operational procedures for the augmentor-wing airplane. The following conclusions are drawn from the comments of the three pilots who flew simulated engine failures, go-arounds, normal landings, landings in turbulence, transitions from flaps up to landing configuration, and landings with various flight control system failures.

3.4.1 Engine Acceleration and Deceleration Characteristics

Figures 3.4-1 and 3.4-2 show the acceleration and deceleration characteristics for the engine used in the simulator. These charts are plotted in terms of total thrust (hot and cold), so that the non-linear relationship between cold thrust and total thrust will actually give different characteristics for the hot gas thrust and the augmented cold flow blowing on the wing. The time lag between a change in engine speed and the change in pressure and mass flow from the flap nozzles was considered to be small enough to be neglected in the simulation. With these characteristics, full power was available two seconds after selection from approach thrust setting to full throttle. This was considered to be adequate for recovery from engine failures, for baulked approaches and for landing flares using power. No adverse comments were received on engine deceleration times even though these had been deliberately slowed down to provide engine protection against back pressure from the flap blowing ducts after rapid throttle retardation.

3.4.2 Engine Surge-Bleed-Valve Operation

The step changes in thrust that occur when the surge-bleed valves open and close were simulated as shown on Figure 3.4-3. The thrust changes were deliberately exaggerated in an attempt to ensure that they were noticeable to the pilot. The pilot could not detect the thrust hysteresis since it was always masked by the thrust change due to throttle motion. The throttle position at which the valve operated was then changed to coincide with the trim position for the flight condition being simulated, Flaps 75°, 60 knot approach speed. Even then the pilot did not detect the thrust change since he never settled the throttle at exactly the position where the bleed valve would operate. Assuming that the surge-bleed valve cannot be operated by changing engine conditions induced by airplane maneuvers, there should be no noticeable accelerations due to surge bleed valve operation.

3.4.3 Pegasus Nozzle Rate and Deadspace in the Nozzle Control System

Time histories of the nozzle response to a step control lever input on the Hawker Siddeley Harrier were obtained from NASA Langley. This data showed a maximum nozzle rate of about 75 deg/sec; however taking into account the small tail on the response, an average value of nozzle rate was 60 deg/sec. To ensure that these average rates were not too low, these were the values used at all times in the simulation.

Control of rate of descent by vectoring the hot thrust was the standard mode of operation of all three pilots. However their technique differed widely between the limits of one or two large changes in vector angle per approach to a technique of almost constant smaller motion of the nozzle levers. At no time was there any comment on lack of response or of poor sensitivity, even though the pilot often moved the lever fast enough to exceed the maximum nozzle rate. In typical go-around situations the nozzles were raised from the nominal approach position to full up in about 1.5 to 3.0 seconds.

A deadspace of $\pm 3^{\circ}$ nozzle angle was added between the pilot's lever and the exahaust nozzle and evaluated during approaches at flaps 75°, 60 knots. Due to the general technique used by the pilots (fairly large open-loop movements of the nozzles) no adverse effects were felt from this deadspace.

3.4.4 Pegasus Nozzle Angular Travel Limits

The static performance available from the nozzle angular range of travel from 18° to 116° is shown on Figure 3.4-4 for the approach configuration. During the simulation period, full downward and forward vectoring of the hot thrust ($\nu = 116^{\circ}$) was rarely used except for short periods during the transition from flaps up cruise to landing. In the flaps up condition full forward vectoring produced an adequate deceleration of 2 to 3 knots per second in level flight. In the approach configuration full forward vectoring produces a rate of descent of 1400 ft/min at a constant 60 knots. This descent capability was more than sufficient to regain the glide path even from some transitions which were started very close to the runway.

The nozzle vector angle also affects controllability during an engine failure. As the nozzle vector angle is changed in order to go-around after an engine failure, the rolling and yawing moments change as described in Section 3.1.7. The rolling moment changes sign and the yawing moment builds up to high values, requiring the pilot to reverse the wheel to trim and to apply rudder to balance the airplane. Prior to the simulator period it was felt that controllability might be improved by restricting the minimum nozzle angle to 40° instead of 18°, thus preventing reversed rolling moments and reducing the peak yawing moments. A short investigation of these effects in the simulator revealed that the effects on controllability were very small and of little

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consequence compared to reduced go-around performance at the new minimum nozzle angle. Also the out-of-balance rolling moments at flap angle of 30° presents a different picture. Restricting the nozzle angle to 40° here would actually increase the rolling moment that needs to be balanced during the go-around.

The available travel of 18° to 116° was therefore considered adequate.

3.4.5 <u>Nozzle Lever Handle Design</u>

The layout of the throttle levers and the vector controls used in the simulator cab had been styled fairly closely after the existing airplane throttle and propeller pitch controls with certain modifications to the latter to serve as nozzle vector controls. Prior to the simulation period it had been felt that the vector controls should be longer than the throttle levers (which are closer to the pilot) in order to ensure that the pilot could easily reach the nozzle levers around the throttles, see Figure 3.4-5. Also, it was felt that there might be a sensitivity problem due to the restricted travel of the nozzle levers (38° overall) which control the Pegasus nozzles through an angle of 98°. The nozzle controls in the simulator were therefore provided with a larger travel than that available in the airplane, but for the initial evaluation this travel was blocked off at 38°.

The initial 3 1/2 to 4 1/2 hours of flying by each pilot was devoted to familiarization. The program included normal two-engine approaches, in calm and gust conditions; engine failure followed by landings or go-arounds; complete transitions from flaps up configuration to landing; approaches without the stability augmentation system and approaches at various flap settings and speeds.

At the conclusion of the familiarization period, and prior to beginning formal testing and evaluation of airplane design features, each pilot was asked to comment on the suitability of the nozzle lever/throttle lever layout. All three pilots indicated that the existing geometry would be adequate for the rest of the evaluations but reserved the right to make a final choice for the airplane following the planned evaluation at various nozzle lever geometries in the final phase of the simulation period.

During the later evaluation of lateral control system design and engine-out control it became obvious that at least one of the pilots was operating the nozzle controls by holding the lever well above the knobs provided. Questioning the pilots revealed a universal opinion that the existing levers were too long.

On the final day of the simulation, flying began with a nozzle lever of the same length but with a stirrup type or 'D' handle. The pilots immediately felt that these would be no improvement over the other long levers and so they were replaced with a set of levers of the same length as the existing propeller pitch controls in the Buffalo (about 1.2" shorter than the throttle levers), see Figure 3.4-6.

Pilot comments on these levers were that they were easier to use, and enabled easy transfer of the hand from throttles to vector levers without diverting the eyes from the airplane instruments or the outside field of view. The only comment received on lever travel was that the 38° available was probably too long and that no sensitivity problems existed.

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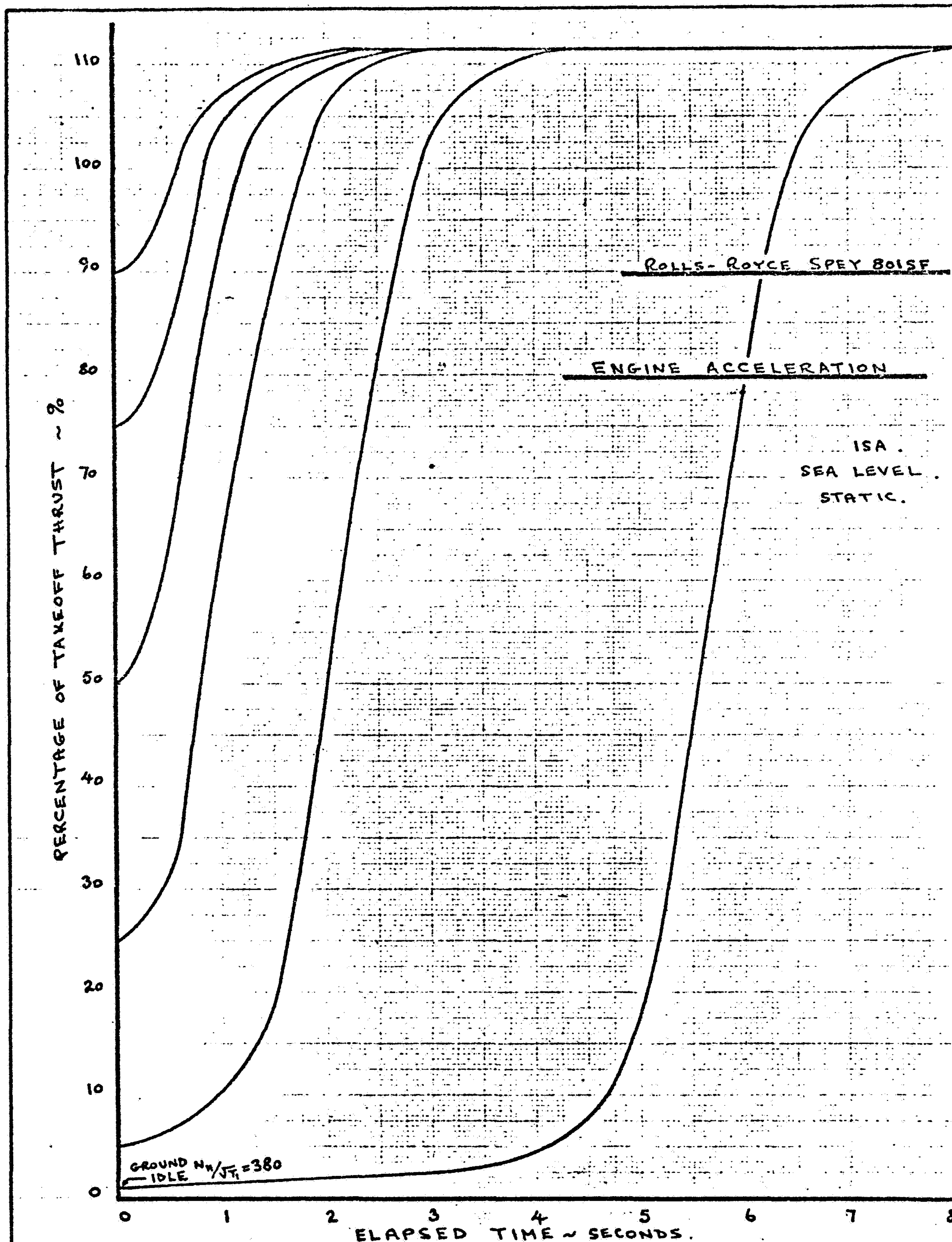

CALC	R.M.T.	SEPT. 4/70	REVISED	DATE	ROLLS-ROYCE SPEY 801SF. ENGINE ACCELERATION.	FIG 3.4-1
CHECK						D6-24806-1
APR						
APR					THE BOEING COMPANY	PAGE 3.69

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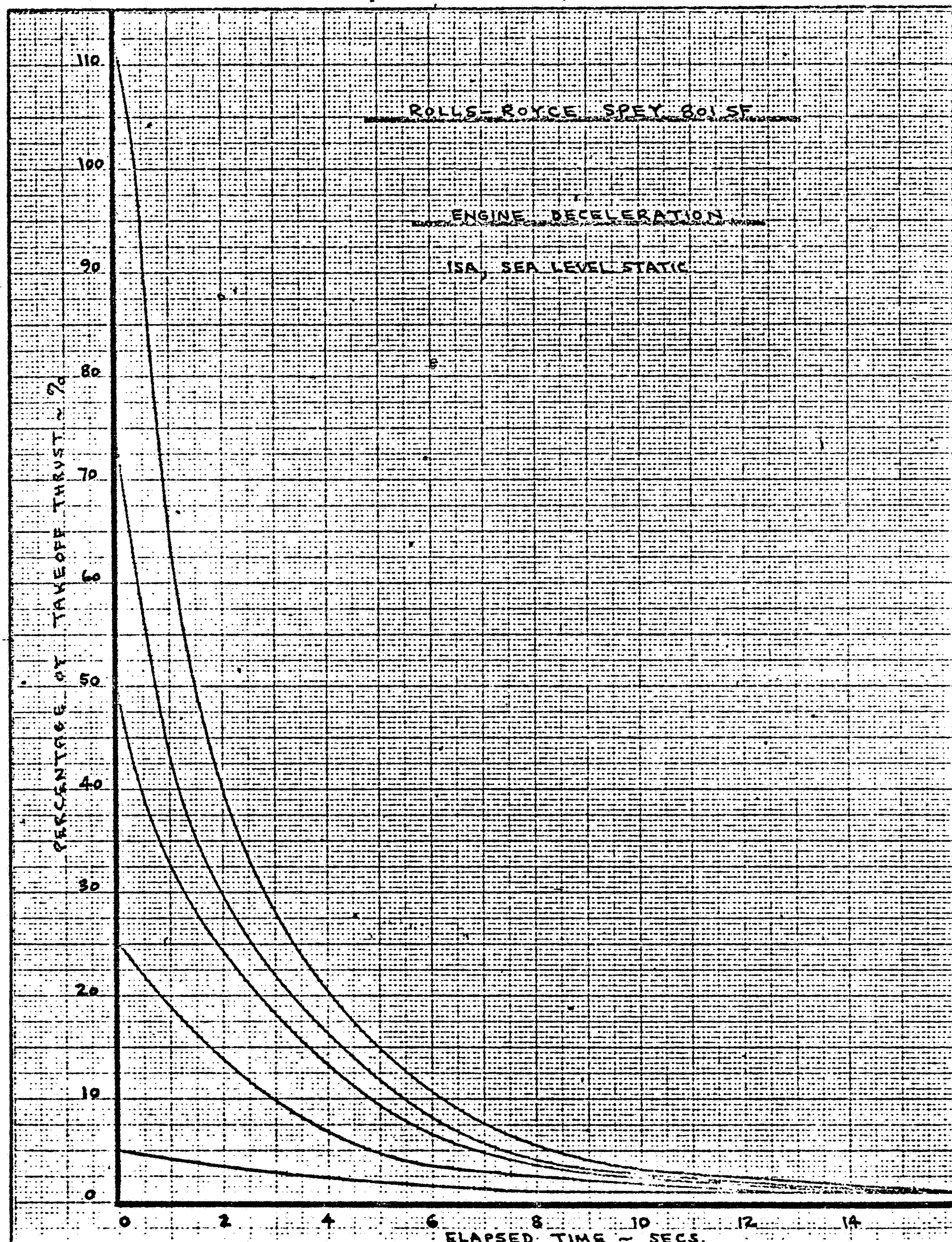
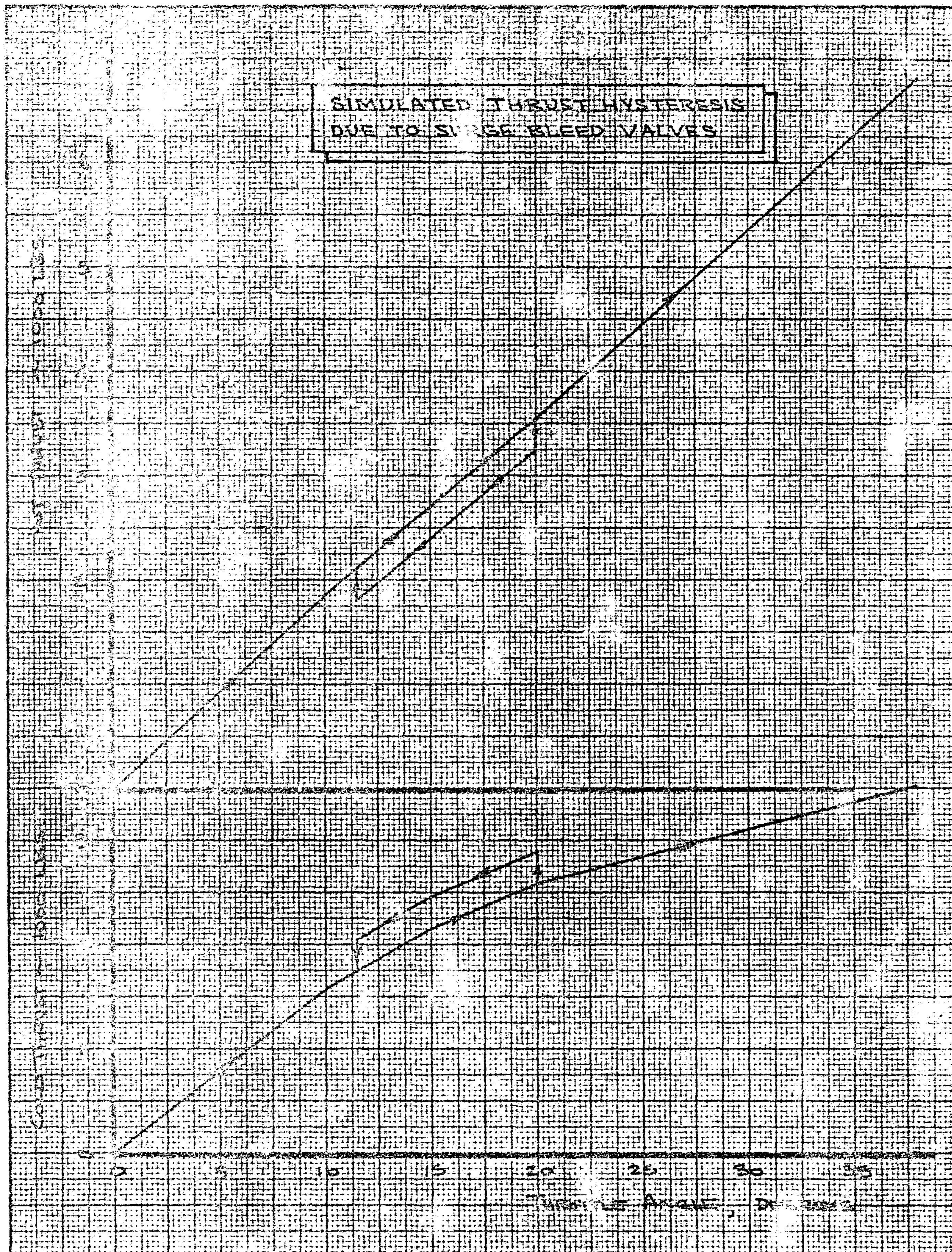
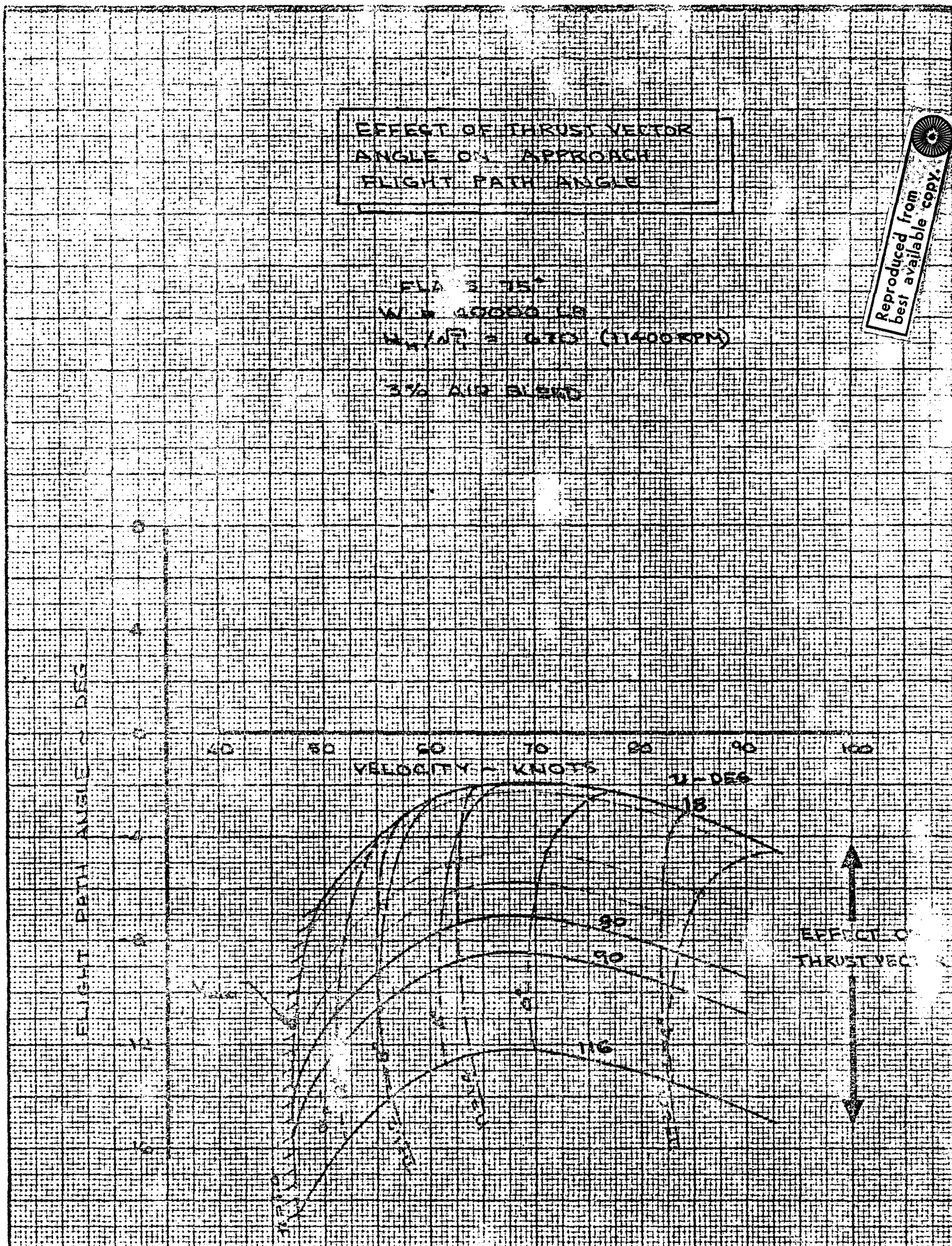


CALC	R.M.T.	OCT. '70	REVISED	DATE	ROLLS-ROYCE SPEY 801 SF	FIG 3.4-2
CHECK					ENGINE DECELERATION.	D6-24806
APR						
APR					THE BOEING COMPANY	PAGE 3.70



CALC	RUMSEY	12/1/70	REVISED	DATE	THRUST HYSTERESIS DUE TO SURGE BLEED VALVES	FIG 3.4-3
CHECK						
APR						D6-24806-1
APR					THE BOEING COMPANY	PAGE 3.71



CALC	K. ELBEL	11/5/70	REVISED	DATE	APPROACH FLIGHT PATH PERFORMANCE	FIG 3.4.4
CHECK	V. PAGE	11-11-70				D6-24806-1
APR						
APR					THE BOEING COMPANY	PAGE 3.72

SIMULATOR COCKPIT WITH LONG NOZZLE CONTROL HANDLES

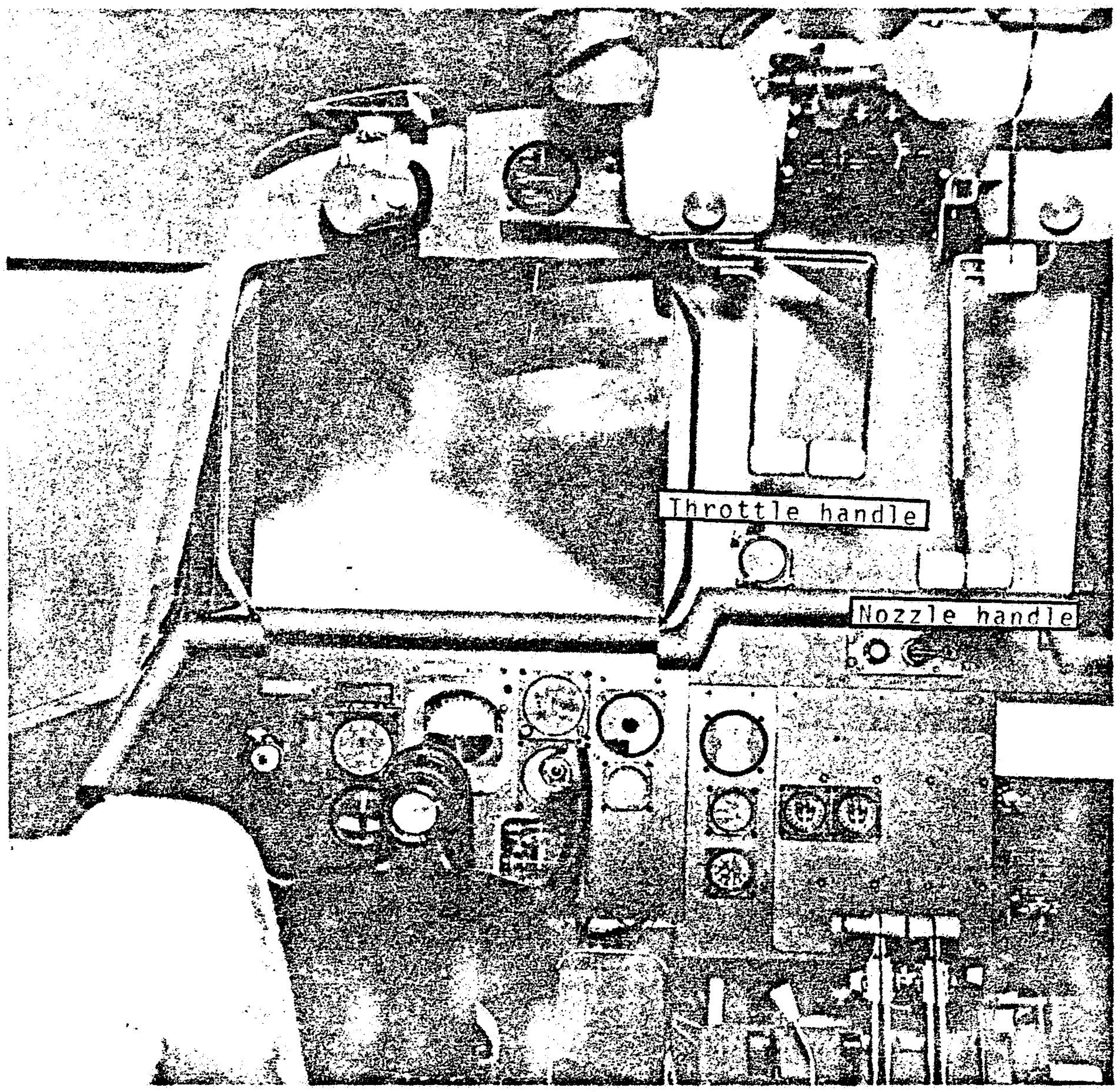


FIG 3.4-5

SIMULATOR COCKPIT WITH SHORT NOZZLE CONTROL HANDLES

A70-5060

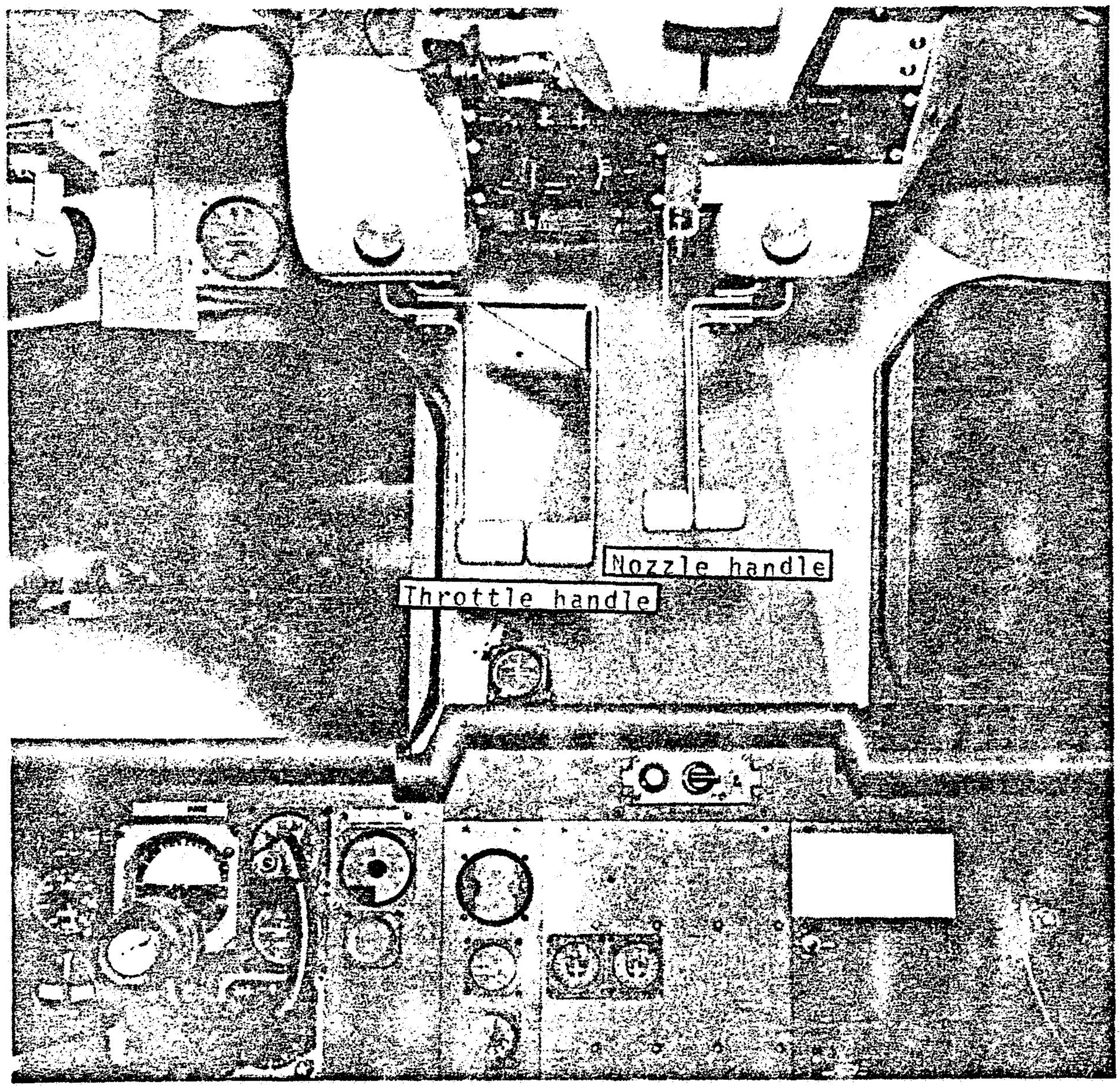


FIG 3.4-6

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
AMES RESEARCH CENTER MOFFETT FIELD CALIFORNIA

3.5 EVALUATION OF STRUCTURAL DESIGN CRITERIA

In dealing with the structural design requirements of the modified Buffalo, which involves an entirely new lift and thrust producing concept, a number of questions arise due to the lack of background knowledge of the vehicle flying characteristics. Past experience with more conventional airplanes is not necessarily applicable to the new design and an overall lack of feel for the airplane at its operating extremes hampers the definition of likely overspeed conditions or recovery load factors, etc. The piloted simulator was used to gain the necessary experience, and a specific investigation was conducted with this in mind.

Two pilots participated in this exercise both in the conduct and the planning of the tests in order to draw upon their extensive backgrounds in certification testing requirements. The testing was split into five parts, each test being aimed at answering questions that had been posed by the Structural Dynamics Design Staff.

3.5.1 Airplane Characteristics at the Placard Speeds

QUESTION: Is the behavior of the airplane conventional at speeds near the placards?

How likely are overspeeds and upset conditions?

The preliminary placard speeds used during this evaluation were 90 knots at flaps 75°, 95 knots at flaps 50°, 105 knots at flaps 30°, and a V_{MO} of 160 knots flaps up. Airplane handling qualities were evaluated at each flap placard speed by investigating turn entries and exits, sprial stability, directional stability, stick-force-per-g and static longitudinal stability. Each placard could be reached by a number of combinations of thrust setting,

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nozzle angle and wing incidence and in the course of the tests these placards were reached in dives, in level flight and in full power climbs. Despite the large negative angles of incidence which were produced at the placard speeds for flap positions of 75 and 50 degrees, no unusual airplane characteristics were found. However, the negative angle of attack at which the leading edge device will stall is not accurately known and these effects were not simulated. It should be noted here that the evaluation at flaps 30° was conducted with the SAS gains set for the flaps 75° configuration.

The only noteworthy event in the handling qualities evaluation was the powerful effect of the Pegasus nozzle control as a producer of instantaneous thrust or drag. In positioning the airplane near the placards with a high power setting and the nozzles vectored to 55° it was relatively easy to accelerate rapidly through the placards by merely rotating the nozzles to the 18° position. Recovery was equally prompt by vectoring to 90° or more.

A complete approach and landing was completed at 85 knots at flaps 75 with no tendency to overspeed past the 90 knot placard and no unusual flying characteristics were noted. A full stall at approach power was conducted with a diving recovery with power on. Again, there was no tendency to exceed the placard speed during recovery and no unusual attitudes were produced.

To further test the adequacy of the placard speeds an overshoot and climb-out condition was set up from a trimmed 60 knot approach at 11,200 rpm (a setting a little lower than the nominal approach flight condition).

At this same engine setting, the Pegasus nozzles were vectored aft and the flaps raised to 50° (at a fixed rate of 6 deg/sec). At a trim speed of 84 knots the airplane was still in a shallow dive at 100 ft/min. The flaps were raised to 32^{o} and the airplane settled at 102 knots in approximately level flight. The flaps were raised to the up position and the speed reached a maximum of 130 - 140 knots. The pilot felt that this exercise demonstrated the adequacy of the preliminary placards at flaps 75, 50 and up, but that there was far too little margin at flaps 30. A repeat of this condition selecting flaps 30^{o} immediately after vectoring the thrust aft showed that 105 knots could easily be exceeded even when the power was pulled back.

Data taken earlier in the simulator investigation (during pilot familiarization with transition techniques) had shown a similar story. Only one of the pilots was briefed on the suggested placard speeds prior to this investigation. This pilot kept 4 out of his 5 runs inside the suggested placards. The one exception was within the placard at 50^{o} flap but passed through the 115 knots at flaps 30. Figure 3.5-1 shows the speeds at flap angles of 30^{o} and 50^{o} used by all three pilots, points from the same transition being conveniently connected by a straight line. The figure shows that the slopes of these lines (for a 2 deg/sec flap retraction rate) are for the most part parallel, and that using this general slope a flap placard speed of 120 knots at 30^{o} flaps would be consistent with 95 knots at 50^{o} flap. If the placard at 30^{o} flaps were raised to 120 knots all but three of the 16 transition cases would be included inside the revised placards.

The effect of an increased flap rate on this picture is not immediately clear although we might expect that an increased flap retraction rate would flatten the slope of the lines. However, bearing in mind the pilot comments from the other tests analyzed, the following recommendations for operational flap placard speeds have been made:

Flap Angle	V_{max} Op.
UP (4.5°)	160 Knots
30°	120 Knots
50°	95 Knots
70°	90 Knots

3.5.2 Overspeeds and Upsets

QUESTION: What would be a typical overspeed value and what are the likely load factors used in recovery from overspeed conditions?

To answer this question the pilots were asked to trim in a 30 degree bank turn in a full power climb at the placard speed for each flap angle. The controls were then released for five seconds, and a prompt recovery to speeds inside the placard was then accomplished. In agreement with our initial findings of good longitudinal static stability and acceptable lateral-directional dynamic stability at the placard speeds, the overspeed values were quite small, except at the flaps up condition where the spiral mode was unstable. The results of these tests were:

FLAP ANGLE	TRIM SPEED KNOTS	FINAL SPEED KNOTS	PEAK RECOVERY LOAD FACTOR
75	88	93	Condition Terminated
	90	94	1.16
50	94	97	1.16
30	104	114	1.40
	105	113	1.28
	103	114	1.32
4.5	166	175*	1.44

---

* In this case there seemed to be an offset in the pilot's ASI. The 166 knots would appear to be 160 knots indicated. This case should be interpreted as nine knots overspeed condition.

During the previous evaluation of airplane flying qualities at the flap placards, inadvertent overspeed conditions occured during the evaluation in turning flight, as follows:

Flap Angle	Maximum Speed
75	94
50	102
30	110

In the dive to the flap 75 placard, a pitch angle of 12° nose down was needed to reach 90 knots. A deliberate overspeed to 99 knots was achieved by increasing the pitch dive angle to 16°. This is considered to be a sufficient deterrent to exceeding the landing configuration placard speed in diving flight.

3.5.3 Step Gusts at Minimum Operation Speeds

QUESTION: What effect do step gusts have when flying near the minimum operational speeds?

How would the pilot recover from such gusts?

Likely minimum operating speeds were set at :

50 Knots	Flaps 75
50 Knots	Flaps 50
60 Knots	Flaps 30
90 Knots	Flaps Up

Again, these speeds may be reached by a number of different combinations of thrust and nozzle angle. One pilot set the conditions for an approach to land at these speeds using enough power to give reasonable incidence values.

The second pilot used approximately the standard approach power setting and simulated an inadvertent slow down to the minimum speeds.

The only gusts used were 15 knot step tail winds. In all cases the airplane response to the step gusts was so fast that the pilots could not adequately compensate. The natural tendency was to overcontrol the airplane in pitch producing a forced oscillation with attendant load factor excursions. However, the peak excursions were still quite small.

Flap Angle	α_{Trim}	Maximum Values in Recovery α	Maximum Values in Recovery n_z
75	0°	15°	1.24
50	3	11	1.25
30	9	12	1.04
UP	11	18	1.00
75	12	22*	1.20
50	18	30*	1.32

3.5.4 Evasive Maneuvers

QUESTION: What are the maximum tail lift coefficients developed during evasive maneuvers?

These tests were to be conducted flaps up at V_{MO}, flaps 50 at the placard, and flaps 75 at approach speeds. Due to lack of time the only condition completed was the landing configuration at 60 knots.

----

* Conditions trimmed at low power stalled in the gust.

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The pilot was asked to perform a sudden evasive maneuver during an approach assuming that he had suddenly sighted an airplane crossing his flight path. The evasive maneuver was a wings level pull up to maximum elevator angle. A peak load factor of 1.28 was reached at an α_F of 17°. The speed fell to 46 knots before recovery. The maximum tail lift coefficient recorded was $C_{L_{tail}} = -.27$ based on wing reference area, or $-.725$ based on horizontal tail area.

3.5.5 Nose-Gear-First Touchdowns

QUESTION: How likely is it that the airplane will touchdown nose-gear first?

The most likely conditions expected to produce nose-gear touchdowns were light weight approaches at speeds above the approach speed. A 35,000 lbs. condition at 60 knots was evaluated in which the airplane was trimmed at $\alpha_F = -1.75$ degs on the 7.5 degree glide slope. Increasing speed to 65 knots during the approach resulted in a nose down pitch angle of $\Theta = 9.0^{\circ}$ and the subsequent touchdown occured at $\Theta = -3.2^{\circ}$. For this condition further rotation to clear the nose gear caused the airplane to climb away again instead of landing.

Previous landing conditions had shown that large nose down pitch angles would be induced by flying with high power levels on the approach. Again in these conditions a flare for touchdown resulted in the nose-gear touching first.

It was therefore concluded that nose-gear first touchdowns were quite likely to occur and should be considered in the structural design.

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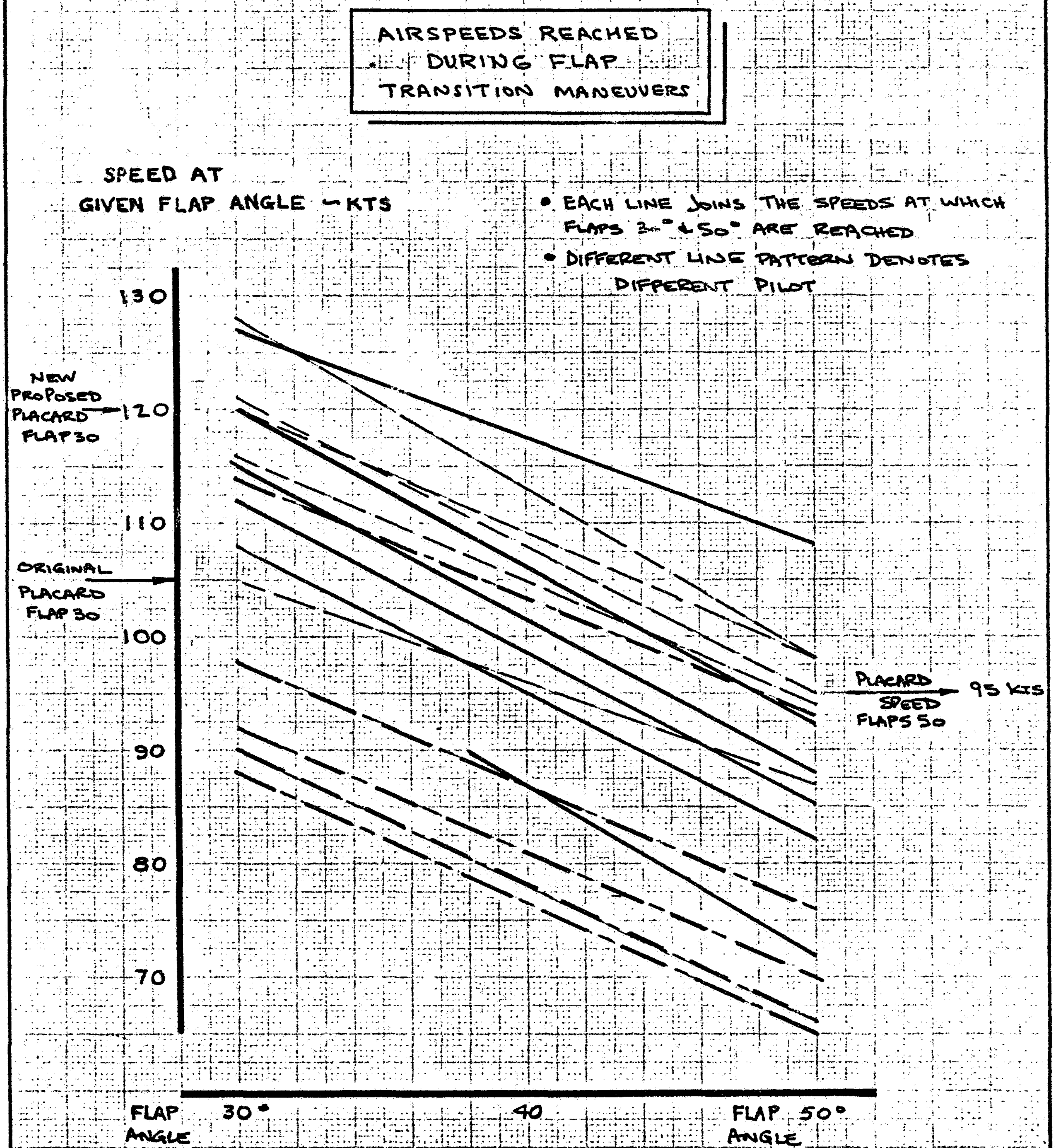

CALC	Rumsey	12/18/70	REVISED	DATE	APPROXIMATE SPEED ~ FLAP ANGLE RELATIONSHIPS DERIVED FROM PILOTED TRANSITIONS.	D6-24806-1
CHECK						FIG. 3.5-1
APR					THE BOEING COMPANY	PAGE 3.82
APR						

AD 461 C-R6 K•E ALBANENE 195L 8.5 G-8000 TRACING PAPER

3.6 HYDRAULIC SYSTEMS DESIGN

The orginal hydraulic systems on the Buffalo used relatively small capacity pumps with inherent limitations on surface rates. Early planning for the simulator study included considerable testing aimed at identifying acceptable flight control surface rate limits. A design decision was reached just prior to the simulator study to use larger hydraulic pumps on the modified airplane. These pumps had hydraulic flow capacity sufficient to provide rapid surface rates on the primary flight controls. A fast flap retraction rate was predicted to be essential to clean-up during a single-engine go-around with minimum altitude loss. Flap retraction between $\delta_F = 75^\circ$ and 50° was to be accomplished in two seconds followed by a slower rate between $\delta_F = 30^\circ$ and flaps up. This flap rate taxed the capacity of the larger hydraulic pumps when demanded concurrently with other control activity. The test plan for the simulator study was to evaluate how fast the flap retraction rate should be. If the fast rate were required, then primary control surface rates were to be evaluated to assure that fast surface response was indeed justified.

3.6.1 Flap Retraction Rates

The flap retraction rate was expected to affect the two-engine baulked approach as well as the single-engine go-around. However, the airplane had good climb performance even at full landing flaps ($\delta_F = 75^\circ$) on two engines at takeoff power and Pegasus nozzles rotated aft. Rate of descent on landing approach ($\delta_F = 75^\circ$, 60 KTS, 7.5 deg glide slope) could be arrested within 15 to 20 feet of altitude by power application followed by Pegasus nozzle rotation. Reversing the sequence and rotating the nozzles aft prior to power application increased the altitude loss to nearly 50 feet before climb out was established. There was no demand for flap retraction, let alone flap rate, to accomplish a two-engine go-around.

Single-engine go-around maneuvers required considerably more altitude loss as discussed in Section 4.1. Flap retraction was necessary on one engine to clean up for the go-around. Maximum altitude in the maneuver was generally reached just as the flaps were reaching $\delta_F \approx 30^\circ$ where positive single-engine climb capability existed on the airplane. Flap retraction time to reach $\delta_F = 30^\circ$ was expected to affect the height loss for a go-around. A considerable number of go-arounds were attempted at varying flap rates; however, there was a great deal of scatter in the height loss data due to variations in technique, pilot delays and experience (learning curve). By judicious choice of the data (choosing only those cases with fairly rapid pilot reaction times and increases in airspeed) the trend of altitude loss from engine failure can be deduced as shown on Figure 3.6-1. Using these trends, lines of constant flap retraction rate can be drawn on a plot altitude loss against speed excursion presented in Figure 3.6-2. All data are presented on the figure regardless of pilot delay or technique. The data scatter tends to invalidate the trend lines. For example, there are more points at $\dot{\delta}_F = 2$ DEG/SEC off the trend line than are on it. The overall conclusion is that the available data are insufficient to support any firm choice of optimum flap rate.

It can be concluded, however, that there is no requirement for a two-speed, fast-slow flap rate. Originally, the flap retraction between $\delta_F = 75^\circ$ and 50° was thought to occur at constant speed and angle of attack, in which case there would be considerable drag reduction with little lift loss.

However, in the piloted simulation the airplane began to sink and increase speed after engine failure. Flap retraction was accomplished during a varying speed condition, and the fast flap rate (12.5 DEG/SEC) was seen to prolong the airplane sink rate as illustrated in Figure 3.6-3. Go-around at a slower flap rate is illustrated in Figure 3.6-4. Sink rate tends toward the climb condition in a more orderly fashion in what appears to be better match of flap rate with single engine performance. The pilot felt that if flap retraction had been a stronger factor in achieving climb-out, then the fast rate would have been effective. However, with the single-engine performance available, the fast retraction rate actually increased the sinking tendency and further degraded the airplane.

With these data in mind the airplane flap actuation system has been designed for a single flap retraction speed adjustable up to 6 DEG/SEC.

3.6.2 Flight Control System Rate Requirements

Without the need for fast flap retraction, the planned testing of primary flight control surface rates was reduced to that reported in Section 3.1.3 (lateral control).

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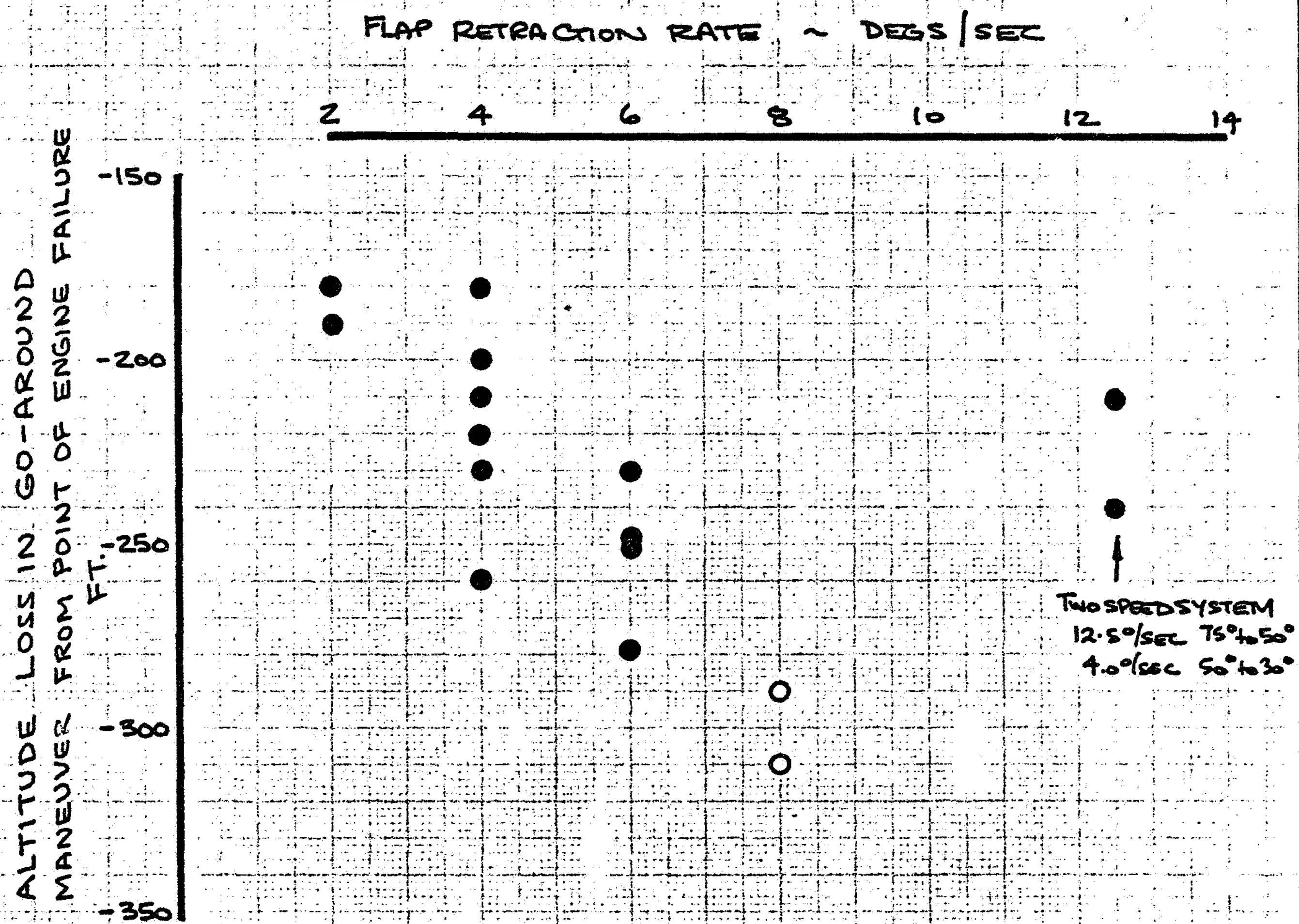
ALTITUDE LOSS FROM SELECTED SINGLE-ENGINE GO-AROUND MANEUVERS



NOTES: CLOSED SYMBOLS ARE CHOSEN DATA POINTS WHERE SPEED INCREASE LIES BETWEEN 15 AND 25 KTS. AND TIME TO $\gamma = 0$ LIES BETWEEN 15 AND 25 SECS.

OPEN SYMBOLS ARE THE ONLY AVAILABLE DATA AT 8°/SEC. BOTH POINTS HAD SPEED INCREASES GREATER THAN 30 KTS.

SIMULATOR DATA

CALC	RUMSEY	12/20/70	REVISED	DATE	THE EFFECT OF FLAP RETRACTION RATE ON THE HEIGHT LOSS IN A ONE-ENGINE GO-AROUND	D6-24806-1
CHECK						FIG 3.6-1
APR						
APR					THE BOEING COMPANY	PAGE 3.86

TD 461 C-R4

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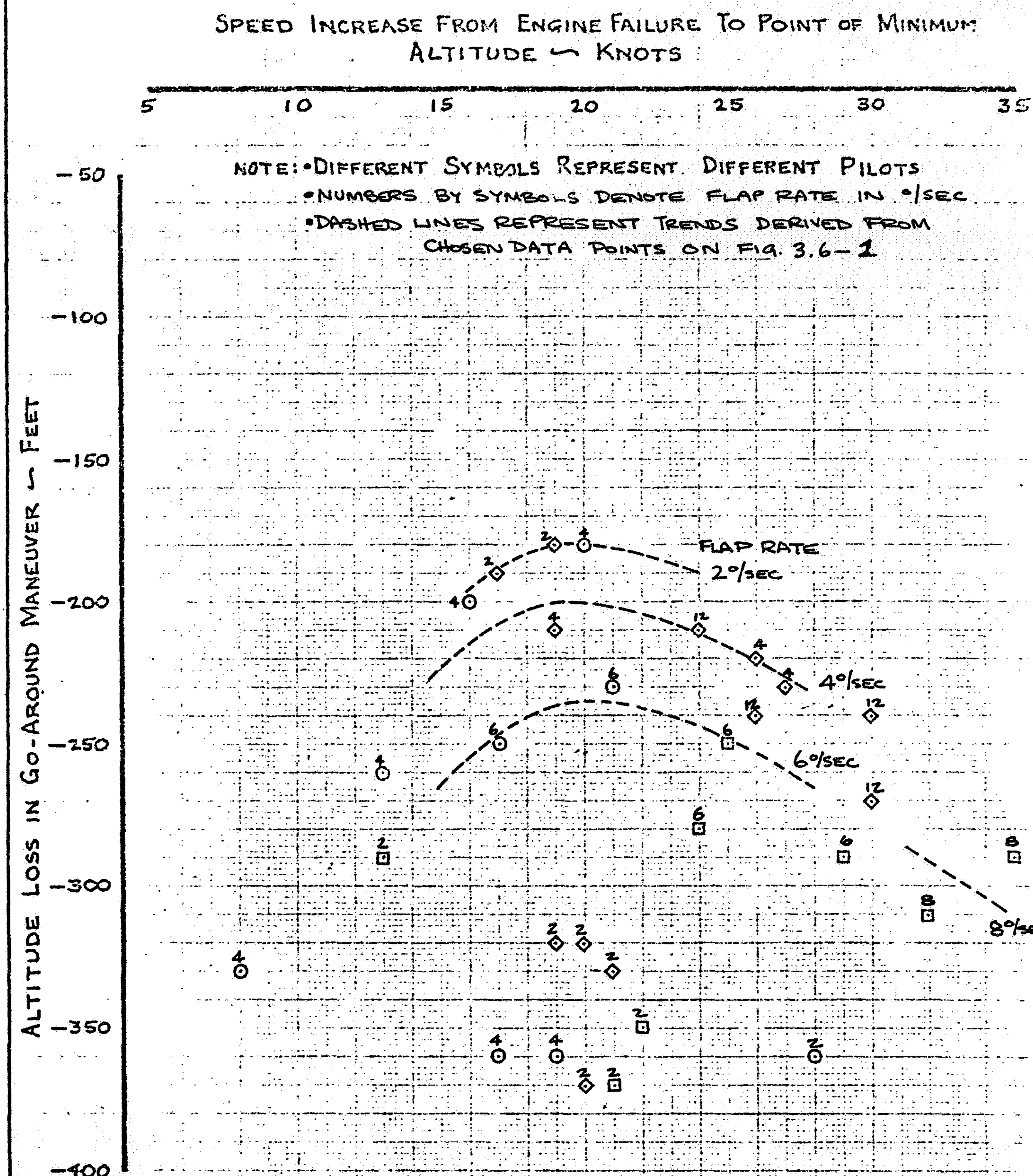
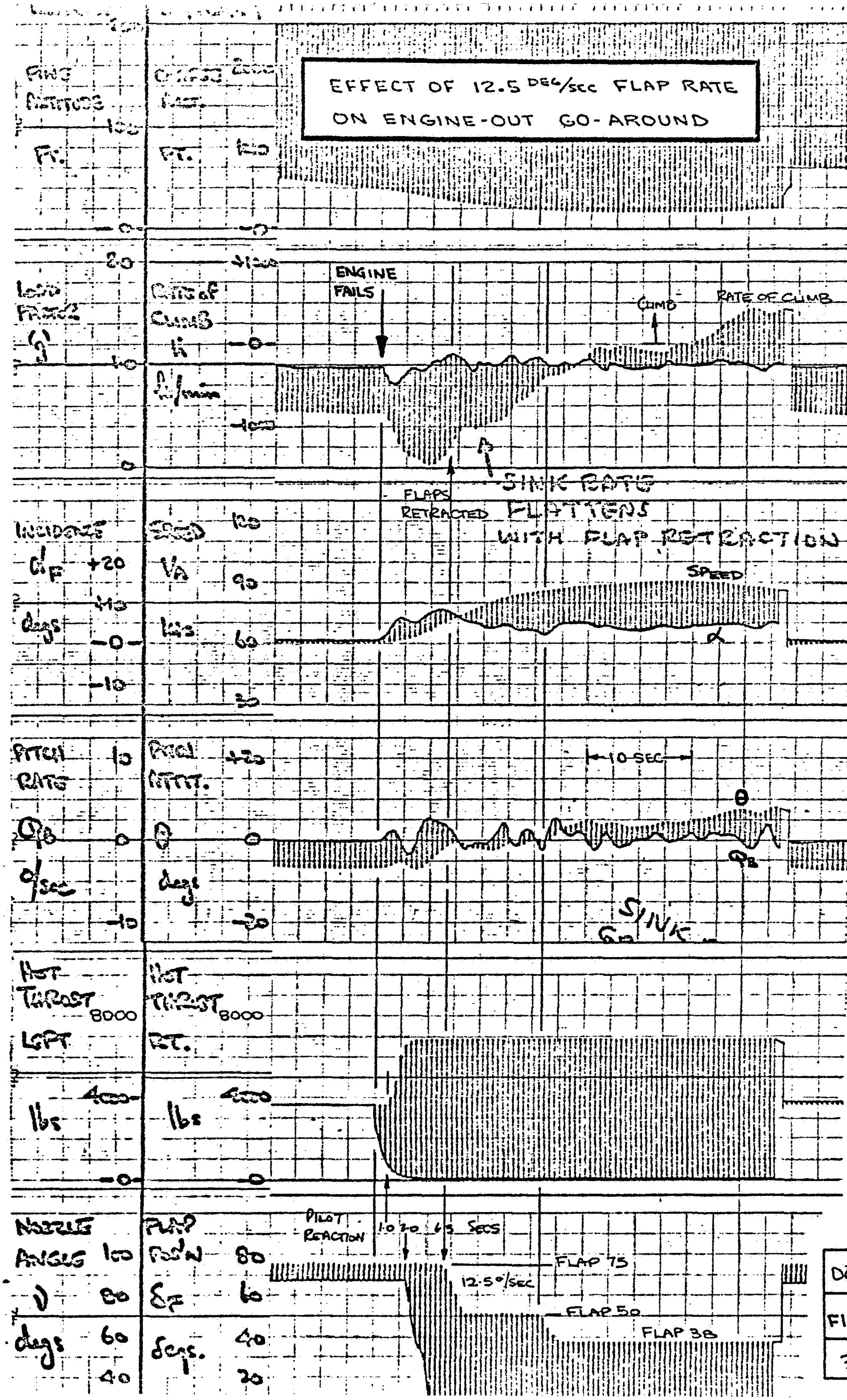
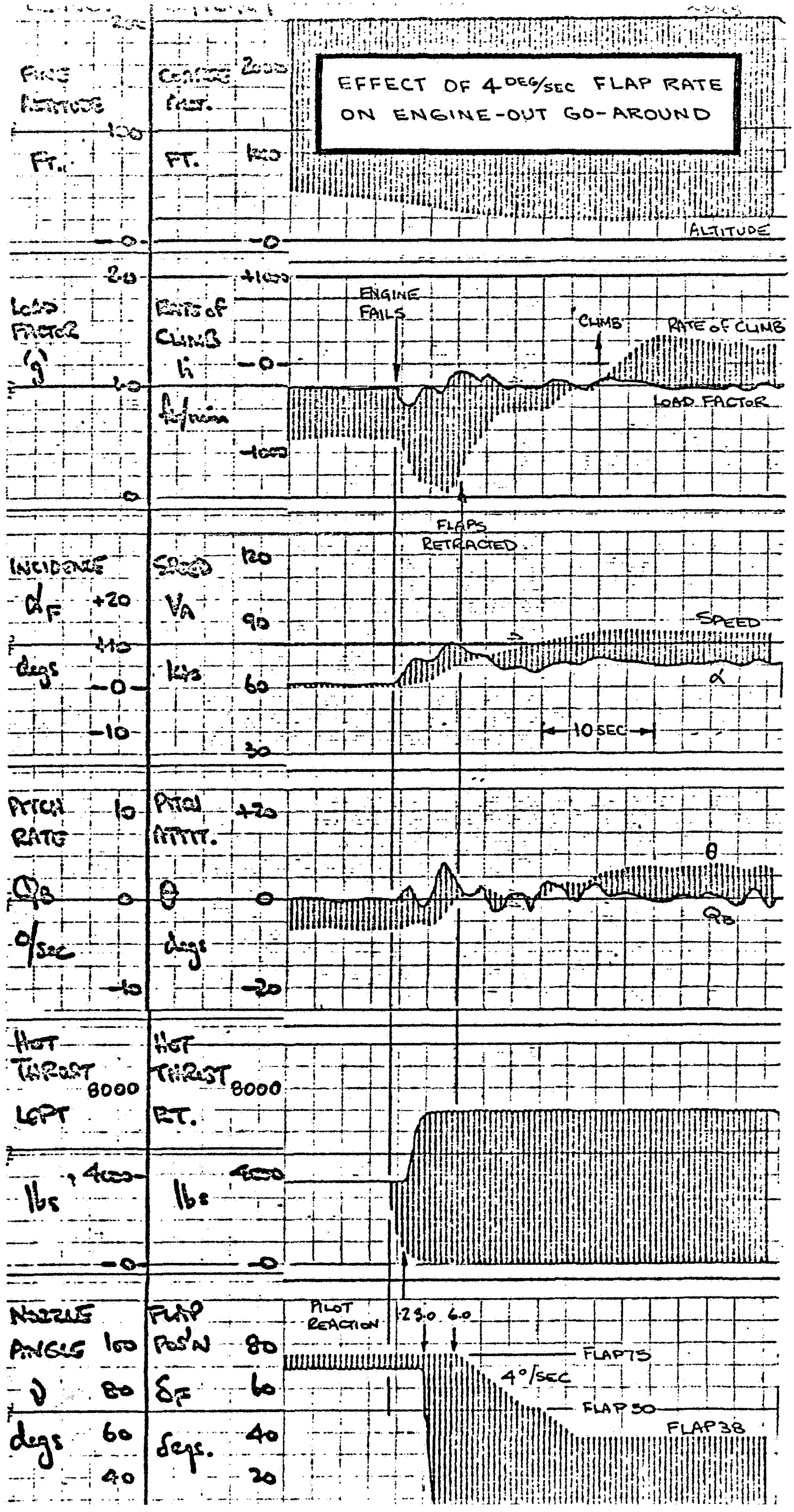


CALC	RUMSEY	12-31-70	REVISED	DATE	SINGLE ENGINE GO-AROUND SIMULATOR RESULTS	D6-24806-1
CHECK						FIG 3.6-2
APR					THE BOEING COMPANY	PAGE 3.87
APR						



D6-2480
FIG 3.6-
3.88

100



D6-24806

FIG 3.6-4

3.89

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4.0 DATA ON OPERATIONAL PROCEDURES FROM THE SIMULATOR

In the course of pursuing the investigation of design parameters for the modified Buffalo airplane a great deal was learned concerning possible operational procedures for this vehicle in the STOL mode. In particular a large number of landings were conducted in the simulator under various conditions of atmospheric environment, flight controls operational status, stability augmentation system status and with one or two engines operational. Although these data do not necessarily represent a statistically significant sample, (especially since not all landings were conducted as deliberate spot landings), the data can be used to determine trends in touchdown parameters. Unfortunately, the digital touchdown print-out was not available during the early part of the testing and was not running at various times during the tests due to line-printer unserviceability. Thus data on touchdown distance from the threshold is rather sparse, and other touchdown parameters must be read directly from the analog traces with subsequently reduced accuracy.

The investigation of lateral control requirements also involved a large number of engine failure conditions. In looking at the control problems introduced by this failure, each pilot determined his own particular technique for flying the go-around or continuing to a landing. Valuable experience has been gained by these pilots from this simulation. Analysis of the variety of conditions investigated can possibly give a guide to the trends in controllability and performance with pilot technique and airplane configuration.

4.1 ENGINE FAILURES

During the lateral control system evaluation over 70 engine failures were simulated, the choice of which engine to fail and at what altitude being randomly selected by the test engineers. After each failure the pilots were allowed to choose whether they would continue the approach and land or initiate a go-around. However, it was arranged that both landings and go-arounds were attempted from engine failures that occurred at 200 feet altitude or below.

4.1.1 Single-Engine Go-Around

The modified Buffalo has a very much reduced performance with one engine failed. With one engine at emergency power in the landing configuration a steady positive climb angle with acceptable stall warning can only be achieved by:

- Vectoring the hot thrust full aft, and
- Increasing the speed above the 60 knot approach speed at which the engine failed, and
- Retracting the flaps from 75° to the 30° position.

The requirement for all three conditions places a heavy workload on the pilot. Data from the simulator will be presented to determine that the faster the pilot's reactions the smaller is the loss in altitude before a positive climb gradient is established. This height loss is larger than that suffered by conventional airplane standards for the following reasons:

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- The STOL airplane starts from an approach glide path of 7 1/2 degrees compared to the more conventional 2 1/2 to 3 degrees.
- After engine failure the total lift available to achieve the required change in flight path angle is limited. This was especially so for the approach conditions simulated where approach power was equivalent to 75% of maximum flap blowing available.
- A loss of an engine causes a significant loss in lift.
- Achieving a positive climb gradient with one engine failed required an airplane configuration change and a speed increase.

A selected example of a one-engine go-around is shown on Figure 4.1-1. This particular case was one of those in which the altitude loss between the point at which the engine fails and the point at which the airplane first begins to climb away on one engine was a minimum. It therefore represents the maximum performance that the pilots were prepared to extract from the airplane and gives a guide to the minimum margins they were prepared to accept in this maneuver.

Examining in closer detail this go-around shows that one of the first indications of engine failure is a rapid increase in the rate of descent. This follows from the fact that the lift lost due to the decrease in flap blowing is a considerable portion of the total lift of the wing. As explained in detail in Section 3.1.8 the pilot is immediately faced with a control task to keep the wings level, the out-of-balance rolling moments arising from the loss of vectored hot thrust from the failed engine. The wheel forces simulated were high enough to require two hands for control of bank angle and this inevitably led to time delays before the pilot could release one hand from the wheel to initiate the thrust increase and nozzle vectoring required to go-around. The order in which these latter actions were made appears to be of little consequence to the final altitude loss but has considerable effect on the roll control task, (see Section 3.1.8).

The result of the downward acceleration, the increased thrust, and the thrust vectoring is an increase in airplane speed which helps to regain margin from the stall and accelerate the airplane towards a positive climb gradient. The next portion of the go-around is flown at approximately constant pitch attitude while flap is retracted and speed increased attempting to maintain the progress towards a steady climb-out.

An analysis of all engine failure conditions flown has been made to help identify pilot techniques which minimize the height loss. Figures 4.1-2 to 4.1-4 tabulate a number of parameters concerning 71 of the nearly 100 engine failures simulated. Of these, 36 were attempted one-engine go-arounds and data from these conditions are plotted in Figures 4.1-5 through 4.1-7.

The resuling height loss for each pilot is plotted against his reaction time in Figure 4.1-5. These data demonstrate the strong correlation between reaction time and height loss, although there is no definite indication that the order of actions taken is of great consequence.

Figures 4.1-6 and 4.1-7 show plots of altitude lost against the time taken to pull up to level flight, and the gain in speed from the point of engine failure to the beginning of the climbaway. There is considerable scatter in these data but it is clear that the minimum altitude loss conditions occur in distinct regions of speed increase and time taken for the recovery to level flight. It would appear that a hasty pull-up and an attempt to regain level flight quickly at the expense of speed inevitably leads to larger height losses since the climb-out cannot be maintained at the low speed. Conversely, a very gentle pull-up allowing a large build-up in speed also requires a large altitude loss. There obviously exists some optimum point at which the required kinetic energy is gained for the least expense in potential energy.

A simple examination of a circular pull-up maneuver at constant speed helps to identify limiting lines on the data of Figure 4.1-6. Figure 4.1-8 developes the equations for this type of pull-up, and the height loss is plotted against time to reach $\gamma = 0$ from an initial $\gamma = -12^{\circ}$ (an approximation to the rate of descent induced by the engine failure). Lines of constant speed are straight lines radiating from the origin. Thus, the requirement for a 75 knot climb-out speed is a lower limit on altitude loss. A given load factor pull-up is a curve across the constant speed lines. Thus $C_{L_{max}}$ constitutes another lower limit on altitude loss. Presumably pilots will always require some margin from the stall during the maneuver, thus an upper limit on α may be a more reasonable consideration than $C_{L_{max}}$. A constant α line is almost co-incident with a constant loss in altitude. This simple analysis indicates the existence of an optimum speed and load factor point that would yield a minimum altitude loss during the recovery from an engine failure. Also obvious is that the altitude loss may be minimized by:

- Giving the airplane a steady climb-out capability at a lower speed, and/or
- Giving the airplane greater load factor capability in the recovery maneuver.

Both solutions require greater installed thrust, the second one also requiring increased flap blowing.

Re-examining the engine failure depicted in Figure 4.1-1 in the light of the previous discussion, it will be seen that the pilot chose to use a maximum incident of 11^{o} ($\alpha_{wing} = 13.5^{o}$), a pitch attitude of 5^{o}, and a pull-out speed of 80-82 knots. Recognizing that the Buffalo can maintain a positive climb gradient at 75 knots at 30^{o} of flap, a reduction in the altitude lost could possibly have been gained by a more prolonged pull-up maintaining the exit speed at 75 knots. This may have been achieved by increasing the initial rotation up to about 8^{o} pitch attitude and thereafter holding 75 knots. The height lost may then have been reduced to 140 to 150 feet, which would seem to be the minimum available unless lower exit speeds and higher α's are going to be acceptable. Remembering the large spread in altitude losses tabulated in Figures 4.1-2 to 4.1-4 it would seem that a 50 to 100 foot scatter from this minimum would be a reasonable assumption to make for flight test purposes. The conclusion is then that successful one-engine go-arounds can be guaranteed only if the engine fails 250 feet or more above the ground.

In the light of the data presented, it is possible to define a technique which may help to produce consistently smaller altitude losses:

- As soon as engine failure is recognized, vector the nozzles fully up (helps to minimize roll upset and improve pilot reaction times).
- Follow this motion as soon as possible by increasing thrust on the remaining engine to the emergency power level. (This will be helped by the planned reduction in wheel forces over

these used in the simulator thus allowing one handed roll control.)

- As the power increases, initiate a smooth pull-up to a predetermined pitch attitude.
- As soon as speed is shown to be definitely increasing, initiate flap retraction directly to 30° flap.
- Continue the pull-up as necessary, or slack-off, to maintain 75 knots.

A possible improvement on this technique could be made if the throttle and vector handles were designed to allow simultaneous movement to the go-around configuration.

4.1.2 Single Engine Landings

When the engine fails below 250 feet a landing is probably inevitable. Figure 4.1-9 shows a successful one engine landing from an engine out at 150 feet. In this case the immediate reaction to an engine failure is to increase power, vector the thrust aft and leave the flaps down for maximum lift capability. Speed is allowed to build-up only as necessary to retain margin from the stall and produce reasonable body attitudes at an allowable rate of descent. As can be seen from this condition there is plenty of aerodynamic flare available and the touchdown rate of descent was held to only 3.5 ft/sec.

The actual vector angle chosen for a continued landing after engine failure varied between different pilots and depended to some extent on the technique used to control rate of descent. Using nozzle vectoring inevitably introduces changing pitch, yaw and roll moments from the operative engine's hot thrust jet. Using power changes, with the nozzle set near 70°, where the pitching moment changes are very small, introduces mainly rolling moments. Two pilots stated a preference for using thrust changes but when faced with the problem in subsequent cases all pilots used combined techniques.

Touchdowns will occur at increasingly higher angles of attack, up to a stalling flare, as the engine failure altitude comes nearer the ground. At very low altitude engine failures will result in harder touchdowns, although it is difficult to imagine rates of descent greater than 15 ft/sec unless the ground effects are excessive or the flare is incorrectly timed. All such landing will be several hundred feet short of the intended touchdown point, indicating the necessity to land this airplane well down the runway during research flying. Once again quick pilot reactions will be necessary to ensure wings level landings (to avoid damage to the gear) and to minimize the airplane excursions laterally.

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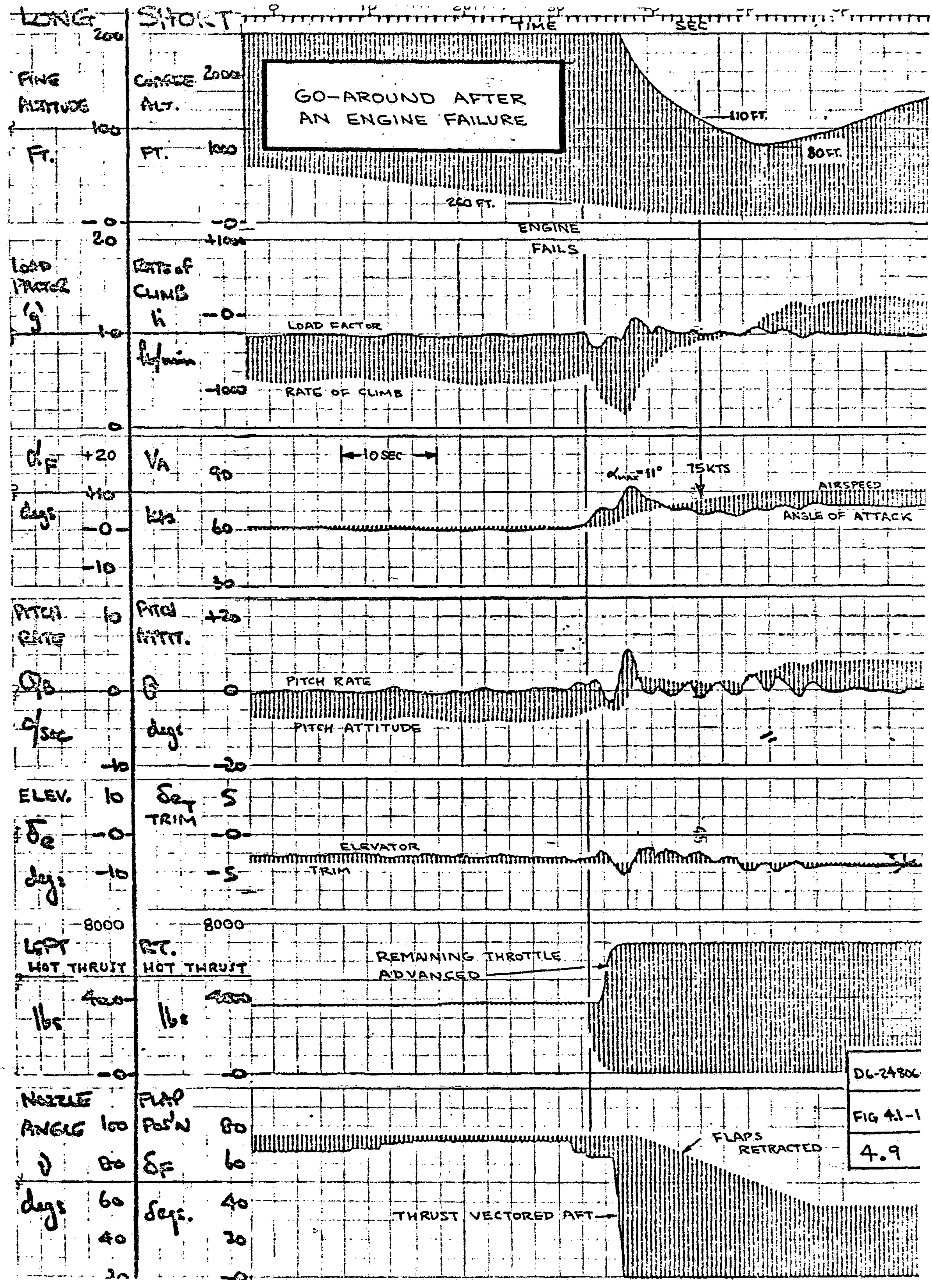

D6-24806

FIG 4.1-1

4.9

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TEST PILOT : BOB INNIS

TEST POINT #	TRIAL #	LND. OR GO AROUND	CONDITIONS AT FAILURE ($\delta_F = 75°$) ALT	V	R/D	ACTION SEQUENCE & (TIME FROM ENG FAILURE) THRUST	NOZZLE	FLAPS	CONDITION AT LEVEL OFF ΔTIME	V	ΔV	ALT	Δ ALT	δ_F	α_F	V AT BEG. OF FLAP RETRACTION	FLAP RATE	REMARKS
2	1	NEITHER	1700'	63	800	① 1.6	② 7.5	—										
	2	✓	1550'	61	680	① 1.6	② 10.0											FIXED BASE — FAMILIARISATION
	3	LND.	1800'	60	760	① 2.0	② 6.8	—										
	4	✓	2000'				②											STARTED ENGINE WITH FAILURE
3	1	G.A.	500	60	800	② 4.5	① .10	③ 36	54	79	+19	180	320	50°	5°	72	2°/SEC	LATE FLAP ACTUATION
	2	✓	380	61	800	① 1.5	② 4.5	③ 22	41.3	82	+21	50	330	38°	6.5°	78	2°/SEC	
	3	LND	650	62	720	① 1.5-2.0	② 5.5	—				—						
	4	G.A.	650	62	780	① 2.0	② 4.5	③ 31.5	55	82	+20	330	320	50°	3°	75	2°/SEC	LATE FLAP ACTUATION
	5	✓	800	60	800	① 1.3	② 5.0	③ 40	59	80	+20	430	370	50°	4°	65	2°/SEC	STARTED WITH ENG FAILURE
	6	LND	480	64	750	① 1.6	② 4.3	—				—						
	7	LND	380	65	900	① 3.0	② 5.0	—				—						
	8	G.A.	330	63	900	① 1.6	② 2.6	③ 5.0	21	80	17	140	190	50°	4°	73	2°/SEC	FLAPS STILL RETRACTING
	9	✓	260	61	800	① 1.6	② 2.6	③ 4.5	20	80	19	80	180	46°	6°	68	2°/SEC	✓ ✓ ✓
4	1	G.A.	350	60	860	① 1.3	② 3.0	③ 5.5	20.5	84	24	140	210	38°	7°	70	12°/SEC	
	2	✓	490	60	840	✓ 1.0	✓ 3.0	✓ 6.5	28.5	90	30	220	270	38°	4°	72	12°/SEC	
	3	✓	470	60	800	✓ 1.2	✓ 3.0	✓ 6.5	24.0	90	30	230	240	35°	4.5°	71	12°/SEC	
	4	✓	370	60	900	✓ 1.4	✓ 2.5	✓ 7.0	20.0	86	26	130	240	38°	5.5°	70.5	12°/SEC	
5	1	G.A.	430	60	850	① 1.2	② 3.0	③ 5.5	20.0	86	26	210	220	38°	6°	71	4°/SEC	
	2	✓	480	61	860	① 1.3	② 3.0	✓ 6.5	15.0	80	19	270	210	42°	6°	69	4°/SEC	
6	1	LND	600	61	770	① 2.0	② 5.0					—	—					
	2	✓	480	59	800	✓ 2.5	✓ 5.0					—	—					
	3	G.A.	510	60	810	✓ 1.2	✓ 2.5	③ 6.0	20.5	87	27	280	230		5°	72	4°/SEC	

Calc.	K. ELBEL	12/10/70	REVISED	DATE	TABULATION OF SIMULATED ENGINE FAILURE CONDITIONS	D6-24806-1
Trac						FIG 4.1-2
Chk.						
Appr.					THE BOEING COMPANY	4.10

DATA SHEET

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TEST PILOT : BOB FOWLER 11/4/70 + 11/5/70

TEST POINT #	TRIAL #	LND. OR GO AROUND	CONDITIONS AT ENG. FAILURE (δ_F=75°) ALT	V	R/D	ACTION SEQUENCE & TIME FROM ENG FAILURE THRUST	NOZZLE	FLAPS	CONDITIONS AT LEVEL OFF ΔTIME	V	ΔV	ALT	ΔALT	δ_F	α_F	V @ BEGIN OF FLAP RETRACTION	FLAP RATE		REMARKS
11/4/70 1	1	LND	740'	64	870	① 3.5	② 13.5	—											
	2	✓	720'	60	940	✓ 3.5	✓ 9.5	—											
	3	G.A.	660'	62	950	✓ 1.5	✓ 9.5	③ 27.5	45.5	80	18	200'	460'	40°	8°	78	2° sec		LATE FLAP ACTUATION
	4	LND	390'	59	920	✓ 2.5	✓ 8.0	—											
	5	✓	350'	66	800	✓ 1.5	✓ 6.0	—											
	6	G.A.	490'	64	890	✓ 1.5	✓ 5.6	③ 16.5	24	77	13	200'	290'	60	6°	77	2°/sec		LATE FLAP ACTUATION
	7	✓	390'	64	830	✓ 1.5	✓ 5.8	✓ 13.5	53	102	38	108'	290'	UP	12°	73	2°/sec		FLAPS RETRACTED FULLY UP
	8	LND.	250'	57	800	✓ 1.2	✓ 3.0	—											
	9	G.A.	600'	63	820	✓ 2.5–4.8	✓ 7.5	③ 11.5	33	85	22	250'	350'	32°	10.5	73	2°/sec		
2	1	LND	800'																DOUBLE ENG FAILURE
	2	G.A.	490'	60	900	① 2.0	② 10.5	③ 17.0	41.5	81	21	120'	370'	27°	8.5°	72	2°/sec		SLOW REACTIONS
	3	✓	550'	60	850	✓ 2.0	✓ 6.0	✓ 15.0	51	106	46	260'	290'	UP	11°	76	2°/sec		FLAPS RETRACTED FULL UP
3	1	G.A.	450'	61	850	① 2.0	② 5.0	③ 11	22.5	85	24	170'	280'	26°	9.5°	74	6°/sec		
	2	✓	440'	59	900	✓ 1.5	✓ 5.0	✓ 11.5	33.0	97	38	160'	280'	UP	12.5	73	✓		FLAPS RETRACTED FULL UP
	3	✓	480'	59	830	✓ 1.4	✓ 4.0	✓ 8.0	28.5	96	37	220'	260'	UP	13.5	71	✓		✓ ✓ ✓ ✓
4	1	G.A.	450'	59	840	① 1.8	② 6.4	③ 14	32	94	35	160'	290'	UP	14.5°	76	8°/sec		✓ ✓ ✓ ✓
	2	✓	380'	58	880	✓ 1.8	✓ 5.0	✓ 15	41	90	32	70'	310'	UP	16°	72	8°/sec		✓ ✓ ✓ ✓
6	1	LND	490'	61	860	① 2.0	② 5.0	—											
	2	✓	470'	59	920	✓ 2.5	✓ 7.5	—											
7	1	LND	640'	61	880	① 2.4	② 6.5												
11/5/70 8	1	G.A.	730'	60	810	① 2.5	② 5.0	③ 10	34.5	89	29	440'	290'	UP	15.5	79	6°/sec		
	2	✓	430'	59	430	✓ 2.0	✓ 6.5	✓ 11	22	74	25	180'	250'	30°	13°	62	6°/sec		
9	1																		
	2	LND	195'	64	700	① 2.0	② 3.5	—											
	3	✓	155'	61	750	✓ 1.2	✓ 2.5	—											

Calc.	KEISEL	12/10/70	REVISED	DATE	TABULATION OF SIMULATED ENGINE FAILURE CONDITIONS	D6-24806-1
Trac						FIG 4.1-3
Chk.						
Appr.					THE BOEING COMPANY	4.11

DATA SHEET

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TEST PILOT : TOM EDMONDS

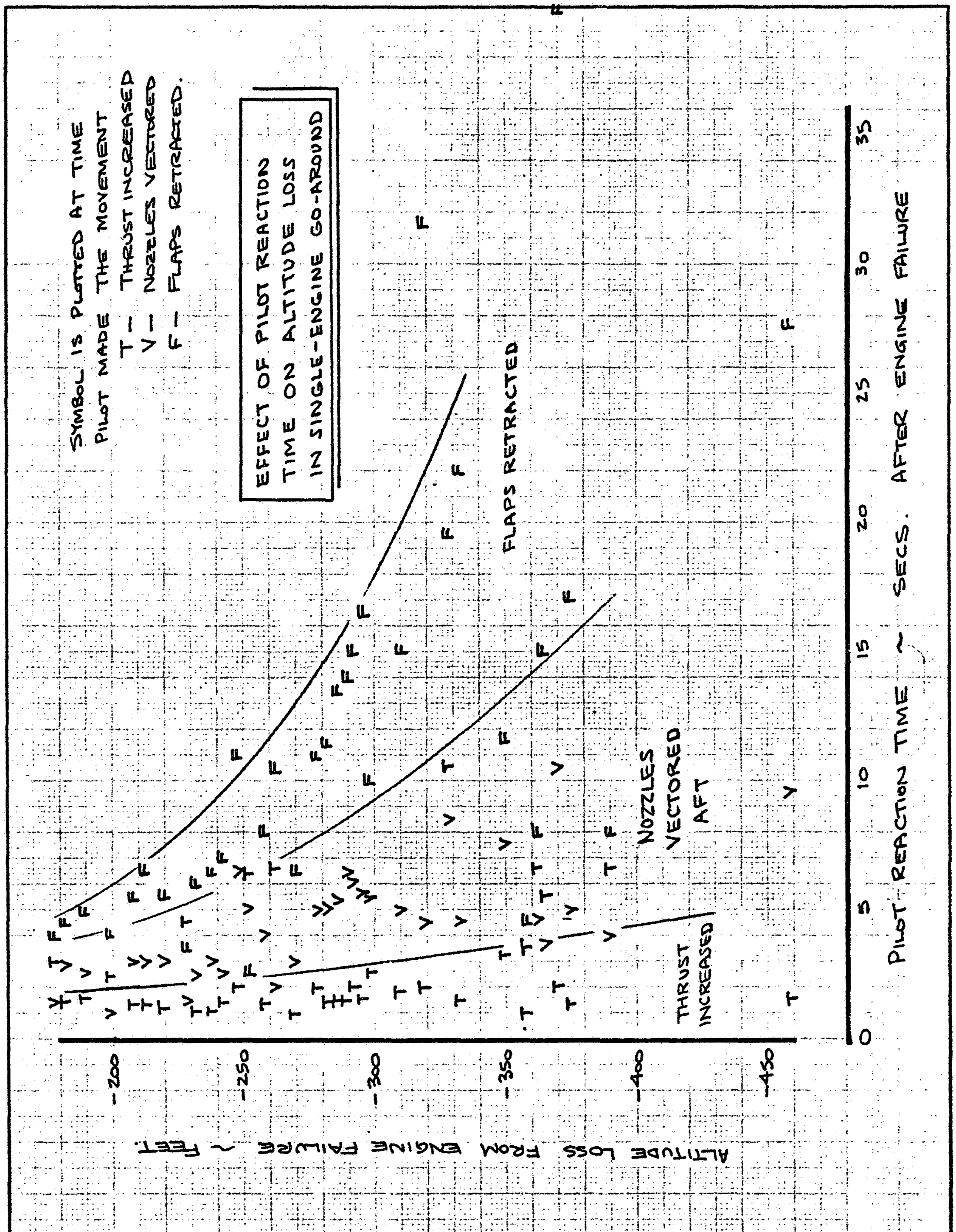
TEST POINT #	TRIAL #	LND. OR GO AROUND	CONDITIONS AT ENG. FAILURE ($\delta_F = 75°$) ALT	V	R/D	ACTION SEQUENCE & TIME FROM ENG. FAILURE THRUST	NOZZLE	FLAPS	CONDITIONS AT LEVEL OFF ΔTIME	V	ΔV	ALT	ΔALT	δ_F	α_F	V @ BEGIN OF FLAP RETRACTION	FLAP RATE		REMARKS
1	1	LND.	1750'	60	770	② 50	① 25												FIXED BASE
	2	LND	1720'	61	770	② 47	① 15												– FAMILIARISATION
2	1	LND	650'	64	920	① 3.5	② 15.5												
	2	✓	650'	64	930	② 18.5	① 6.5												
	3	✓	500'	63	750	✓ 1.6	✓ 1.0												
	4	✓	480'	62	778	✓ 4.5	✓ 2.2												
	5	✓	400'	64	800	✓ 4.0	✓ 1.0												
	6	✓	650'	64	920	✓ 4.5	✓ 2.0	③ 9.0											RAISED FLAPS TO LAND
	7	G.A.	550'	63.5	920	✓ 6.5	✓ 4.0	✓ 8.0	45	109	45.5	160'	390'	UP	8 - 9°	75	2°/SEC		FLAPS RAISED FULLY UP
	8	✓	420'	62	780	✓ 3.5	✓ 1.0	✓ 4.5	22	90	28	60'	360'	40°	7.5°	70	✓		LARGE SPEED INCREASE
3	1	G.A.	440'	64	700	② 6.5	① 4.5	③ 8.0	24	83	19	80'	360'	38°	10°	74	4°/SEC		USED. TO. POWER ONLY
	2	✓	750'	60	800	✓ 23.0	✓ 6.5	NEVER				NO POSITIVE R/C					✓		FLAPS NOT RETRACTED
	3	LND.	150'	57	790	✓ 3.0	✓ 1.0	—				—							
	4	G.A.	600'	66	960	✓ 6.5	✓ 4.0	③ 36.	38	83	17	50'	550'	70°	4°	84	4°/SEC		LATE FLAP RETRACTION
	5	✓	510'	64	830	✓ 2.5	✓ 1.0	③ 4.	15.5	82	16	310'	200'	38°	7°	72	✓		
4	1	G.A.	590'	64	850	② 6.5	③ 5.0	① 2.5	12.5	81	17	340'	250'	38°	11°	73	6°/SEC		THRUST LAST
	2	✓	490'	63	740	② 4.5	① 1.5	③ 3.5	12.0	84	21	260'	230'	38°	8°	69	✓		✓ ✓
5	1	G.A.	600'	64	870	② 7.5	① 2.0	③ 10.5	12.0	77	13	340'	260'	38°	4.5°	78	4°/SEC		
6	1	G.A.	600'	64	980	② 5.5	① 3.5	③ 15.	32.6	81	17	240'	360'	38°	8°	74	4°/SEC		SLOW REACTION
	2	✓	500'	64	750	✓ 18.5	✓ 8.5	✓ 19.5	24.0	72	8	170'	330'	57°	7.5°	71	✓		SLOW REACTION
7	1	G.A.	390'	62	770	② 3.0	① 1.5	③ 4.	20.	82	20	210'	180'	38°	7.5°	69	4°/SEC		
	2	LND.	180'	59	710	✓ 3.5	✓ 1.0												NO TOUCH DOWN
	3	—	450'	60	900	✓ 6.0	✓ 2.5												
	4	LND.	490'	60	800	✓ 5.0	✓ 2.5												

Calc.	K. ELBEL	12/10/70	REVISED	DATE	TABULATION OF SIMULATED ENGINE FAILURE CONDITIONS	D6-24806-1
Trac						FIG 4.1-4
Chk.					THE BOEING COMPANY	4.12
Appr.						
Appr.						

AD 360 B-R4

DATA SHEET

113



CALC	RUMSEY	12/7/70	REVISED	DATE	EFFECT OF PILOT REACTION TIME ON ALTITUDE LOSS IN SINGLE ENGINE GO-AROUND.	D6-24806-1
CHECK						
APR						FIG 4.1-5
APR					THE BOEING COMPANY	PAGE 4.13

TD 461 C-R4

K+E ALBANENE 196L TRACING PAPER

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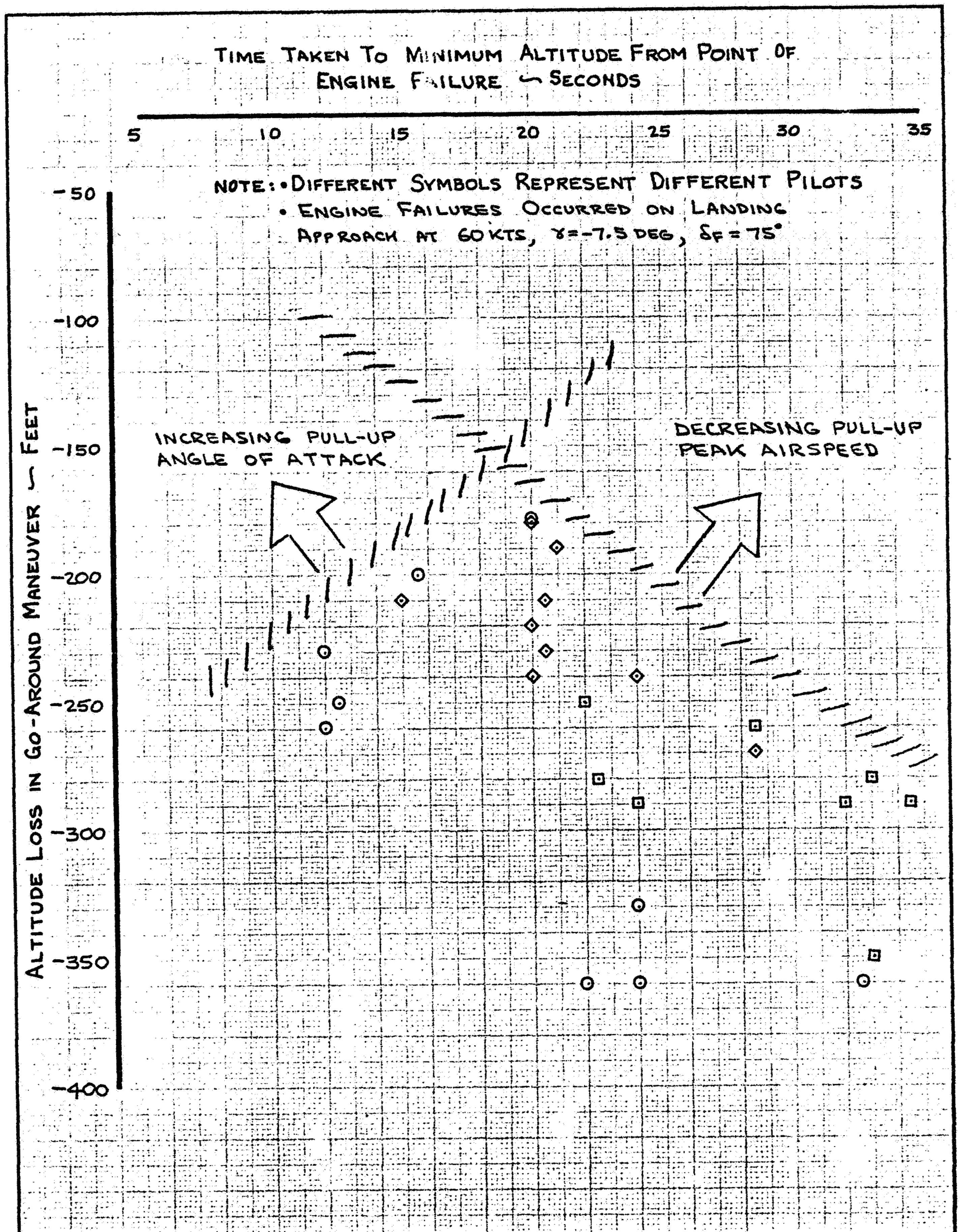
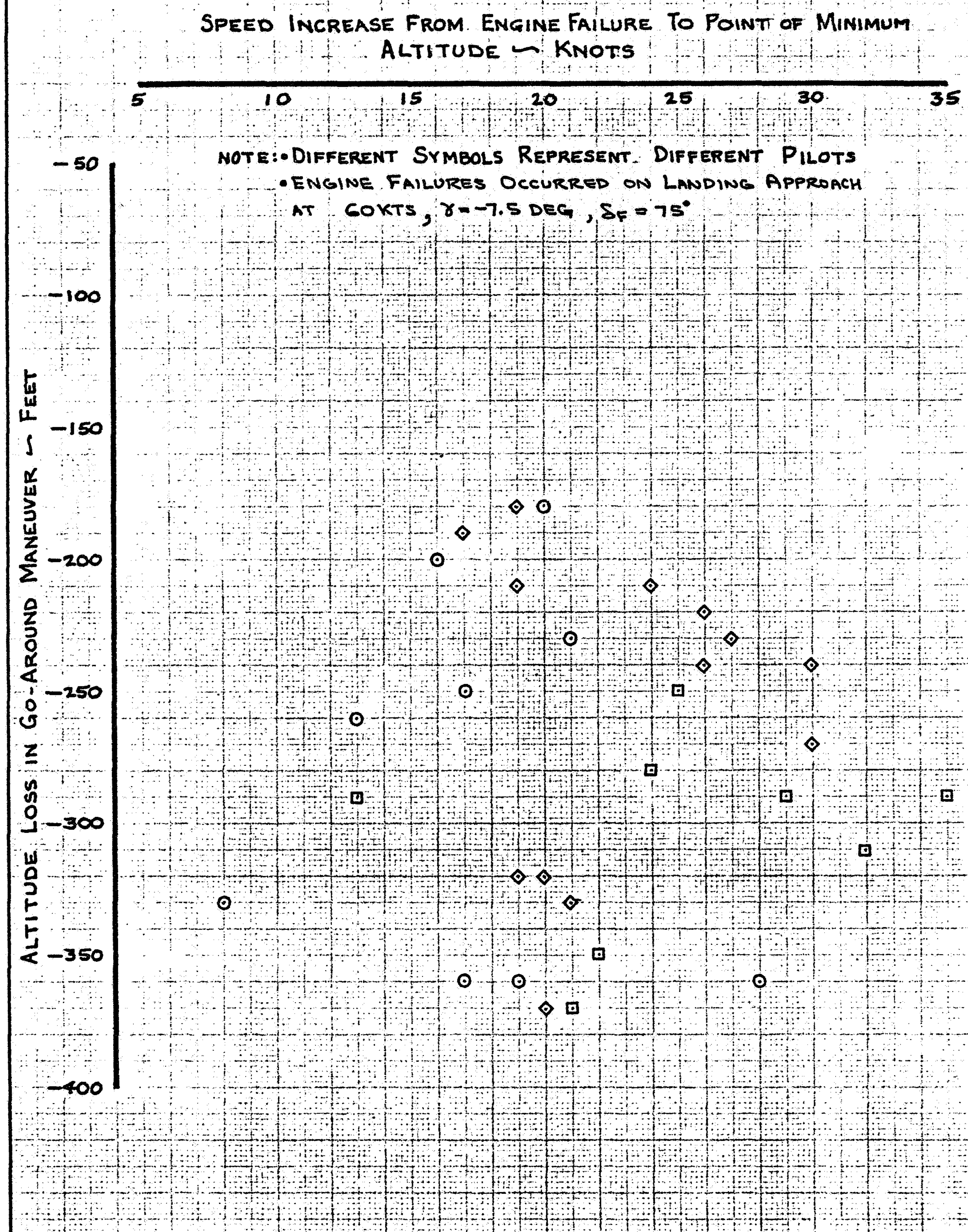


CALC	RUMSEY	12-31-70	REVISED	DATE	SINGLE ENGINE GO-AROUND SIMULATOR RESULTS	D6-24806-1
CHECK						FIG 4.1-6
APR					THE BOEING COMPANY	PAGE 4.14
APR						

AD 463 C-R6

K+E ALBANENE 196L 8.5 G-8000



CALC	RUMSEY	12-31-70	REVISED	DATE
CHECK				
APR				
APR				

SINGLE ENGINE GO-AROUND SIMULATOR RESULTS

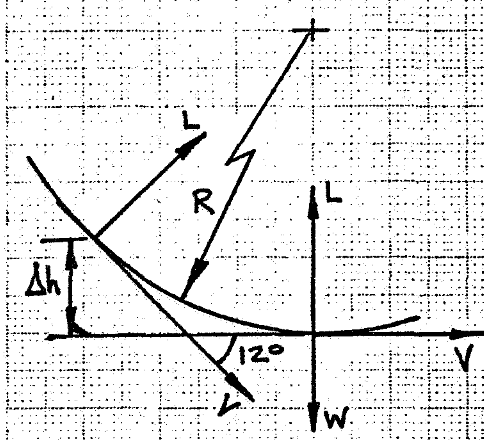
THE BOEING COMPANY

D6-24806-1

FIG 4.1-7

FOR A SYMMETRICAL CIRCULAR PULL-UP AT CONSTANT SPEED

from $\gamma = -12°$ to $\gamma = 0°$



pull-up at constant load factor $n = \frac{L}{W}$

HEIGHT LOSS $\Delta h = R(1 - \cos 12°)$

$\doteq .02\,R$

TIME TAKEN $\Delta t = R \times \frac{12}{57.29} \times \frac{1}{V}$

$\doteq .21\,\frac{R}{V}$

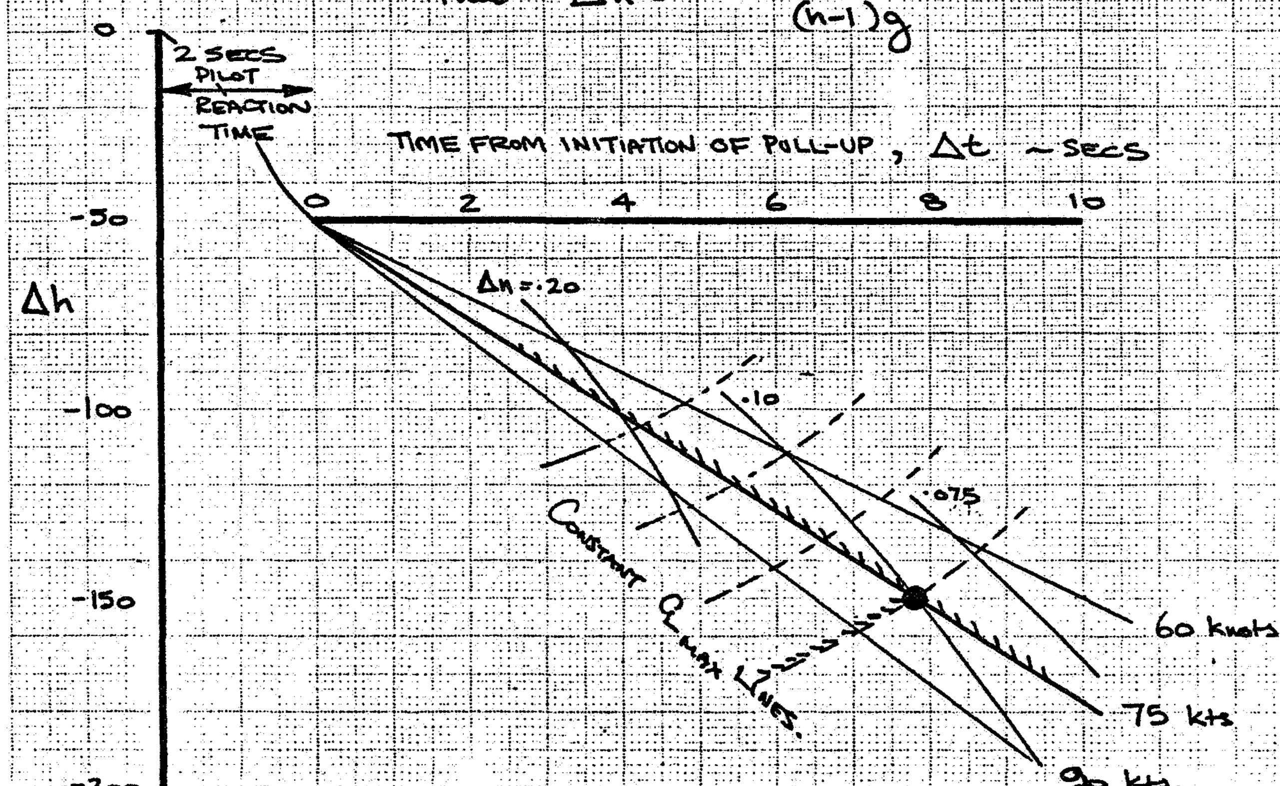
Thus height loss and time are related by:–

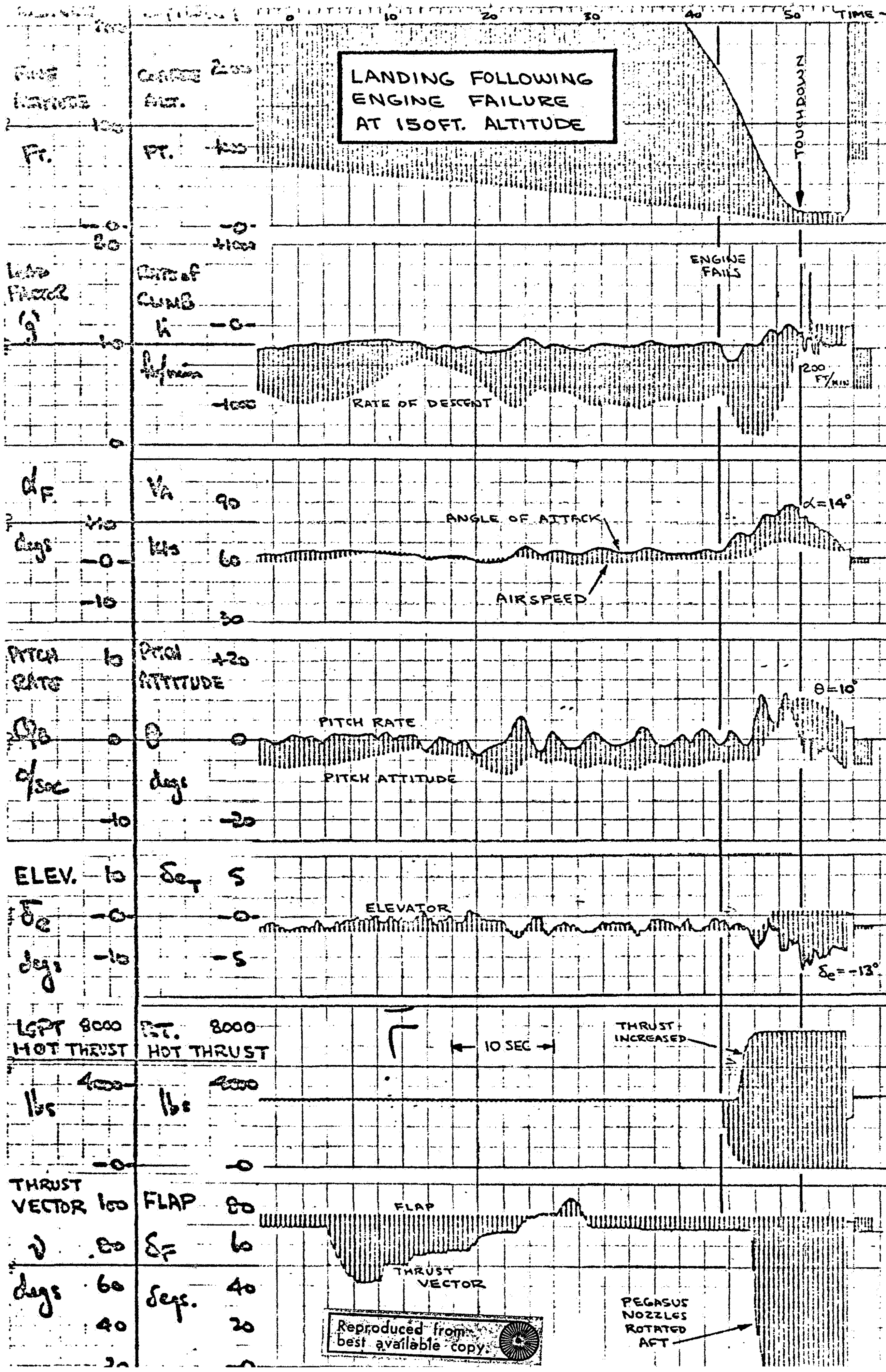
$$\Delta h \doteq .1\,V\Delta t$$

By mechanics of the pull-up

$$\frac{L-W}{W} = \frac{V^2}{gR} = n-1$$

Thus $\Delta h \doteq .02\,\frac{V^2}{(n-1)g}$





D6-2480

FIG 4.1-

4.17

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4.2 FLARE TECHNIQUES

All three simulator pilots experienced difficulties in consistently producing successful landings in the simulator. Their comments suggested that the visual cues were in error (possibly from an error in mechanization, which was never found, or more likely due to the lack of depth of field and peripheral vision which is a recognized shortcoming of closed-circuit TV visual presentations). The most frequent complaint was that the flare had to be initiated too high above the ground. The altitude shown on the radio altimeter at the initiation of flare seemed to be lower than the altitude indicated by the visual scene. Typically pilots underestimated the touchdown rate of descent by 4 ft/sec.

This problem, coupled with the severe simulated ground effects on aerodynamic lift, produced very high touchdown velocities for most of the landings accomplished. A direct application of the results of the flare maneuver from the simulator to actual flight is therefore not considered accurate. However the simulator data can be used as a guide to flare techniques which may be useful in alleviating problems which might occur in full scale flight testing.

The simulated ground effects, see Appendix 7.2, were taken direct from the Phase V Ames 40x80 wind tunnel test data, Reference 7. Ground effects were fixed at the values for 75° flaps deflection with the hot thrust jets vectored fully down and blowing hard. This configuration produces the worst possible ground effects, which were almost certainly over-estimated in the wind tunnel data due to ground board boundary layer interference and incorrect hot jet simulation. The wind tunnel model nozzle exhaust was considerably closer to the ground than in the actual airplane configuration. With full ground effects as simulated

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the airplane suffered a 15% loss in lift during a typical flare condition. No action by the pilot could alleviate this since the simulator nozzle vectoring or thrust reduction did not reduce the ground effect as it would in flight.

In full ground effect touchdown rates of descent were averaging about 9 ft/sec. However, there was a fairly significant difference between pilots.

One pilot, who was particularly sensitive to the erroneous indicatinn of flare height, began his flare at a significantly lower height than the other two pilots. This pilot began flare at about 35 to 40 feet. Touchdown attitudes were about 3° (from an approach attitude of -6°). This rotation, and the sink induced by the severe ground effect,produced an angle of attack of 10° (fuselage datum) at touchdown, giving a descent angle of about 7°. At a touchdown speed of 60 knots this gave a rate of descent of 11 to 12 ft/sec.

The second pilot tended to ease the nose up gently from about 150 feet altitude, flaring hard from 55 feet on average. Touchdown attitudes were 5 to 6° with an α of 14°. The early pitch change however had significantly reduced the approach speed below 60 knots and touchdown speed was in the order of 55 knots, giving a rate of descent of 13 ft/sec on average.

The third pilot generally approached a little fast at about 64 knots and flared from 50 feet. Touchdown pitch attitude was 6°; α was 11°, giving a rate of descent of 8 ft/sec at 60 knots touchdown speed.

Note that these data were gathered from landings made during the evaluation of the lateral control system and therefore do not necessarily represent the best

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performance from each pilot. In fact, during later tests the second pilot developed a flare technique using power and elevator which gave touchdowns of 5 ft/sec at attitudes of 1 to 2°.

A number of landings were also made with a reduced level of ground effect. Drag and wing-body pitching moment changes were put to zero (since previous investigation had shown these terms to have very little effect on the flare) and the loss of lift was reduced by 50%. The downwash changes were left in at full strength. Figure 4.2-1 shows a typical flare using elevator alone in this modified ground effect.

Figure 4.2-2 shows a summary of touchdown rates of descent achieved under each of the different ground effect conditions. Full ground effects were run on two separate occasions; the first two days of flying, and a little later during the lateral control evaluation. No strong learning curve effect was seen, and the average touchdown rate of sink for both sets of data was 8.8 ft/sec. With no ground effects (except for the downwash changes) the touchdowns averaged 5.2 ft/sec., a figure which was bettered later in the simulator flying using a combined power and elevator flare technique with the modified ground effects. This latter technique was developed by one of the pilots to take advantage of the strong lift control (at relatively constant speed) that is available from thrust increases with nozzles vectored at 90° (see Section 3.2.1). Figure 4.2-3 shows a landing using this technique which reduces the elevator required to flare and allows flare initiation to be delayed to more conventional altitudes.

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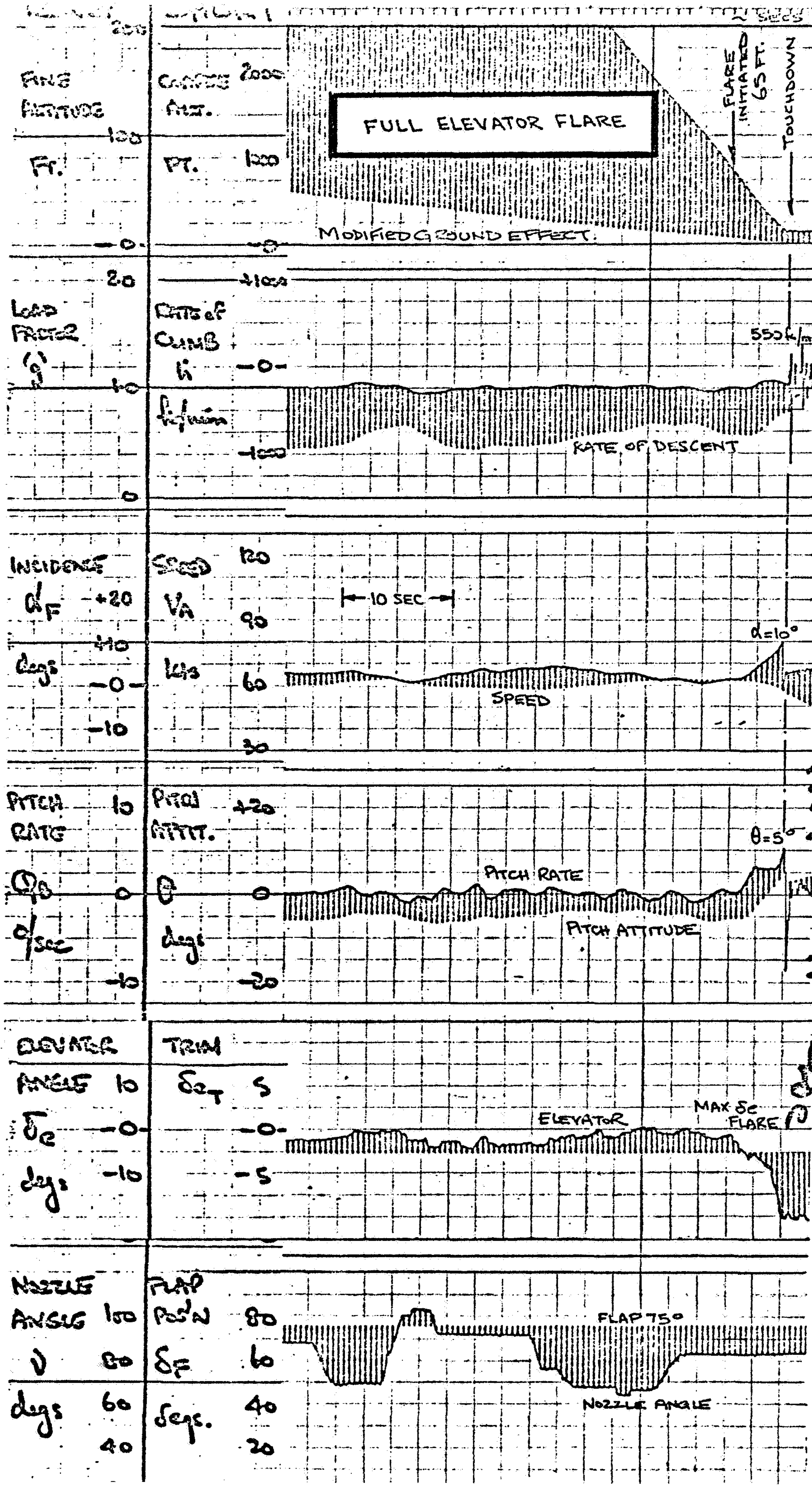
The difficulties experienced by the pilots in successfully flaring the simulator and the severity of the ground effects demonstrated to them led to the discussion of procedures which could be used in the initial flight test stage to move slowly and deliberately into the regions where ground effects may be at their worst. Simulated landings had already shown that landings from 3 degree glideslope approaches at 90-100 knots and flaps at 30° were quite conventional in character. This was therefore assumed to be the initial takeoff and landing flap setting.

Landings at other flap settings or with the Pegasus nozzles vectored down would be delayed until clearance could be given. This clearance would be based on the airplane behaviour during ground hops as detailed in Bob Fowler's report in Appendix 7.1. Should very large lift losses be evident at the landing flap configuration then flare techniques with power and elevator would have to be developed to ensure reasonable touchdowns from STOL approaches. Development would obviously begin with simulated flares at altitude followed by landings at flap and nozzle configurations which showed little or no ground effect during the ground hop tests.



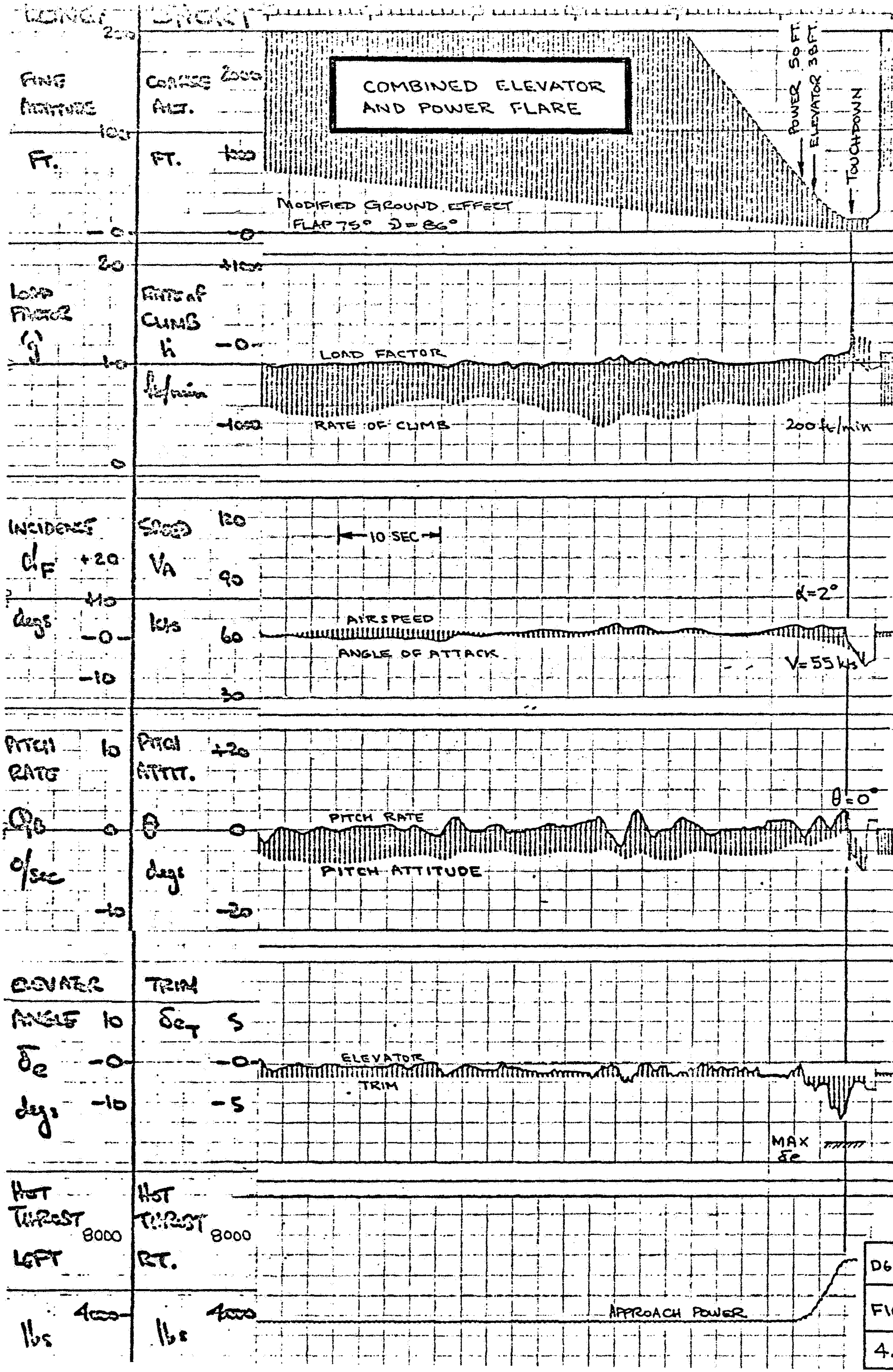
D6-24806

FIG 4.2-1

SUMMARY OF TOUCHDOWN RATES-OF-DESCENT

DATE	CONDITIONS	AVERAGE TOUCHDOWN R/D ft/sec	NUMBER OF LANDINGS
10/27/70	Full Ground Effect	9.2	11
10/28/70	No Ground Effect	5.2	22
10/30/70	Full Ground Effect	8.6	23
Overall (4 Days Analyzed)	Full G.E.	8.8	34
	Modified G.E.	4.9	8
	No G.E.	5.2	22

FIG 4.2-2



D6-2481

FIG 4.2-

4.24

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4.3 LATERAL-DIRECTIONAL HANDLING QUALITIES

A brief description of lateral-directional dynamic response is provided here to aid in analysis of pilot comments on airplane handling qualities. Use is made of the airplane dynamic response to a step wheel input (held in for three seconds) in comparison with responses known to give good pilot rating.

Generally speaking, if the parameters affecting turn coordination are well behaved for a simple wheel input then good turn performance can be generated by the pilot with very little compensation on his part. This characteristic is a requirement for good pilot rating.

4.3.1 Stabilized Airplane

Figure 4.3-1 shows the airplane response to a step wheel input with the Stability Augmentation System engaged.

With SAS on there is very little difference in airplane response over the range of possible values of the derivative C_{ℓ_β}. With $C_{\ell_\beta} = -.25$ there is a little less dutch roll damping available, whereas at $C_{\ell_\beta} = 0$ larger amplitude SAS inputs were required to tame the spiral mode instability. In the turn entry, only small sideslip angles are induced thus requiring little or no pilot rudder inputs for coordination. However the roll mode time constant is perhaps a little too long for optimum pilot opinion requiring some pilot anticipation to roll up to and hold a selected bank angle. As the airplane rolls into the turn, there is a noticeable delay before airplane heading responds. SAS on, this delay is of the order of two seconds which should be acceptable. One of the simulator pilots successfully used rudder to improve the heading response. However he commented that on this airplane it was difficult to coordinate use of the rudder

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because of the long dutch roll period. This configuration was rated overall at 4.5 to 5.0 on the Cooper-Harper scale. Better ratings were obtained by:

- increased roll damping in the SAS
- reduced airplane roll moment of inertia
- increased roll control effectiveness
- use of a control wheel steering augmentation system with attitude feedback and increased roll rate damping.

4.3.2 Free Airplane

Figure 4.3-2 shows the airplane response to a step wheel input without SAS and with $C_{\ell\beta} = -.25$. Without SAS, the aerodynamic cross coupling induces large sideslip angles in the turn entry making turn coordination almost impossible. The heading response lag is now over four seconds, an unacceptable situation.

With $C_{\ell\beta} = 0$ the spiral mode is so unstable that the airplane response to this type of step input is a continuing roll rate even when the step input is removed. To fly the airplane wings level, requires almost 100% lateral control. In mild gusts the airplane response was considered "wild".

This configuration was rated 8.0 to 9.0 on the Cooper-Harper scale. However, simulated flights were successfully accomplished SAS off in gusts, in IFR conditions, and even with one engine failed.

Airplane dynamic characteristics improve rapidly with speed and/or reduced flap deflection below 50°. Approaches and landings were successfully made with manual control of the ailerons (simulating a dual hydraulic failure) and SAS off at 90 to 100 knots and with a flap angle of 30°.

At nominal cruising speeds flaps up, airplane dynamics are considered acceptable without SAS. The only mildly objectionable feature is the still slightly unstable spiral mode. Figure 4.3-3 gives a summary of the main dynamic characteristics for four flight conditions.

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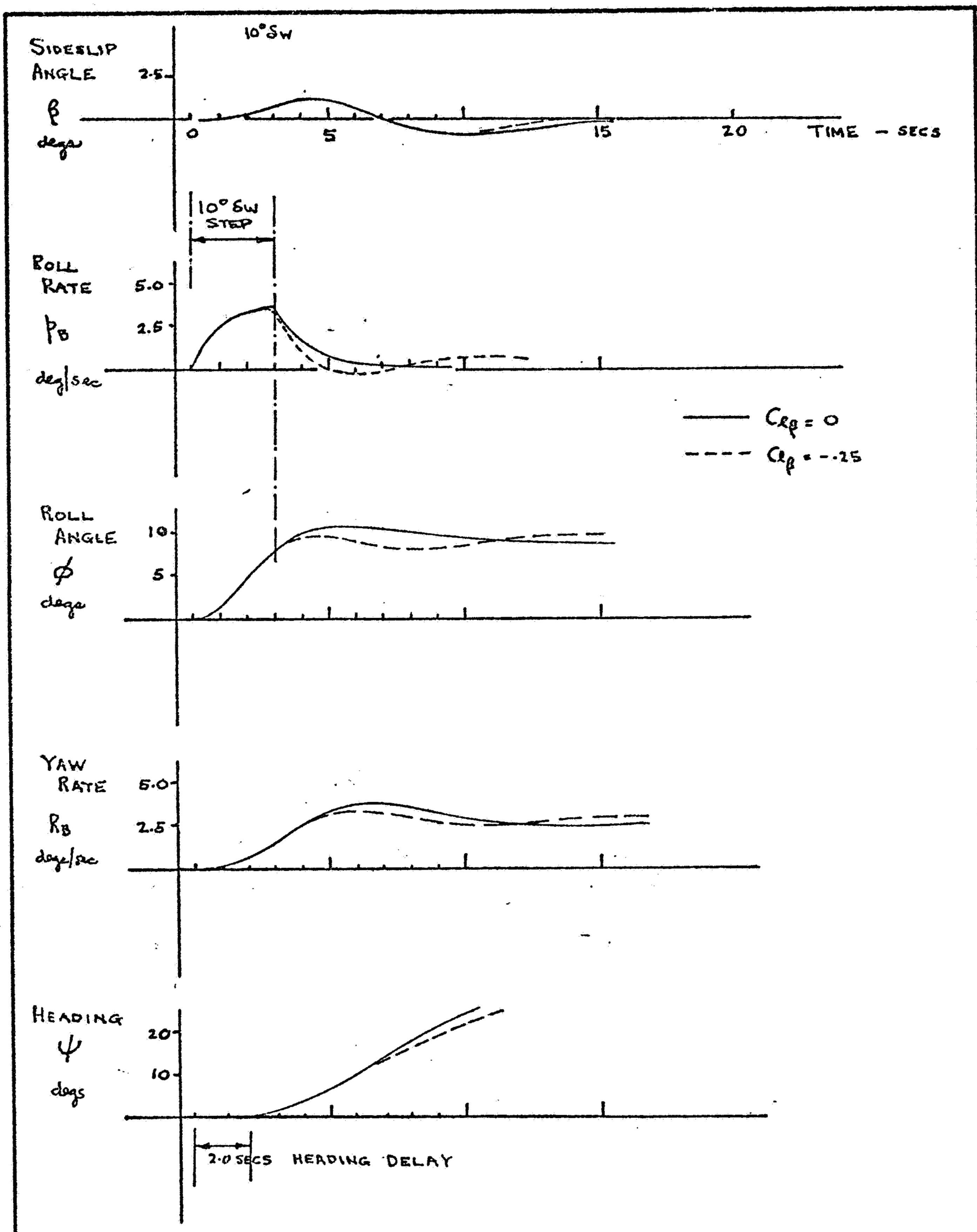

			REVISED	DATE		
CALC					RESPONSE TO 10° WHEEL STEP FOR 3 SECS	D6-24806-1
CHECK					SAS ON ~ VARYING C_{ℓ_β}.	
APPD	MARK	1-11-71			STOL LANDING CONFIGURATION	FIG 4.3-1
APPD					THE BOEING COMPANY RENTON, WASHINGTON	PAGE 4.27

TD 873 D-R3 6 7000

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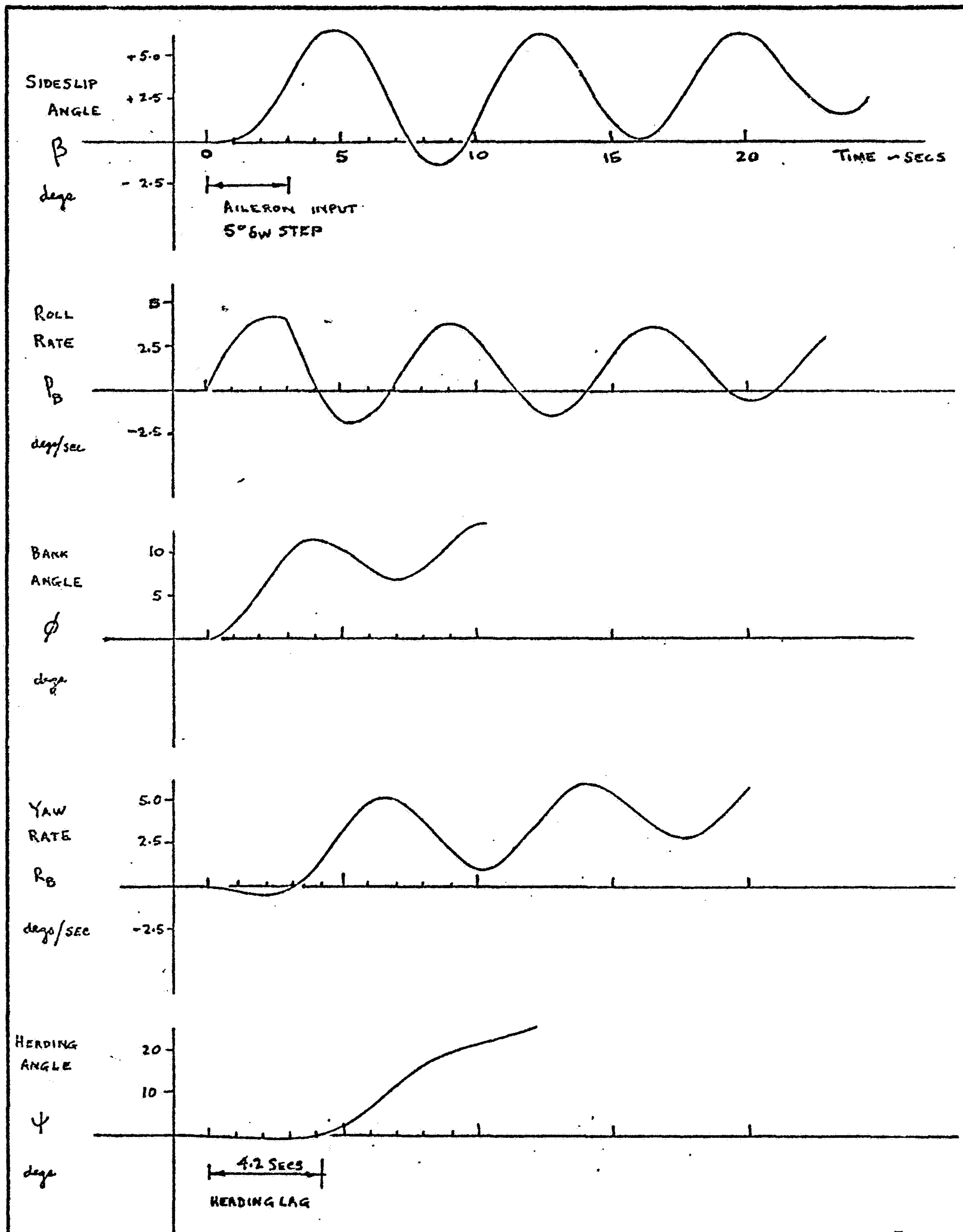

			REVISED	DATE		
CALC					RESPONSE TO 5° WHEEL STEP FOR 3 SECS	D6-24806-1
CHECK					SAS OFF ~ $C_{\ell\beta}$ = -.25	
APPD	NARK	1-11-71			STOL LANDING CONFIGURATION	FIG 4.3-2
APPD					THE BOEING COMPANY RENTON WASHINGTON	PAGE 4.28

TD 873 D-R3

UNAUGMENTED LATERAL-DIRECTIONAL DYNAMICS

FLAPS	V_e~KTS	α_F~DEG	$\frac{W}{qS}$	$C_{j_{TOT}}$	C_{l_β} PER DEG	DUTCH ROLL MODE ζ	ω_N CPS	$C_{1/2}$	DAMPED PERIOD P~SEC	$\left\|\frac{\phi}{\beta}\right\|$ APPROX	SPIRAL MODE T_2 SEC TIME-TO-DOUBLE	ROLL MODE τ_R~SEC
LANDING:												
75°	60 KTS	0°	3.8	.50	0	.25	.14	.45	7.2	.6	3.5	1.65
✓	✓	✓	✓	✓	-.004	.05	.14	2.2	7.2	✓	13	1.25
TAKEOFF												
50°	80KTS	-4°	2.1	.38	0	.25	.15	.45	6.6	.6	4	1.40
✓	✓	✓	✓	✓	-.004	.05	.15	2.2	6.6	✓	10	1.10
CRUISE												
UP	150	3.6	.6	.04	-.0026	.15	.21	.73	4.8	1.0	NEUTRAL	.77
UP	110	10	1.1	.07	-.0026	.20	.17	.55	5.7	✓	20	.83

FIG 4.3-3

4.4 EMERGENCY LANDING CONFIGURATION

The design philosophy for the modified Buffalo has generally been to provide a vehicle suitable for STOL research in moderate atmospheric turbulence. Handling qualities are expected to be good enough to allow STOL approaches to be flown with a reasonable pilot workload with all systems working and to allow safe retrieval of the airplane in the event of likely single failures. No attempt has been made to design for STOL operation with any systems not fully operational. Multiple failures in the STOL mode could well lead to unacceptable pilot workload. For this reason the philosophy has been that a single system failure of any kind would lead to termination of most testing and a final landing using an emergency,non-STOL,configuration.

The results of this simulation have re-inforced the need for identifying and using such a configuration. Airplane handling qualities in the STOL configuration with complete hydraulic failure, or with a total failure of the stability augmentation system,are so poor that STOL work should be avoided when partial failure of the hydraulic or electrical systems has occurred. The control of engine failure is so demanding that STOL landings should be avoided when any other system is inoperative. Simulated landings at 90 to 100 knots at flaps 30° have proved to be very conventional in character. This configuration was also chosen as the most acceptable for manual reversion landings. A configuration similar to this is expected to be designated as an emergency landing configuration.

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5.0 CONCLUSIONS AND RECOMMENDATIONS

5.1 DESIGN CHANGES

The simulator tests reported in this document have resulted in design changes and redefinition of requirements in certain areas of the aircraft modification program.

- Fuel system modifications are in hand to reduce airplane moments of inertia.
- Cold gas flap and aileron nozzle areas are being redefined to ensure the level of asymmetric blowing required to alleviate and control the effect of engine failures.
- Increased emphasis has been placed on safe-life design of the duct system due to the seriousness of the control problem caused by a burst duct.
- Lateral control system trim rates and wheel forces are being redefined to suit pilot preferences.
- Horizontal tail plane incidence is being defined to ensure adequate elevator for flare.
- Nozzle lever handle design is being influenced by pilot preferences.
- Stability augmentation system programming with flap angle has been defined.
- The requirement for a fast two-speed flap retraction rate has been eliminated.
- Flap placard speeds giving adequate maneuver margin have been identified.

5.2 FLIGHT TEST PROCEDURES

Testing techniques have been outlined which will help to:

- Minimize problems due to engine failures.
- Investigate possible landing flare problems in a logical, safe sequence.
- Maintain safety of operation by use of an emergency landing configuration in the event of partial system failures or other operational problems.

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5.3 RECOMMENDATIONS

5.3.1 Further Design Modifications

Based on the simulator results, it is recommended that NASA give consideration to the following design modifications:

- Possible configuration changes which would reduce the altitude lost in single engine go-arounds.
- Provision of partial rudder deflection to improve manual reversion control with two hydraulic systems failed.
- Incorporation of a powered longitudinal control system with added trim authority, tailored stick forces and provision for series SAS.
- Increase of rudder trim rate.

5.3.2 Further Simulator Testing

It is recommended that follow-on simulator investigations should include the following items:

- A check of engine-out go-around for the final choice of approach and landing flaps, including finalized flap retraction rates.
- An investigation of simple pilot cues which will produce consistent performance during engine-out go-arounds.
- An investigation of the critical height for touchdown rate of sink after an engine failure.
- An investigation of the improvement in longitudinal handling qualities that could be offered by a powered control system & trimmable horizontal stabilizer.
- A check of final SAS configuration, including failure modes and possible rudder-to-aileron interconnect.
- Takeoff in ground effects.
- Operation at 45,000 lbs. gross weight.
- Further investigation of the tendency towards pilot induced oscillations in the pitch plane.

6.0 REFERENCES

*1) DHC-DIR 69-9, Augmentor-Wing FTV Analogue Simulation, Handling and Control Study, Longitudinal Mode, J. E. Farbridge and P. Maritan, January, 1969

*2) DHC 70-4, The Augmentor-Wing Flight Test Vehicle Handling and Control Simulation, J. E. Farbridge, June, 1970

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7.0 APPENDICES

Four appendices are attached to this simulator report. The first, Appendix 7.1, is included in this volume and contains overall summary statements made by the pilots themselves.

The remaining three appendices (simulator checkout, daily logs and transcripts of pilot comments) are included in a second volume, D6-24806-2, due to their lengthy nature.

7.1 PILOTS' OVERALL SUMMARIES

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7.1.1 SUMMARY REPORT of the Augmentor-Wing Flight Test Vehicle Simulation

By

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1.0 QUALITY OF SIMULATION

1.1 PRINCIPAL ADVANTAGES

Examination of the augmentor-wing FTV on the FSAA six degree-of-freedom moving cab simulator was a significant advance on the original three degree-of-freedom moving cab simulation for the following reasons:

- Motion cues were not confusing, and seemed at all times to be in phase with visual and instrument depictions of aircraft motion.
- Cab and cockpit layout were superior in the FSAA simulator.
- Engine and nozzle controls more accurately simulate Buffalo power and propeller lever (when shortened) positions.
- Pilot's wheel and longitudinal/lateral trim button easier to use.
- Excellent control of pilot seating position.
- Better instrumention of attitudes, localizer/glideslope, with alpha and beta readouts improved.
- Sound intrusions associated with the motion systems were significantly less distracting, and could be almost entirely eliminated by appropriately setting the level of simulated turbine noise.
- Continuous intercom boom-mike arrangement made for much improved communications.

1.2 PRINCIPAL CRITICISMS

- Airspeed instrument rather crude for STOL.300 or 400 kt. drum type, with one rotation of drum per 100 kt, and 3/32" representing two knots is more precise tool.

- The simulator initial condition button, if placed within easy access of pilot's left thumb would more readily come to hand than on the right grip, which to operate, requires him to first release the nozzle or engine controls.
- The I.C., HOLD and OPERATE button lights should be within the pilot's peripheral view.

It goes without saying that the principal criticism relates to the airspeed instrument.

1.3 OVERALL COMMENTS

This simulation is the closest thing to flying that this pilot has ever experienced while groundborne, and serious model examination requires little if any of the "tongue-in-cheek" quality in pilot attitude. As a result, fairly extended periods in the cab seemed to pass very quickly. This was also due to the well prepared and conducted NASA/Boeing programme.

2.0 GENERAL HANDLING CHARACTERISTICS WITHOUT STABILITY AUGMENTATION

2.1 LATERAL/DIRECTIONAL

Without stability augmentation the simulation was very similar to the one previously studied on the 3 deg. simulator, however it did not seem quite as demanding in the approach and landing. The lower wheel forces, and lower stick force gradient decreased the pilot workload and the urgency of requirement for longitudinal trim, particularly in transitions. The unstable spiral mode and low lateral/directional damping made the steering task difficult, and avoidance of several cycles of lateral PIO when setting a desired bank angle, or when rolling back to level flight was all but impossible. As a result, the lateral control tends to be overworked when attempting to freeze the aircraft in either of these situations.

Adverse yaw was still present in gross proportions, however there seemed to be more adverse sideslip effect following the application of aileron when entering or exiting a turn than the direct effect on yaw rate which was present in the previous simulations. Along with low directional damping, all of this tended to make the pilot's hands and feet somewhat of a blur when attempting to coordinate turns at bank angles of 10 to 20 deg., and sideslip angles of up to 15 deg. were routine. With SAS-Off, a value of C_{ℓ_β} of -.2 seemed to more favorably affect lateral/directional handling qualities than when zero, this was particularly so in gusts.

2.2 LONGITUDINAL

The "springy" elevator quality was exaggerated to an extent that a longitudinal PIO was easily encountered on the approach, and during the single engine go-around, as flaps are retracted and speed is increased.

In spite of the springy elevator characteristics and longitudinal PIO, speed stability seemed improved, and fewer overspeed excursions were experienced. While vectored thrust authority seemed improved for glidepath and speed control, the lift loss accompanying nozzle excursions from 90 deg. to 18 deg. seemed to produce an initial sinking tendency.

3.0 THE INFLUENCE OF STABILITY AUGMENTATION SYSTEMS

3.1 SAS-OFF

The landing is a very demanding task and would merit a CR of 4.5 to 5. The lateral/directional characteristics are such that the landing could rapidly become marginal in anything greater than minimum gust levels, or nominal crosswinds. Unaugmented, the low spiral mode demanded excessive attention to prevent lateral/directional divergence while attempting to hold the aircraft in level flight. The low roll damping required a great deal of short-term lateral control to set and hold a desired bank angle while turning, or to precisely stop the rollout at wings level when terminating a turn, and turn coordination was virtually impossible throughout.

3.2 SAS-ON

The principal benefits of SAS seemed to focus on the extent to which it affects the use of lateral control. The effects of spiral augmentation alone was difficult to appreciate beyond some improvement of the unattended lateral/directional divergence from a wings level condition. The max roll damping gain which was used toward the end of the session did much to decrease the constant use of ailerons which was required to establish and hold any desired condition of bank. The model did not seem as sensitive to directional damping, as to lateral damping, and though turn coordination was accomplished by the SAS to acceptable levels, one tended to become a little impatient with the rate at which small directional changes could be made in the latter stages of the approach. As a result, it was difficult to resist quickening the short-term steering (aileron) inputs with rudder. This may be an infection contracted through STOL experience, however it is doubtful that the average pilot is going to be entirely satisfied with the sort of "feet on the floor" directional response which can be afforded by the best SAS simulated, while approaching a landing at 60 kt. using ailerons alone, particularly in turbulence. The lower level of lateral inertia in all cases improved lateral handling qualities, and by itself accounts for a lowering of the CR by an increment of .5.

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SAS-ON roll-due-to-yaw appeared to be neutral following rudder kicks, but showed an unstable dihedral effect following aileron release in steady side-slips. In itself this may not be particularly disturbing, since it tends to decrease the extent to which controls must be crossed in performing a wing-down (steady sideslip) crosswind approach. Otherwise, as the rudder requirement grows in the flare, positive dihedral effect would demand an increasing aileron input to prevent the upwind wing rising, which would only heighten the overall task.

4.0 NOZZLE VECTORING CONTROL

The sign and magnitude of nozzle-induced pitching moments still tend to assist in the glideslope-following task. While the longitudinal PIO does not seem to be excited by nozzle excursions, it seems more prominent in the latter stages of the approach, and in the baulked landing where there is more of a tendency to make sharp elevator step inputs. Nozzle slew rate seemed quite acceptable.

While the longer nozzle levers could be located and handled fairly easily, since they moved in a different arc from the power levers it was momentarily difficult to move from the PLs to them if the pilot did not visually re-reference them. After they were shortened, this was much improved, and the slight increase in sensitivity of the vectoring control was welcomed. In any future selection of knobs for the vectoring levers, T handles should be avoided, two hemispheres combining to make a single ball handle would seem acceptable. Stirrup handles are pleasant to use, but would seem inappropriate since the power levers are already presented in this form.

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An effect which seems to stem from the increased power level used for the approach, is the sinking tendency which occurs immediately following movement of the nozzle to the 18 deg. position when correcting underspeed errors in the final stages of the approach. It would appear that initially the aircraft responds vertically more quickly to the vectored thrust lift loss than does its speed to the change in effective thrust.

5.0 REDUCED SPEED APPROACHES

Approaches performed at 50 kts. seemed to indicate that the aircraft had greater speed stability than at 60 knots, and it appeared that the reduction in alpha margin was not as great as the reduction in speed margin. After the original. 60 knot attitude was restored by an increase in power, with nozzles set at slightly more than 90°, the available elevator was adequate for the flare.

6.0 ENGINE-OUT LANDINGS:

Since engine landings were only performed SAS-"ON", they did not seem to pose any difficulty insofar as achieving an acceptable touchdown was concerned. Following the failure, when asymmetric blowing virtually balanced the rolling moment produced by the remaining nozzle, power could be increased on the operating engine with the nozzles still at the landing approach condition of about 90 deg. Once power had been increased, the remaining nozzle could be slewed in increments which could be controlled adequately in both roll and yaw. It is here that the main criticism of the asymmetric blowing arrangement arises, since vectoring of the nozzle aft, produces a yawing moment toward the failed engine, and allows the blowing asymmetry to roll the aircraft in the opposite

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direction. While the perturbation in roll and yaw thus produced is initially perplexing, it is not difficult to limit the magnitude of nozzle excursion to manageable levels.

Unless a few degrees of bank are held against the operating engine the aircraft is prone to lateral excursions from the approach centre line, if bank angle is allowed to change sign for anything but the shortest period.

7.0 BAULKED LANDING POST-ENGINE-FAILURE

The rate of sink immediately following engine failure seemed significantly greater than in the previous simulation. As in the engine out landing, it was found that the asymmetric blowing permitted the immediate application of full power on the remaining engine. After this, the nozzle can be fairly rapidly moved to the 18 deg. position, and flap retraction commenced. Until flaps have reached 35 deg. at a speed of 75 to 80 knots, there is insufficient total thrust to increase speed and climb, while simultaneously retracting flaps. Until the flaps are fully retracted, climb performance is very sensitive to angle of attack, and extremely small errors in pitch attitude can quite seriously affect the initial height loss, and the horizontal distance required to establish the aircraft in the final climb configuration. A few degrees of bank applied toward the operating engine significantly decreases the rudder requirement during the transition from the approach, (following the failure), to the final cleaned-up climbout.

The longitudinal PIO, and the sensitivity in pitch, combine with a marginal flaps-down thrust availability, to make the one-engine baulked landing very demanding, and post-failure height loss was typically 200 to 300 feet. The overall task would merit a CR of 5. to 5.5.

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While positive dihedral effect due to sideslip seemed to most favourably affect handling qualities on the approach, a neutral value appeared the most desirable for the single engine go-around, where it decreased the amount of peak roll control required in the transition to the baulk following the failure.

8.0 INFLUENCE OF AERODYNAMIC ROLL COMPENSATION MAGNITUDE ON ENGINE FAILURE

Comparisons between cross-ducting ratios of 50/50 and 60/40, showed that the asymmetric ratio produced the smallest post-cut disturbance in roll, and was superior in terms of the magnitude and duration of lateral control required in the post-failure tasks.

9.0 TRANSITION MANOEUVRE

The following transition procedure appeared reasonable, and capitalized on the availability of thrust vectoring to the extent that only one engine power change was required. From level flight:

1. Set nozzles to 120 deg.
2. Increase N_h to 94%.
3. At 130 knots select full flap in two steps i.e., 35 & 75 deg.
4. Select lateral and directional SAS "ON".

The most desirable flap operating rate for this manoeuvre and the baulked landing would lie between 6 deg. and 4 deg. per second.

10.0 INFLUENCE OF THE SIMULATED GROUND EFFECT

From the beginning, simulated ground effects made judgement of the flare and achievement of acceptable vertical touchdown velocities rather difficult, touchdown velocities of 500 - 700 fpm being routine. This meant that more of the

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available alpha range was used in the flare attempting to mitigate touchdown velocity than would normally be considered good STOL practice, where a slightly underflared landing usually produces the most repeatable performance.

Removal of the ground effects seemed to make possible acceptable flared landings with touchdown velocities of more nominal levels. Judgement of the point at which to initiate the flare however was difficult due to visual effects which gave the impression of excessive height. This was also the case at the moment of touchdown.

11.0 FAILURE MODES

The simulated SAS hardovers did not appear to pose any real problem. The single nozzle failure at the 90 deg. position produced a situation very sensitive to symmetric changes in engine power, requiring careful consolidation of lateral and directional control with any change in engine power, or with operation of the remaining nozzle.

12.0 CONTROL WHEEL STEERING (See Section 3.3.7)

CWS, while capable of accurately holding a given bank angle, once achieved, required some learning to accurately set bank while avoiding a series of small step inputs similar to a lateral PIO. It was observed however that an increase in the lateral damping gain appeared to significantly reduce the number of steps required. To a slightly lesser extent the same applied when returning to a wings-level attitude. While holding the aircraft in a turn at a fixed bank angle the system returns the wheel to neutral, with the result that the small aileron excursions, required to fix the bank angle, must be made across the breakout range which is slightly irritating.

Short term aileron inputs give an impression of low spiral stability, as the system rolls the aircraft to a bank angle appropriate to the integrated short term roll command. This requires a little time for pilot adjustment, and demands something of a "wooden" quality in the use of lateral control.

13.0 AUTOMATIC SPEED CONTROL

As in the three degrees-of-freedom moving base simulator, the overall approach task is improved to a CR = 2.5 SAS "ON", and CR = 4.0 SAS "OFF", with the addition of auto-speed (thrust) control. The pilot is able to handle the wheel with both hands throughout the approach, and manual operation of the nozzles is only required at some point prior to completion of the flare. The system did not cause excessive slewing of the nozzles, as a result the nozzle induced pitching moments did not increase the longitudinal control task.

14.0 HIGHLIGHTS AND CONCLUSIONS

This simulation of the Augmentor-Wing FTV has highlighted several aspects of the handling and performance characteristics of the aeroplane which could bear directly upon the scope and manner of the investigations to be performed on the aircraft.

(i) Baulked Landing With An Engine-Out

Thrust derating, arising from community noise and engine-nozzle matching considerations, has given rise to marginal engine-out baulked landing capability. To an extent, that under the best of circumstances, height losses up to 300 feet before a level flight condition is reached are typical following an engine failure in the landing configuration. In addition, the handling task in the initial stages of the manoeuvre, when retracting the flaps prior to becoming established in the climb, is quite exacting as the minimal initial climb performance can only be achieved if angle-of-attack is held within a very narrow range. This implies that with the aircraft in the STOL landing configuration it should not come below 300 ft (plus some margin) unless it is over a runway of sufficient length that continuation of the approach and landing is assured. The overall task of completing the landing or baulking a landing with an engine-out would both be Cooper Rates at 5.0 and 6.0 under gusty conditions.

With one engine out the landing technique requires conscious separation of lateral and directional requirements to an extent that intuitive pilot reactions could be self defeating. This, coupled with the marginal engine-out baulked landing capability, is felt will severely restrict the range of pilots to whom the aircraft should be offered for flying.

Some pilot discussions at NASA Ames about the value of a pilot safety device such as the Yankee Extractor System, grew out of the marginal single engine go-around capability of the FTV. Upon reflection, however, one cannot help feel that, if the above recommendation concerning a 300 ft. margin over the end of the runway is acceptable, the prospect of a forced landing at 60 knots would seem more attractive.

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(ii) Other Poor Qualities Further Limiting the Range of Demonstration Pilots are:

(a) SAS "OFF" turn coordination is all but impossible.

(b) If the ground effects are as simulated touchdown velocities of 500 - 700 fpm may be routine.

(c) SAS "OFF" approach and landing in turbulent and gusty conditions would merit a CR = 4.5 to 5.0 and 6.0 in crosswinds.

(d) Nozzle excursions to 18° late in the approach appear to cause an increased sink rate due to the vectored thrust lift loss.

(e) Longitudinal static stability is weak with both engines at high power during a baulked landing with the nozzles at 18° deflection.

15.0 SOME PRELIMINARY THOUGHTS ON THE INITIAL STOL FLIGHT TEST METHODS

The selection of flap angle for early exploratory flights could be determined from protracted skips at the end of the high speed taxi trials. In order to avoid adverse ground effect initially, it is anticipated that a low flap angle (of order 30°) could be chosen with the hot nozzle thrust deflection at 18°. Then the aircraft could be landed in a relatively conventional manner with little chance of jet thrust ground impingement and with less possibility of encountering large changes in lift and drag. At this low flap angle, safe takeoff and baulked landings could then be performed with adequate climb capability in the event of engine failure.

When it comes to the demonstration of the first STOL landings, it would be highly desirable to have a reasonable understanding of any ground effect characteristics arising from deflected jet impingement. It is therefore suggested that high speed taxi runs leading to protracted runway skips could become the means by which an accurate picture of the ground effects could be realized prior to the first STOL flight.

The first skips could be undertaken at a flap angle which predicts a good trade-off of longitudinal stability and control characteristics, and a zero lift reference which would permit the aircraft to unstick after a 5 degree rotation, and fly the skip in a level fuselage attitude at approximately 1.1 to 1.3 times the unstick speed. The level fuselage reference is desirable to limit thrust changes which could result from changes in pitch attitude independent of nozzle angle.

For a given flap angle, the selected engine power should not be considered a variable once the brakes have been released. Initial acceleration should be made with the nozzles at 18°, and a 5° longitudinal rotation should be made and maintained in advance of the expected unstick speed. With this attitude held on the mainwheels, thrust to unstick and maintain vehicle speed and height above the runway should be controlled with the nozzles only. Vectored lift as a cosine effect would not normally be expected to produce sharp-edged effects in the $90^\circ \pm 30^\circ$ nozzle range, and should therefore permit initial ground effects to be encountered gradually as power levels are increased for subsequent runs. The post unstick speed range would make sufficient lift coefficient available to counter any undesirable vectored lift vertical effects with elevator.

The aircraft could be landed from the skip by slowly moving the nozzle toward 120°, and after touchdown further reverse thrust could be applied to minimize brake heating by then increasing engine power with the nozzles still set at 120°.

For each flap angle examined in this way, the initial engine power level could be such that nozzle need not be handled too carefully to permit the aircraft to become light on the wheels, or to barely become airborne during the initial run in any specific configuration. Power for each run could be increased in increments which previous runs indicated to be prudent.

Vehicle height and speed could be increased in each run as confidence was gained, at all times proceeding to the next increment from a well examined previous situation, which could be retreated to safely if any divergent trends are encountered.

In this manner the runway hops could be performed with the pilot in complete command of the aircraft while controlling thrust through one means only, which is not dependent upon engine acceleration or deceleration capability. Held in reserve at all times is the ability to reduce engine power if vectored thrust alone does not meet the pilots' needs in adequately limiting vehicle speed or height above the runway.

When a sufficient flap range had been examined in this manner, the configuration of flap offering the best combination of handling qualities and performance could then be used for the takeoff and landing on the first and subsequent initial STOL flights, following which the airborne investigations would expand the envelope and allow a broader range of configuration examination.

This approach to the initial flight testing of the FTV is principally intended as a guide in restricting the number of variables with which the pilot would have to deal in the runway hops, while giving him a maximum of indoctrination in ground effects and aircraft handling qualities prior to the actual first STOL flight in the aircraft.

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7.1.2 SUMMARY of the Augmentor-Wing Flight Test Vehicle Simulation

By

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A simulation of the Augmentor-Wing Flight Test Vehicle was flown on the Full Scale Advanced Aircraft simulator at NASA/Ames from October 27 to November 6, 1970. The comments below refer to the airplane characteristics for flap angles of 50 degrees and greater and with the Stability Augmentation System operative unless otherwise stated.

1.0 NORMAL OPERATION

1.1 PITCH AXIS

The pitch axis does not respond the same as the basic Buffalo. No static longitudinal stability appears to exist. However, the most disconcerting characteristic is the drift in pitch attitude in the direction of the initial displacement. The longitudinal control forces are twice as high as that desired. The inclusion of a powered elevator would allow the control forces to be tailored to optimum. More precise elevator control would be available and more pilot confidence in the ability to recover from high lift coefficients would exist with the addition of a trimmable stabilizer and powered elevator.

The incorporation of control wheel steering about all axes would provide a vast improvement in handling qualities and reduce the pilot workload tremendously. However, this would not make up for the lack of performance for the engine out case. The automatic speed control also reduces the pilot workload but drives the nozzles in the wrong direction after an engine failure.

After several touchdowns were made, the ground effect levels were reduced below those predicted in the wind tunnel due to inadequate flare capability in the simulation. With this change the landing flare capability is still so low that full elevator does not reduce the touchdown sink rates to acceptable levels (3-4 f.p.s.). The sink rates experienced during the normal touchdowns were of such magnitude that today's airplane pilots would consider every landing a hard landing and an engine out landing catastrophic.

1.2 LATERAL DIRECTIONAL AXES

With SAS on lateral/directional flying qualities are acceptable except that the dihedral effect is the reverse of conventional airplanes. This requires lateral and directional control operation that is opposite to what pilots normally use.

2.0 FAILURE CONDITIONS

A single failure such as SAS or an engine produces a condition that can require full control authority to maintain the desired airplane attitudes.

After an engine failure, the ability to arrest the sink rate below 200 feet altitude, to avoid contacting the ground, requires very rapid pilot reaction to reset nozzle angle, advance engine throttle and reduce flap angle.

This transition may take as much as 300 feet altitude and the use of full control authority. The recovery from an engine failure below 200 feet would not be possible and would result in a very hard landing. The loss of an engine, at an altitude that permits recovery, produces rolling pitching and yawing moments that are extremely difficult to control especially without SAS.

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3.0 RECOMMENDATIONS

Should STOL landings be required, it is recommended that the minimum altitude over the end of the runway be restricted to 300 feet to enable the pilot to, at best, reach the runway in case of engine failure.

It is recommended that a study be conducted to determine the feasibility of the installation of a crew escape system such as the Yankee Extraction System. This is due to the unacceptable handling qualities after a single failure such as SAS, an engine or the lack of climb performance without a configuration change. In any event it is recommended that the crew be limited to a maximum of two people due to the low performance and handling qualities.

Unless the performance can be improved with 50 degrees of flap or greater and the handling quality improved, the existing characteristics will not permit satisfactory demonstrations to be conducted to other than research oriented personnel.

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7.1.3 Summary of the Augmentor-Wing Flight Test Vehicle Simulation

by

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In general, the simulation of the augmentor wing on the FSAA was considered to be quite satisfactory. In addition to achieving the primary objective of obtaining design information for the modification, it provided the subject pilots with invaluable experience with the aircraft's handling characteristics, performance and control procedures in both normal and emergency flight conditions. I felt that the main deficiency of the simulator itself was the inadequacy of the depth perception cues from the visual scene between flare and touchdown. This prevented the pilots from accurately assessing the landing behavior and the severity of any adverse characteristics in ground effect. On the other hand, without the rather elaborate motion cues, I doubt if we could have obtained any realistic simulation of the various failure modes such as engine failure, SAS hardover, etc.

As far as the airplane was concerned, I felt that the handling characteristics, as simulated, were adequate to perform the mission for which the modification was intended, i.e., demonstrate the augmentor wing concept and develop the operational techniques required to control the integrated lift and propulsion system. Without stability augmentation, the handling qualities were unsatisfactory but the airplane could be safety flown and landed within the environment in which we normally expect to operate it: namely, day, VFR, and relatively smooth air. With the proposed stability augmentation system operating, the handling qualities were, in general, satisfactory and should allow us to investigate simulated instrument approaches (as long as they are simple and straight forward), performance characteristics, and some operation in turbulence and crosswinds. These handling qualities would be inadequate, however, for a commercial STOL transport because of the high workload, frequency of exposure and more complicated approach geometry expected in this type of operation. The most questionable aspects of the handling qualities of the augmented airplane concern the longitudinal axis. These include low static and dynamic stability, possible inadequate trim capability and high control forces. In view of the expanded scope of the augmentor wing flight program, it would be desirable to include a powered elevator and trimmable stabilizer in the modification schedule if at all possible. This would allow stability augmentation to be incorporated in the longitudinal axis as well as provide improved controllability.

Another area of concern was the marginal performance with one engine failed particularly during the approach or waveoff. A rather abrupt loss in normal acceleration occurs when the engine fails followed by an appreciable loss of altitude (about 200') before recovery can be effected. Even after recovery, flaps have to be retracted to the takeoff position before any significant climb gradient can be achieved. The full significance of this problem will have to be assessed during the flight tests, but we should face the fact that we may be forced to place some restrictions on the operation of the aircraft if we expect to maintain a high level of safety.